SSP 41172

Qualification and Acceptance Environmental Test Requirements

International Space Station Program

Revision U

28 March 2003

Canadian Space Agency
Agence spatiale canadienne

agenzia spaziale italiana
(Italian Space Agency)

National Aeronautics and Space Administration
International Space Station Program
Johnson Space Center
Houston, Texas

REVISION AND HISTORY PAGE

REV.	DESCRIPTION	PUB. DATE
	SRR Version (Reference: SSCBD 00002)	01/24/94
A	Updated to reflect incorporation of approved exceptions taken by the Product Groups; Added Appendices A, B, and C that documents these exceptions. Incorporates ECP 145 (SSCBD 000145, Effective 10/31/95)	11/21/95
B	Updated to reflect incorporation of approved exceptions taken by the Product Groups; Incorporates ECP 217 (SSCBD 000217, Effective 02/26/96)	04/04/96
	DCN 001 incorporates ECP 151 (SSCBD 000151, Effective 02/18/98)	03/19/98
	DCN 002 incorporates ECP 263 (SSCBD 000263, Effective 10/03/97)	03/19/98
	DCN 003 incorporates ECP 346 (SSCBD 000346, Effective 03/28/97)	03/19/98
	The following DCNs have been cancelled. The contents of the SSCNs authorizing the release of the DCNs have been incorporated into Revision C.	
	DCN 006 (SSCN 000845) (Administrative Cancel) DCN 007 (SSCN 000671) (Administrative Cancel) DCN 008 (SSCN 000877) (Administrative Cancel) DCN 009 (SSCN 000915) (Administrative Cancel) DCN 010 (SSCN 000844) (Administrative Cancel) DCN 011 (SSCN 001029) (Administrative Cancel) DCN 012 (SSCN 001030) (Administrative Cancel) DCN 013 (SSCN 001051) (Administrative Cancel) DCN 014 (SSCN 001052) (Administrative Cancel) DCN 004 (SSCN 000290) (Administrative Cancel) DCN 015 (SSCN 000823) (Administrative Cancel) DCN 016 (SSCN 000673) (Administrative Cancel) DCN 017 (SSCN 001050) (Administrative Cancel) DCN 018 (SSCN 001056) (Administrative Cancel) DCN 019 (SSCN 001392) (Administrative Cancel) DCN 020 (SSCN 000672) (Administrative Cancel) DCN 021 (SSCN 000908) (Administrative Cancel) DCN 022 (SSCN 000909) (Administrative Cancel) DCN 023 (SSCN 001048) (Administrative Cancel) DCN 024 (SSCN 000809) (Administrative Cancel) DCN 025 (SSCN 000842) (Administrative Cancel) DCN 026 (SSCN 001085) (Administrative Cancel) DCN 027 (SSCN 001198) (Administrative Cancel) DCN 028 (SSCN 001182) (Administrative Cancel) DCN 029 (SSCN 001112) (Administrative Cancel) DCN 030 (SSCN 001391) (Administrative Cancel) DCN 031 (SSCN 000585) (Administrative Cancel) DCN 032 (SSCN 001047) (Administrative Cancel) DCN 033 (SSCN 000542) (Administrative Cancel) DCN 034 (SSCN 001276) (Administrative Cancel) DCN 035 (SSCN 001338) (Administrative Cancel) DCN 036 (SSCN 001506) (Administrative Cancel) DCN 037 (SSCN 001390) (Administrative Cancel) DCN 005 (SSCN 000333) (Administrative Cancel)	

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C	Revision C, Type 2 Incorporates SSCNs 000290, 000333, 000542, 000585, 000671, 000672, 000673, 000809, 000823, 000842, 000844, 000845, 000877, 000908, 000909, 000915, 001029, 001030, 001047, 001048, 001050, 001051, 001052, 001056, 001085, 001112, 001182, 001198, 001276, 001338, 001390, 001391, 001392, 001506, and 001073. The following DCNs have been cancelled. The contents of the SSCNs authorizing the release of the DCNs have been incorporated into Revision D. DCN 038 (SSCN 001268) (Administrative cancellation) DCN 039 (SSCN 000473) (Administrative cancellation) DCN 040 (SSCN 001410) (Administrative cancellation) DCN 041 (SSCN 001724) (Administrative cancellation) DCN 042 (SSCN 000640) (Administrative cancellation) DCN 043 (SSCN 000855) (Administrative cancellation) DCN 044 (SSCN 001088) (Administrative cancellation) DCN 045 (SSCN 001379) (Administrative cancellation) DCN 046 (SSCN 001520) (Administrative cancellation) DCN 047 (SSCN 001651) (Administrative cancellation) DCN 048 (SSCN 001747) (Administrative cancellation) DCN 049 (SSCN 001802) (Administrative cancellation) DCN 050 (SSCN 001933) (Administrative cancellation) DCN 051 (SSCN 001845) (Administrative cancellation) DCN 052 (SSCN 000890) (Administrative cancellation) DCN 053 (SSCN 001703) (Administrative cancellation) DCN 054 (SSCN 002171) (Administrative cancellation) DCN 055 (SSCN 002285) (Administrative cancellation) DCN 056 (SSCN 002394) (Administrative cancellation) DCN 057 (SSCN 002395) (Administrative cancellation) DCN 058 (SSCN 002172) (Administrative cancellation) DCN 059 (SSCN 002366) (Administrative cancellation) DCN 060 (SSCN 002016) (Administrative cancellation) DCN 061 (SSCN 002522) (Administrative cancellation) DCN 062 (SSCN 002389) (Administrative cancellation)	02/25/99
D	Revision D, Type 2 Incorporates SSCNs 000473, 000640, 000855, 000890, 001088, 001268, 001379, 001410, 001520, 001651, 001703, 001724, 001747, 001802, 001845, 001933, 002016, 002171, 002172, 002285, 002366, 002389, 002394, 002395, and 002522. Revision D incorporates Non–Prime SSCNs 001200, 001250, 01306, 001348, 001350, 001382, 001487, 001595, 001628, 001763, 001851, 002013, 002157, 002201, 002373, 002374, and 002505.	10/15/99
	DCN 063 incorporates SSCD 002398 Eff. 10/26/99.	01/10/00
	The following DCNs have been cancelled. The contents of the SSCNs authorizing the release of the DCNs have been incorporated into Revision E. DCN 064 incorporates SSCN 002601 (Administrative cancellation) DCN 065 incorporates SSCN 002594 (Administrative cancellation) DCN 066 incorporates SSCN 002338 (Administrative cancellation) DCN 067 incorporates SSCN 002591 (Administrative cancellation)	

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	DCN 068 incorporates SSCN 002593 (Administrative cancellation) DCN 069 incorporates SSCN 002767 (Administrative cancellation) DCN 070 incorporates SSCN 002768 (Administrative cancellation) DCN 071 incorporates SSCN 002582 (Administrative cancellation) DCN 072 incorporates SSCN 002585 (Administrative cancellation) DCN 073 incorporates SSCN 002640 (Administrative cancellation) DCN 074 incorporates SSCN 002216 (Administrative cancellation) DCN 075 incorporates SSCN 002592 (Administrative cancellation) DCN 076 incorporates SSCN 002442 (Administrative cancellation) DCN 077 incorporates SSCN 002757 (Administrative cancellation)	
	DCN 078 incorporates SSCN 002275 (Administrative cancellation) DCN 079 incorporates SSCN 002795 (Administrative cancellation) Note: SSCNs 002275 and 002795 bring onto Prime the Non–Prime changes incorporated into Revision D.	
E	Revision E, Type 2 Incorporates SSCNs 002216, 002338, 002442, 002582, 002585, 002591, 002592, 002593, 002594, 002601, 002640, 002757, 002767, 002768, 002275, 002795, and 002620.	04/11/00
	DCN 080 (SSCN 002914) (Administrative cancellation) DCN 081 (SSCN 001586) (Administrative cancellation) DCN 082 (SSCN 002660) (Administrative cancellation) DCN 083 (SSCN 002982) (Administrative cancellation) DCN 084 (SSCN 002400) (Administrative cancellation)	
F	Revision F, Type 2 Incorporates SSCNs 001586, 002400, 002660, 002914, 002982, and 003008.	01/16/01
	DCN 085 (SSCN 002587) (Administrative cancellation) DCN 086 (SSCN 002964) (Administrative cancellation) DCN 087 (SSCN 002934) (Administrative cancellation) DCN 088 (SSCN 003255) (Administrative cancellation) DCN 089 (SSCN 003082) (Administrative cancellation) DCN 090 (SSCN 003571) (Administrative cancellation) DCN 091 (SSCN 003570) (Administrative cancellation) DCN 092 (SSCN 003569) (Administrative cancellation) DCN 093 (SSCN 003531) (Administrative cancellation) DCN 094 (SSCN 003532) (Administrative cancellation) DCN 095 (SSCN 003533) (Administrative cancellation) DCN 096 (SSCN 003682) (Administrative cancellation) DCN 097 (SSCN 003492) (Administrative cancellation) DCN 098 (SSCN 003683) (Administrative cancellation) DCN 099 (SSCN 003684) (Administrative cancellation) DCN 100 (SSCN 003511) (Administrative cancellation) DCN 101 (SSCN 003542) (Administrative cancellation) DCN 102 (SSCN 002586) (Administrative cancellation) DCN 103 (SSCN 003292) (Administrative cancellation) DCN 104 (SSCN 003293) (Administrative cancellation) DCN 105 (SSCN 003294) (Administrative cancellation) DCN 106 (SSCN 003572) (Administrative cancellation) DCN 107 (SSCN 003573) (Administrative cancellation)	

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	DCN 108 (SSCN 003642) (Administrative cancellation) DCN 109 (SSCN 003654) (Administrative cancellation) DCN 110 (SSCN 003699) (Administrative cancellation) DCN 111 (SSCN 003701) (Administrative cancellation)	
G	Revision G, Type 2 Incorporates SSCNs 002586, 002587, 002934, 002964, 003082, 003255, 003292, 003293, 003294, 003492, 003511, 003531, 003532, 003533, 003542, 003569, 003570, 003571, 003572, 003573, 003642, 003654, 003682, 003683, 003684, 003699, 003701, and 003702. DCN 112 (SSCN 002689) (Administrative Cancellation) DCN 113 (SSCN 002793) (Administrative Cancellation) DCN 114 (SSCN 003123) (Administrative Cancellation) DCN 115 (SSCN 003074) (Administrative Cancellation) DCN 116 (SSCN 003034) (Administrative Cancellation) DCN 117 (SSCN 003700) (Administrative Cancellation) DCN 118 (SSCN 004064) (Administrative Cancellation) DCN 119 (SSCN 004296) (Administrative Cancellation)	01/16/01
H	Revision H, Type 2 Incorporates SSCNs 002689, 002793, 003034, 003074, 003123, 003700, 004064, 004296, and 001637. DCN 120 (SSCN 003133) (Administrative Cancellation) DCN 121 (SSCN 003952) (Administrative Cancellation) DCN 122 (SSCN 002995) (Administrative Cancellation) DCN 123 (SSCN 003955) (Administrative Cancellation) DCN 124 (SSCN 003979) (Administrative Cancellation) DCN 125 (SSCN 003606) (Administrative Cancellation) DCN 126 (SSCN 003528) (Administrative Cancellation) DCN 127 (SSCN 003963) (Administrative Cancellation) DCN 128 (SSCN 003408) (Administrative Cancellation) DCN 129 (SSCN 003044) (Administrative Cancellation) DCN 130 (SSCN 001371) (Administrative Cancellation) DCN 131 (SSCN 003035) (Administrative Cancellation) DCN 132 (SSCN 004105) (Administrative Cancellation) DCN 133 (SSCN 001450) (Administrative Cancellation) DCN 134 (SSCN 003530) (Administrative Cancellation) DCN 135 (SSCN 004289) (Administrative Cancellation) DCN 136 (SSCN 004917) (Administrative Cancellation) DCN 137 (SSCN 004647) (Administrative Cancellation) DCN 138 (SSCN 003276) (Administrative Cancellation) DCN 139 (SSCN 003978) (Administrative Cancellation) DCN 140 (SSCN 004434) (Administrative Cancellation) DCN 141 (SSCN 003751) (Administrative Cancellation) DCN 142 (SSCN 004306) (Administrative Cancellation) DCN 143 (SSCN 004458) (Administrative Cancellation) DCN 144 (SSCN 004902) (Administrative Cancellation) DCN 145 (SSCN 003526) (Administrative Cancellation) DCN 146 (SSCN 003491) (Administrative Cancellation) DCN 147 (SSCN 004134) (Administrative Cancellation)	02/23/01

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	DCN 148 (SSCN 003887) (Administrative Cancellation) DCN 149 (SSCN 004509) (Administrative Cancellation) DCN 150 (SSCN 003599) (Administrative Cancellation)	
J	Revision J, Type 2 Incorporates SSCNs 001371, 001450, 002995, 003035, 003044, 003133, 003276, 003408, 003491, 003526, 003528, 003530, 003599, 003606, 003751, 003887, 003952, 003955, 003963, 003978, 003979, 004105, 004134, 004289, 004306, 004434, 004458, 004509, 004647, 004902, 004917, and 003920. DCN 151 (SSCN 004947) (Administrative Cancellation) DCN 152 (SSCN 005019) (Administrative Cancellation) DCN 153 (SSCN 003273) (Administrative Cancellation) DCN 154 (SSCN 004354) (Administrative Cancellation) DCN 155 (SSCN 004383) (Administrative Cancellation) DCN 156 (SSCN 002027) (Administrative Cancellation) DCN 157 (SSCN 003477) (Administrative Cancellation) DCN 158 (SSCN 004684) (Administrative Cancellation) DCN 159 (SSCN 004287) (Administrative Cancellation) DCN 160 (SSCN 005281) (Administrative Cancellation) DCN 161 (SSCN 004309) (Administrative Cancellation) DCN 162 (SSCN 003775) (Administrative Cancellation) DCN 163 (SSCN 004485) (Administrative Cancellation) DCN 164 (SSCN 003953) (Administrative Cancellation) DCN 165 (SSCN 003655) (Administrative Cancellation) DCN 166 (SSCN 003434) (Administrative Cancellation) DCN 167 (SSCN 003608) (Administrative Cancellation) DCN 168 (SSCN 003825) (Administrative Cancellation) DCN 169 (SSCN 004457) (Administrative Cancellation) DCN 170 (SSCN 004644) (Administrative Cancellation) DCN 171 (SSCN 002486) (Administrative Cancellation) DCN 172 (SSCN 003877) (Administrative Cancellation) DCN 173 (SSCN 004948) (Administrative Cancellation) DCN 174 (SSCN 005271) (Administrative Cancellation) DCN 175 (SSCN 004433) (Administrative Cancellation) DCN 176 (SSCN 003529) (Administrative Cancellation)	05–18–01
K	Revision K, Type 2 Incorporates SSCNs 002027, 002486, 003273, 003434, 003477, 003529, 003608, 003655, 003775, 003825, 003877, 003953, 004287, 004309, 004354, 004383, 004433, 004457, 004485, 004644, 004684, 004947, 004948, 005019, 005271, 005281, and 005124. DCN 177 (SSCN 003334) (Administrative Cancellation) DCN 178 (SSCN 005503) (Administrative Cancellation) DCN 179 (SSCN 003286) (Administrative Cancellation) DCN 180 (SSCN 002944) (Administrative Cancellation) DCN 181 (SSCN 002186) (Administrative Cancellation)	07–25–01
L	Revision L, Type 2 Incorporates SSCNs 002186, 002944, 003286, 003334, 005503, and 002341.	11–13–01

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	DCN 182 (SSCN 005612) (Administrative Cancellation) DCN 183 (SSCN 005727) (Administrative Cancellation) DCN 184 (SSCN 003479) (Administrative Cancellation) DCN 185 (SSCN 003493) (Administrative Cancellation) DCN 186 (SSCN 005939) (Administrative Cancellation) DCN 187 (SSCN 005896) (Administrative Cancellation)	
M	Revision M, Type 2 Incorporates SSCNs 003479, 003493, 005612, 005727, 005896, 005939, and 005321.	12/21/01
	DCN 188 (SSCN 006077) (Administrative Cancellation) DCN 189 (SSCN 005895) (Administrative Cancellation) DCN 190 (SSCN 001000) (Administrative Cancellation) DCN 191 (SSCN 006220) (Administrative Cancellation) DCN 192 (SSCN 004209) (Administrative Cancellation) DCN 193 (SSCN 006242) (Administrative Cancellation)	
N	Revision N, Type 2 Incorporates SSCNs 001000, 004209, 005895, 006077, 006220, 006242, and 004996.	04–08–02
	DCN 194 (SSCN 006347) (Administrative Cancellation) DCN 195 (SSCN 006359) (Administrative Cancellation) DCN 196 (SSCN 006438) (Administrative Cancellation) DCN 197 (SSCN 006392) (Administrative Cancellation) DCN 198 (SSCN 003517) (Administrative Cancellation) DCN 199 (SSCN 004383S) (Administrative Cancellation) DCN 200 (SSCN 003524) (Administrative Cancellation)	
P	Revision P, Type 2 Incorporates SSCNs 003517, 003524, 004383S, 006347, 006359, 006392, 006438, and 006383.	06–27–02
	DCN 201 (SSCN 006703) (Administrative Cancellation) DCN 202 (SSCN 004879) (Administrative Cancellation)	
R	Revision R, Type 2 Incorporates SSCNs 004879, 006703, and 006681.	10–07–02
	DCN 203 (SSCN 006890) (Administrative Cancellation) DCN 204 (SSCN 006243) (Administrative Cancellation) DCN 205 (SSCN 007021) (Administrative Cancellation)	
T	Revision T, Type 2 Incorporates SSCNs 006243, 006890, 007021, and 007047.	12–17–02
	DCN 206 (SSCN 005004) (Administrative Cancellation) DCN 207 (SSCN 006258) (Administrative Cancellation) DCN 208 (SSCN 007355) (Administrative Cancellation)	
U	Revision U, Type 2 Incorporates SSCNs 005004, 006258, 007355, and 007356.	04–11–03

ERU: /s/ Beth Mason 4–11–03

PREFACE

The contents of this document are intended to be consistent with the tasks and products to be prepared by the International Space Station Program Participants. The Qualification and Acceptance Environmental Test Requirements shall be implemented on all new Space Station Program contractual and internal activities and shall be included in any existing contracts through contract changes. This document is under the control of the Space Station Control Board, and any changes or revisions will be approved by the Deputy Director.

This document establishes the minimum uniform requirements for qualification and acceptance environmental tests for Components, Structures, Assemblies and Flight Elements.

NASA/ASI

INTERNATIONAL SPACE STATION PROGRAM

QUALIFICATION AND ACCEPTANCE ENVIRONMENTAL TEST REQUIREMENTS

28 MARCH 2003

Dale Thomas

For NASA DATE

Andrea Lorenzoni

For ASI DATE

NASA/CSA

INTERNATIONAL SPACE STATION PROGRAM

QUALIFICATION AND ACCEPTANCE ENVIRONMENTAL TEST REQUIREMENTS

28 MARCH 2003

Dale Thomas

For NASA DATE

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For CSA DATE

Agreed to in principal subject to completion of detailed review by CSA and its contractor.

NASA/ESA

INTERNATIONAL SPACE STATION PROGRAM

QUALIFICATION AND ACCEPTANCE ENVIRONMENTAL TEST REQUIREMENTS

28 MARCH 2003

Dale Thomas

For NASA DATE

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NASA/NASDA

INTERNATIONAL SPACE STATION PROGRAM

QUALIFICATION AND ACCEPTANCE ENVIRONMENTAL TEST REQUIREMENTS

28 MARCH 2003

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For NASDA Concurrence DATE

NASA/RSA

INTERNATIONAL SPACE STATION PROGRAM

QUALIFICATION AND ACCEPTANCE ENVIRONMENTAL TEST REQUIREMENTS

28 MARCH 2003

Dale Thomas

For NASA DATE

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For RSA DATE

INTERNATIONAL SPACE STATION PROGRAM

QUALIFICATION AND ACCEPTANCE ENVIRONMENTAL TEST REQUIREMENTS

28 MARCH 2003

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INTERNATIONAL SPACE STATION PROGRAM

QUALIFICATION AND ACCEPTANCE ENVIRONMENTAL TEST REQUIREMENTS

LIST OF CHANGES
28 MARCH 2003

All changes to paragraphs, tables, and figures in this document are shown below:

SSCBD	ENTRY DATE	PARAGRAPH	TITLE
5004	03/28/03	3.1	Requirement for Test
		4.2.11	Leak Test, Component Qualification
		4.2.11.1	Purpose
		4.2.11.2	Test Description and Alternatives
		4.2.11.3	Test Levels and Duration
		4.2.11.4	Supplementary Requirements
		4.4	Flight Element Qualification Test
		4.4.4.1	Purpose
		4.4.4.2	Test Description
		4.4.4.3	Test Levels and Duration
		4.4.4.4	Supplementary Requirements
		5.1.7	Leak Test, Component Acceptance
		5.1.7.1	Purpose
		5.1.7.2	Test Descriptions and Alternatives
		5.1.7.3	Test Levels and Duration
		5.1.7.4	Supplementary Requirements
		5.2	Flight Element Acceptance Test
		5.2.3.1	Purpose
		5.2.3.2	Test Description
		5.2.3.3	Test Levels and Duration
		5.2.3.4	Supplementary Requirements
		10.2	Definitions
6258		4.2.11.3	Test Levels and Duration
		4.4.4.3	Test Levels and Duration
		5.1.7.3	Test Levels and Duration
		5.2.3.3	Test Levels and Duration
7355		4.2.2.5.2	Automated Power Down
7356		4.2.2.5.2	Automated Power Down

TABLE(S)

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		Table 5–2	Flight Element Acceptance Tests

FIGURE(S)

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1.0 INTRODUCTION

1.1 PURPOSE

This document establishes the minimum uniform requirements for qualification and acceptance environmental tests for components, structures, and flight elements.

1.2 SCOPE

These test requirements are applicable to the International Space Station (ISS) with the exception of the Attached Pressurized Module segment, the Russian segment, and the Japanese Experiment Module segment. These segments shall follow the testing standards of the following agencies: European Space Agency, Russian Space Agency, and National Space Development Agency of Japan. However, it is expected that these standards meet or exceed the standards in this document. These test requirements specify component, structures, and flight element qualification tests; component and flight element acceptance tests; and component, structures, and flight element protoflight tests. Test procedures and margins are specified also. Any deviation to this document shall be documented and processed in accordance with the applicable contract requirements.

1.3 PRECEDENCE

The verification requirements contained herein shall take precedence over any conflicting verification requirements, with the exception of the ISS System, Segment and End Item Specifications.

1.4 DELEGATION OF AUTHORITY

Where provisions are provided in this document for alternative strategies or requirements requiring ISS Program approval, the ISS Test and Verification Control Panel (T&VCP) shall be the designated approving authority.

2.0 APPLICABLE DOCUMENTS

The following applicable documents of the exact issue shown in the current issue of SSP 50257 form a part of this specification to the extent specified herein. Inclusion of applicable documents herein does not in any way supersede the order of precedence identified in 1.3. The references show where each applicable document is cited in this document.

DOCUMENT NO.	TITLE
SSP 30223	Problem Reporting and Corrective Action for the Space Station Program References: 8.1
SSP 30234	Instructions for Preparation of Failure Modes and Effect Analysis and Critical Items List for Space Station References: 3.1, 9.0
SSP 30237	Space Station Electromagnetic Emission and Susceptibility Requirements for Electromagnetic Compatibility References: 4.2.12.2, 4.2.12.3
SSP 30238	Space Station Electromagnetic Techniques References: 4.2.1.12.2
SSP 30243	Space Station Systems Requirements for Electromagnetic Compatibility References: 4.4.2.2
SSP 30559	Space Station Structural Design and Verification Requirements References: 4.2.10.3B, 4.2.10.3C, 4.2.13.1, 4.3.1.2, 4.3.1.4, 4.4.4.3, 5.1.9.2, 6.1.2
JSC 20584	Spacecraft Maximum Allowable Concentrations for Airborne Contaminants References: 5.2.4.4
MIL–STD–1474	Noise Limits for Army Material References: 5.2.5.2
MSFC–PROC–404	Procedure Gases, Drying and Preservation Cleanliness Level and Inspection Methods References: 5.1.9.2
NSTS 21000–IDD–ISS	International Space Station Interface Definition Document References: 4.2.5.3A, 4.2.6.3, 4.4.3.3, 5.1.4.3A, 5.1.5.3, 6.1.1C, 6.2.1

3.0 GENERAL

3.1 REQUIREMENT FOR TEST

All qualification, acceptance, and protoflight tests shall, as a minimum, comply with the test requirements of this document. If a requirement exists in another document (e.g., the applicable development specification) to perform a test that is not identified in this document as a required test, then the test shall be performed in the manner specified in this document. Hardware classified as criticality 1 or 2 from Failure Modes and Effects Analysis (FMEA) performed in accordance with SSP 30234 shall be tested in accordance with sections 4, 5 or, for protoflight, 6 of this document. Hardware classified as criticality 3 based on an integrated FMEA assessment by the Reliability and Maintainability Panel or its designee may be tested in accordance with section 9 of this document.

The objective of these requirements is to specify reasonable, prudent, and technically meaningful tests. Planned tests shall be evaluated to assure they meet these criteria.

The tests specified herein should only be conducted by personnel formally trained to conduct the applicable test(s). It is recommended that formal certification of personnel knowledge and skills for conducting the tests specified herein be maintained.

3.2 TEST CONDITION TOLERANCES

Test condition tolerances shall be applied to the nominal test values specified in this document. Unless otherwise specified, the following maximum allowable tolerances shown in Table 3–1 on test conditions shall apply:

TABLE 3–1 MAXIMUM ALLOWABLE TOLERANCES ON TEST CONDITIONS (PAGE 1 OF 2)

Temperature	+/– 5.4 degrees F (+/– 3.0 degrees C)
Pressure	
Above 133 pascals (Pa) (>1 Torr)	+/– 10 percent
0.133 to 133 Pa (0.001 to 1 Torr)	+/– 25 percent
Less than 0.133 Pa (<0.001 Torr)	+/– 80 percent
Relative Humidity	+/– 5 percent
Acceleration	+/– 10 percent
Vibration Frequency	
25 Hz and above	+/– 2 percent
Below 25 Hz	+/– 1/4 Hz
Sinusoidal Vibration Amplitude	+/– 10 percent
Static Load	+/– 5 percent
Time	+/– 2 percent
Random Vibration Power Spectral Density	
20 to 500 Hz (25 Hz or narrower)	+/– 1.5 dB
500 to 2000 Hz (50 Hz or narrower)	+/– 3.0 dB
Random overall grms	+/– 1.5 dB
Duration	+10/–1 percent

TABLE 3–1 MAXIMUM ALLOWABLE TOLERANCES ON TEST CONDITIONS (PAGE 2 OF 2)

Sound Pressure Levels (1)	
1/3–Octave Midband Frequencies	
31.5 to 40 Hz	+/– 5.0 dB
50 to 2000 Hz	+/– 3.0 dB
2500 to 10000 Hz	+/– 4.0 dB
Overall	+/– 1.5 dB
Duration	+10/–1 percent
Shock Response Spectrum (Peak Acceleration, Q = 10) (2)	
Natural Frequencies Spaced at 1/6–Octave Intervals	
At or below 5000 Hz	+/– 6.0 dB
Above 5000 Hz	+9.0/–6.0 dB

Notes:
(1) The statistical degrees of freedom shall be at least 100.
(2) At least 50 percent of the spectrum values must be greater than the nominal test specification.

4.0 QUALIFICATION TESTING

This paragraph provides the minimum requirements for qualification tests and the application of those tests to components, structures, and flight elements.

4.1 ACCEPTANCE AND QUALIFICATION RELATIONSHIP

Qualification test levels and duration shall in all cases envelope worst–case service life environments including acceptance test levels and duration (including test tolerances) and accommodate acceptance retesting. Acceptance testing, including functional and environmental, shall be conducted on all test articles prior to or in conjunction with qualification tests. See appendix C, PG3–81, for the exception to this paragraph.

4.2 COMPONENT QUALIFICATION

The component qualification tests that are a Program requirement are designated in Table 4–1 according to test category and component category. Where components fall into more than one category, the required tests for each category shall be applied. Subsequent paragraphs describe in detail the requirements for each test category. The word ”required” means that, as a minimum, the component is required to be tested if the subject environment is experienced during the component's life cycle. See appendix A, PG1–221, for exception to this paragraph. See appendix C, PG3–77, for the exception to this paragraph.

TABLE 4–1 COMPONENT QUALIFICATION TESTS (PAGE 1 OF 2)

Test	Electronic or Electrical Equipment	Antennas	Moving Mechanical Assembly	Solar Panel	Batteries	Fluid or Propulsion Equipment	Pressure Vessels	Thermal Equipment	Optical Equipment
Functional (1)	R	R	R	R	R	R	R	R	R
Thermal Vacuum (4)	R	R	R	R	R	R	–	R	R
Thermal Cycling	R	R	–	R	–	R	–	R(8)	R
Depress/Repress (5)	R	–	R	–	–	R	R	R(9)	R
Sinusoidal Vibration	–	–	–	–	–	–	–	–	–
Random (3) Vibration	R	R	R	–	R	R	R	R	R
Acoustic Vibration	R(3)	R(3)	–	R	–	–	–	–	–
Pyro Shock	R	–	R	–	R	R	–	R	R
Acceleration	–	–	–	–	–	–	–	–	–
Humidity	–	–	–	–	–	–	–	–	–
Pressure	–	–	–	–	R(2)	R	R	–	–
Leak	R(2)	–	–	–	R(2)	R	R	R	R
EMI/EMC	R	–	–	–	–	–	–	–	–
Life	–	–	–	–	–	–	–	–	–

TABLE 4–1 COMPONENT QUALIFICATION TESTS (PAGE 2 OF 2)

Test	Electronic or Electrical Equipment	Antennas	Moving Mechanical Assembly	Solar Panel	Batteries	Fluid or Propulsion Equipment	Pressure Vessels	Thermal Equipment	Optical Equipment
Corona (6) (7)	R	–	–	–	–	–	–	–	–

LEGEND: R = REQUIRED – The ISS requires as a minimum that the article be tested if the subject environment is experienced during the article's life cycle.

Notes:
(1) Functional tests shall be conducted prior to and following environmental test.
(2) Required only on sealed or pressurized equipment.
(3) Either random vibration or acoustic vibration test required with the other optional.
(4) External components only.
(5) Internal components only.
(6) Corona testing is not required for components with a sealed chassis or components which are powered on and operating under space vacuum conditions only.
(7) See Table 4–1a for component voltage criteria dictating corona testing.
(8) Thermal Cycling shall not be required for passive thermal equipment.
(9) A depress/repress test is not required if ultimate pressure testing provides a more severe differential pressure across the unit.

TABLE 4–1a CORONA TEST VOLTAGE CRITERIA

Peak Voltage (V) (1)	Transient Duration (microseconds)	Corona Testing Required
< 150	N/A	N
150 <= V < 190	< 250	N
150 <= V < 190	>= 250	Y
>= 190	N/A	Y

Notes:
(1) Applies to both steady–state and transient conditions. Also applies to input voltages, internal voltages, and output voltages.

4.2.1 FUNCTIONAL TEST, COMPONENT QUALIFICATION

4.2.1.1 PURPOSE

This test verifies that the functional performance of the component meets the operational requirements of the component specification (i.e. optical, thermal, etc.).

4.2.1.2 TEST DESCRIPTION

Electrical tests shall include application of expected voltages, impedance, frequencies, pulses, and waveforms at the electrical interfaces of the component, including all redundant circuits. Mechanical tests shall include application of torque, load, and motion as appropriate. These parameters shall be varied throughout their specification ranges and the sequences expected during its life cycle, and the component output shall be measured to verify the component performance to specification requirements. Functional performance shall also include electrical continuity, stability, response time, alignment, pressure, leakage, or other special functional tests related to a particular component configuration.

4.2.1.3 SUPPLEMENTARY REQUIREMENTS

A functional test shall be conducted prior to each of the environmental tests to assure that performance meets the requirements of the particular specification. The same functional test shall be conducted after each environmental test. Functional tests shall not be required after application of ultimate loads during a structural loads test.

4.2.2 THERMAL VACUUM TEST, COMPONENT QUALIFICATION

4.2.2.1 PURPOSE

This test demonstrates the ability of the component to perform in a thermal vacuum environment that simulates the maximum and minimum predicted level temperature environment for the component.

4.2.2.2 TEST DESCRIPTION

The component shall be mounted in a vacuum chamber on a thermally controlled heat sink or in a manner to simulate the flight environment.

With the chamber at the test–pressure level, radio frequency equipment shall be monitored to assure that multipacting does not occur. A temperature cycle begins with the chamber and component at ambient temperature. The temperature of the chamber is reduced to bring the component to the specified low qualification level and stabilized. Temperature stability has been achieved when the rate of change is no more than 5.4 degrees F/hour (3 degrees C/hour).

Components that operate on orbit shall be turned off, then started after a soak period sufficient to ensure the component internal temperature has stabilized at the specified temperature, and then functionally tested. With the component operating, the temperature of the chamber is increased to bring the component qualification temperature to the upper temperature level. After the component temperature has stabilized at the specified level, the component shall be turned off, then started after electrical circuits have been discharged. The temperature of the chamber and component shall then be reduced to ambient conditions. This constitutes one complete temperature cycle.

See appendix A, PG1–77 and PG1–87, for the exceptions to this paragraph. See appendix C, PG3–46, for the exception to this paragraph.

4.2.2.3 TEST LEVELS AND DURATION

A. Pressure. The pressure shall be reduced from atmospheric to below 0.0001 Torr (0.0133 Pa). See appendix A, PG1–190, for the exception to this paragraph. See appendix B, PG2–109 and PG2–143, for exceptions to this paragraph. See appendix D, GFE–75, for the exception to this paragraph.

B. Temperature. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle. The minimum limits shall be below 30 degrees F (–1.1 degrees C) where possible. See Figure 4–1. For electrical/electronic equipment where the minimum sweep (paragraph 5.1.3.3B) does not encompass the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F. See appendix A, PG1–75, PG1–172, PG1–183, PG1–209, and PG1–219, for the exceptions to this paragraph. See appendix B, PG2–39, PG2–40, PG2–59, PG2–65, PG2–82, PG2–86, PG2–87, PG2–89, PG2–98, PG2–120, PG2–133, and PG2–134, for the exceptions to this paragraph. See appendix C, PG3–84, PG3–100, and PG3–195, for the exceptions to this paragraph. See appendix D, GFE–101 and GFE–113, for the exceptions to this paragraph.

C. Duration. A minimum of three temperature cycles shall be used. During each cycle, the component shall undergo a dwell period after stabilization at both the high and low temperature extremes. The dwell period shall be long enough for the component to reach internal thermal equilibrium for not less than 1 hour. The time required to reach thermal equilibrium shall be determined by pre–qualification analysis or test or by measuring the component's internal thermal response during an extended dwell period of not less than 12 hours at each temperature extreme of the first qualification thermal vacuum cycle. See appendix A, PG1–152, PG1–158, PG1–182, and PG1–235, for the exceptions to this paragraph. See appendix B, PG2–78, PG2–109, and PG2–148, for the exceptions to this paragraph. See appendix C, PG3–82, for the exception to this paragraph. See appendix D, GFE–08 and GFE–49, for the exceptions to this paragraph.
NOTE: See appendix A, PG1–77, PG1–83, PG1–85, and PG1–88, for exceptions existing prior to the current requirements.

See appendix A, PG1–63, PG1–73, PG1–93, PG1–133, PG1–170, PG1–186, PG1–194, PG1–222, PG1–223, PG1–256, PG1–257, and PG1–270, for the exceptions to these paragraphs. See appendix B, PG2–61, PG2–67, PG2–70, PG2–73, PG2–81, PG2–90, PG2–126, and PG2–138, for the exceptions to these paragraphs. See appendix C, PG3–50, for the exception to these paragraphs.

4.2.2.4 SUPPLEMENTARY REQUIREMENTS

Functional tests shall be conducted, as a minimum, at the maximum operating temperature plus 20 degrees F (11.1 degrees C) and at the minimum operating temperature minus 20 degrees F (11.1 degrees C) during the first and last operating cycle after the dwell and after return of the component to ambient temperature. Dwell periods at the maximum predicted temperature may be performed with power on or off. During the remainder of the test within the component's operating temperature range, electrical and electronic components, including redundant circuits and paths, shall be monitored for failure and intermittent performance. See appendix A, PG1–63, PG1–133, PG1–170, PG1–186, PG1–189, PG1–194, PG1–222, PG1–223, PG1–254, PG1–256, and PG1–257, for the exceptions to this paragraph. See appendix B, PG2–59, PG2–61, PG2–67, PG2–70, PG2–73, PG2–81, PG2–90, PG2–124, PG2–126, and PG2–138, for the exceptions to this paragraph. See appendix D, GFE–89, for the exception to this paragraph.

NOTE: See appendix A, PG1–73 and PG1–93, for exceptions existing prior to the current requirements. See appendix C, PG3–50, for exceptions existing prior to the current requirements.

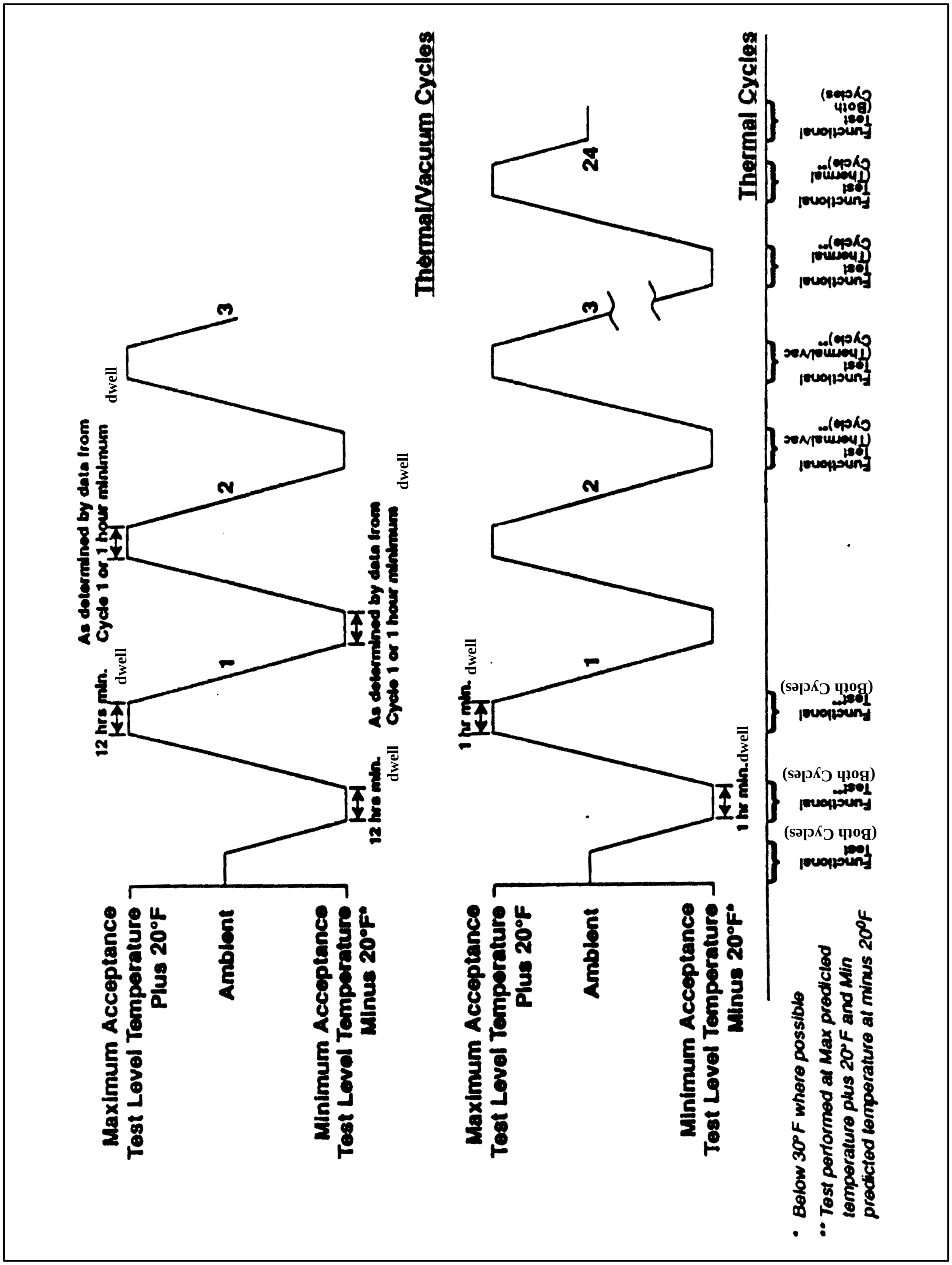


FIGURE 4–1 COMPONENT THERMAL/VACUUM QUALIFICATION TESTS

4.2.2.5 DEPRESS/REPRESS VACUUM REQUIREMENTS

Internal components shall be subjected to a depressurization and repressurization test in accordance with either 4.2.2.5.1 or 4.2.2.5.2. A thermal vacuum qualification test in accordance with 4.2.2.1 through 4.2.2.4 may be substituted for this depressurization/repressurization qualification test. See appendix C, PG3–50 and PG3–116, for the exceptions to this paragraph. See appendix D, GFE–02, GFE–41, and GFE–103, for the exceptions to this paragraph.

4.2.2.5.1 OPERATING

The following test shall be conducted on components that operate in the event of a depressurization event.

The component shall be mounted in a vacuum chamber in a manner to simulate the on–orbit environment. With the component powered up, the chamber pressure shall be reduced from ambient to the specified lowest level. The component shall be allowed to stabilize at the expected temperature and maintained for a minimum of 12 hours. The chamber shall be returned to ambient conditions. A functional test shall be conducted both prior to and after completion of the vacuum test.

See appendix C, PG3–68, for the exception to these paragraphs.

4.2.2.5.2 AUTOMATED POWER DOWN

The following test shall be conducted on components that are automatically powered down in the event of a depressurization event.

The component shall be mounted in a vacuum chamber in a manner to simulate the on–orbit environment. The chamber pressure shall be reduced from ambient to the specified lowest level. The component shall be operated until its minimum operating pressure is reached, and then it shall be powered down. The component shall be held at the specified low pressure level for a sufficient duration to ensure any outgassing or internal pressure leakage has stabilized. As the chamber is being re–pressurized, the component shall be powered on when its minimum operating pressure level is reached within the chamber. The chamber shall be returned to ambient conditions. A functional test shall be conducted both prior to and after completion of the vacuum test.

See appendix C, PG3–69, for the exception to these paragraphs. See appendix D, GFE–115 and GFE–116, for the exceptions to these paragraphs.

4.2.3 THERMAL CYCLING TEST, COMPONENT QUALIFICATION

4.2.3.1 PURPOSE

This test demonstrates the ability of components to operate over the design temperature range and to survive the thermal cycling screening test imposed upon the component during acceptance testing.

4.2.3.2 TEST DESCRIPTION

A thermal cycle begins with the component at ambient temperature. With the component operating (power on) and while parameters are being monitored, the chamber temperature shall be reduced to bring the component to the specified low qualification temperature level as measured at a representative location on the component such as the mounting point on the baseplate for conduction–dominated internal designs or at a representative location(s) on the case for radiation–controlled designs. The component shall be powered off and stabilized at the cold temperature. The component shall then be started after a dwell period sufficient to ensure that component internal thermal equilibrium has been achieved. With the component operating, the chamber temperature shall be increased to bring the component to the upper qualification temperature level. After the component temperature has stabilized at the specified level, the component may be powered off or remain with power on. The component shall then undergo a dwell period sufficient to achieve internal thermal equilibrium. The component shall then be powered off (if not already so) and then powered back on after electrical circuits have discharged. The temperature of the chamber and component shall then be reduced to ambient conditions. This constitutes one thermal cycle. See appendix B, PG2–57 and PG2–146, for the exceptions to this paragraph.

NOTE: See appendix A, PG1–89, for the exception existing prior to the current requirements.

4.2.3.3 TEST LEVELS AND DURATION

A. Pressure. Ambient pressure is normally used; however, the thermal cycle test may be conducted at reduced pressure, including vacuum conditions. When unsealed components are tested, the chamber shall be flooded with dry air or nitrogen to preclude condensation on and within the component at low temperature.

B. Temperature. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle. The minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible. See Figure 4–1. For electrical/electronic equipment where the minimum sweep (paragraph 5.1.3.3–2) does not encompass by the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F. See appendix A, PG1–75, PG1–139, PG1–168, PG1–173, PG1–205, and PG1–231, for the exceptions to this paragraph. See appendix B, PG2–39, PG2–40, PG2–57, PG2–60, PG2–63, PG2–65, PG2–72, PG2–82, PG2–87, PG2–99, PG2–120, and PG2–132, for the exceptions to this paragraph. See appendix C, PG3–93, PG3–100, PG3–109, PG3–130, PG3–142, PG3–148, PG3–152, PG3–157, PG3–172, PG3–178, PG3–192, PG3–207, PG3–210, PG3–221, and PG3–225, for the exceptions to this paragraph. See appendix D, GFE–14, GFE–37, GFE–56, GFE–60, GFE–61, GFE–67, GFE–96, GFE–101, GFE–102, and GFE–104, for the exceptions to this paragraph.

C. Duration. The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total. If Thermal Vacuum (T/V) cycling temperatures encompass those required for thermal cycling, these cycles may be included in the required 24. This test may be performed in thermal vacuum and combined with the thermal vacuum test of paragraph 4.2.2, provided that the temperature limits, number of cycles, rate of temperature change, and dwell times conform to this test. Each cycle shall have a one hour minimum dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on. The dwell time at the high and low levels shall be long enough to obtain internal thermal equilibrium. The transitions shall be at a rate no less than 1.0 degree F (0.56 degrees C) per minute. See appendix A, PG1–74, PG1–90, PG1–98, and PG1–158 for the exceptions to this paragraph. See appendix B, PG2–50, PG2–52, PG2–57, PG2–78, and PG2–148, for the exceptions to this paragraph. See appendix C, PG3–44, PG3–48, PG3–54, PG3–115, PG3–153, and PG3–161, for the exceptions to this paragraph. See appendix D, GFE–09, GFE–17, GFE–50, GFE–87, and GFE–104, for the exceptions to this paragraph.

See appendix A, PG1–170 and PG1–218, for the exceptions to these paragraphs. See appendix C, PG3–57, for the exception to these paragraphs. See appendix D, GFE–03, GFE–24, GFE–32, GFE–42, GFE–91, and GFE–111, for the exceptions to these paragraphs.

4.2.3.4 SUPPLEMENTARY REQUIREMENTS

Functional tests shall be conducted, as a minimum, at the maximum predicted temperature plus 20 degrees F (11.1 degrees C) and at the minimum predicted temperature minus 20 degrees F (11.1 degrees C) during the first and last thermal cycles and after return of the component to ambient. During the remainder of the test, electrical components, including all redundant circuits, shall be cycled through various operational modes and parameters monitored for failures and intermittences. See appendix A, PG1–99, PG1–170, PG1–218, and PG1–227, for the exceptions to this paragraph. See appendix B, PG2–57, PG2–60, PG2–63, and PG2–124, for the exceptions to this paragraph. See appendix D, GFE–15, GFE–24, GFE–32, GFE–38, GFE–91, GFE–105, and GFE–111, for the exceptions to this paragraph.

4.2.4 SINUSOIDAL VIBRATION TEST, COMPONENT QUALIFICATION

4.2.4.1 PURPOSE

Sine wave vibration testing is used for one or more of the following purposes:

A. To demonstrate the ability of the component to withstand or, if appropriate, to operate at the design levels of the sinusoidal or decaying sinusoidal–type dynamic vibration environment specified for the component.

B. To determine any resonant conditions which could result in failure in flight or in subsequent vibration tests.

C. To evaluate fixtures.

D. To conduct diagnostic testing.

4.2.4.2 TEST DESCRIPTION

The component shall be mounted to a fixture through the normal mounting points of the component. The component shall be tested in each of three mutually perpendicular axes. Significant resonant frequencies shall be noted and recorded. The induced cross–axis accelerations at the attach points should be limited to the maximum test levels specified for the cross axes.

4.2.4.3 TEST LEVELS AND DURATION

Tests conducted to determine resonant conditions or to evaluate fixtures (purposes 4.2.4.1B, C, or D) shall be conducted using test levels and durations sufficient to provide diagnostic capability. Sinusoidal excitation may be applied as a dwell at discrete frequencies or as a frequency sweep with the frequency varying at a logarithmic rate. The sweep rate for diagnostic tests shall be slow enough to allow identification of significant resonances. Tests conducted to demonstrate the degree of ruggedness (purpose 4.2.4.1A) shall use two minutes per octave unless the sweep rates and dwell times can be based on the persistence of the environment in service use. The vibration levels shall be sufficient to cover the severity of the maximum design levels.

4.2.4.4 SUPPLEMENTARY REQUIREMENTS

Sinusoidal vibration tests to determine the degree of ruggedness (purpose 4.2.4.1A) shall be considered a required test where significant sinusoidal vibration is expected in service usage. A functional test shall be conducted before the sinusoidal vibration test and after its completion. Electrical/electronic components shall be powered during the test, and their parameters monitored for failures or intermittences. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

4.2.5 RANDOM VIBRATION TEST, COMPONENT QUALIFICATION

4.2.5.1 PURPOSE

This test demonstrates the ability of the component to withstand the stresses and accumulated fatigue damage resulting from the maximum random vibration environment.

4.2.5.2 TEST DESCRIPTION

The component shall be mounted to a rigid fixture through the normal mounting point of the component. The component shall be tested in each of three mutually perpendicular axes. Valves shall be pressurized to operating pressure for this test and monitored for internal pressure decay if pressurized during ascent. See appendix A, PG1–242, for the exception to this paragraph. See appendix B, PG2–105, for the exception to this paragraph.

4.2.5.3 TEST LEVELS AND DURATION

The test duration in each of the three orthogonal axes shall be three times the expected flight exposure time to the maximum predicted level and spectrum or three times the component random vibration acceptance test time if that is greater, but not less than three minutes per axis. Component random vibration test levels and spectrums shall be the envelope of the following:

A. The maximum predicted flight level and spectrum, but not less than a level derived from an acoustic environment of 141 dB (whose spectrum is defined by NSTS 21000–IDD–ISS, Table 4.1.1.5–1).

B. Acceptance test levels and spectrum plus test tolerances. See appendix A, PG1–138, PG1–181, PG1–196, and PG1–242, for the exceptions to this requirement. See appendix B, PG2–100 and PG2–105, for the exceptions to this requiement. See appendix C, PG3–43, PG3–78, PG3–79, PG3–80, PG3–106, and PG3–145, for the exceptions to this requirement. See appendix D, GFE–20, GFE–58, and GFE–59, for the exceptions to this requirement.

See appendix A, PG1–136, PG1–147, PG1–170, PG1–234, and PG1–236, for the exceptions to these paragraphs. See appendix B, PG2–44, PG2–64, PG2–74, PG2–76, PG2–111, PG2–123, and PG2–139, for the exceptions to these paragraphs. See appendix C, PG3–59, PG3–70, PG3–90, PG3–97, PG3–121, and PG3–197, for the exceptions to these paragraphs. See appendix D, GFE–04, GFE–19, GFE–26, GFE–34, GFE–43, GFE–57, GFE–64, GFE–66, GFE–80, and GFE–100, for the exceptions to these paragraphs.

4.2.5.4 SUPPLEMENTARY REQUIREMENTS

Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test. For components mounted on isolators in the flight vehicle, the qualification random vibration test shall be assessed to determine whether the component shall be mounted on the isolators or hard–mounted to the shaker during the test.

Qualification test input to the component shall in all cases envelope acceptance test levels plus test tolerances. Where hard–mounting during qualification random vibration testing is required, a second qualification random vibration test may be required to demonstrate the adequacy of the isolator design.

See appendix A, PG1–147, PG1–170, PG1–229, and PG1–244, for the exceptions to these paragraphs. See appendix B, PG2–54, PG2–76, PG2–111, PG2–121, and PG2–139, for the exceptions to these paragraphs. See appendix C, PG3–52, PG3–59, PG3–64, PG3–65, PG3–70, PG3–71, PG3–72, PG3–73, PG3–86, PG3–88, PG3–95, PG3–104, PG3–112, PG3–122, PG3–132, PG3–138, PG3–143, PG3–159, PG3–173, PG3–179, PG3–193, PG3–205, PG3–208, and PG3–223, for the exceptions to these paragraphs. See appendix D, GFE–04, GFE–21, GFE–26, GFE–34, GFE–43, GFE–52, GFE–77, and GFE–80, for the exceptions to these paragraphs.

4.2.6 ACOUSTIC VIBRATION TEST, COMPONENT QUALIFICATION

4.2.6.1 PURPOSE

This test demonstrates the ability of the component to withstand the design–level acoustic environment. Acoustic tests shall be conducted when the analysis defines this environment, rather than random vibration, to be the worst–case condition, or if random vibration is impractical because of the components size and weight. An acoustic test may be conducted provided that the test can excite the component to the appropriate vibration levels.

4.2.6.2 TEST DESCRIPTION

The component shall be installed in a reverberant acoustic cell capable of generating desired sound pressure levels. A uniform sound energy density throughout the chamber is desired. The configuration of the component during launch, such as deployed or stowed, shall be as it is during subjection to the flight dynamic environment. The preferred method of testing shall be with the component mounted on flight–type support structure and with ground–handling equipment removed.

4.2.6.3 TEST LEVELS AND DURATION

The sound pressure level shall be at least the maximum predicted flight level and spectrum, but not less than a level derived from an acoustic environment of 141 dB overall, (whose spectrum is defined by NSTS 21000–IDD–ISS, paragraph 4.1.1.5). The duration shall be three times the expected flight exposure time to the maximum predicted environment or three times the acoustic acceptance test duration, whichever is greater, but not less than three minutes.

4.2.6.4 SUPPLEMENTARY REQUIREMENTS

A functional test shall be conducted before and following the acoustic test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. Components characterized by large ratios of surface area to volume, such as large antennas and solar arrays, cannot be tested in a manner that suitably simulates imposition of the service dynamic environment by employing mechanical vibration. For such component configurations, acoustic testing shall be required. When acoustic component testing is required, random vibration testing shall not be required.

4.2.7 PYROTECHNIC SHOCK TEST, COMPONENT QUALIFICATION

4.2.7.1 PURPOSE

This test demonstrates the capability of the component to withstand the maximum predicted shock environment.

4.2.7.2 TEST DESCRIPTION

The component shall be mounted to a fixture through the normal mounting point of the component. The selected test method shall be capable of meeting the required shock spectrum with a transient that has a duration comparable to the duration of the expected in–flight shock. A mounting of the equipment on actual or dynamically similar structure provides a more realistic test than does a mounting on a rigid structure such as a shaker armature or slip table.

4.2.7.3 TEST LEVELS AND EXPOSURE

The shock spectrum in each direction along each of the three orthogonal axes shall be at least the maximum predicted level plus 6 dB for that direction. A sufficient number of shocks shall be imposed to meet the amplitude criterion in both directions on each of the three orthogonal axes at least three times. However, if a suitable test environment can be generated to satisfy the amplitude requirement in all six axial directions by a single application, this test environment shall be imposed three times. See appendix B, PG2–141, for the exception to this requirement.

4.2.7.4 SUPPLEMENTARY REQUIREMENTS

A visual inspection shall be made before and after the test. The visual inspection shall not entail any disassembly. Electrical and electronic components shall be energized and monitored if they are required to operate during flight shock events. A functional test shall be performed before and after all shock tests, and parameters monitored during the shocks to evaluate performance and detect any failures. Relays shall not transfer and shall not chatter in excess of specification limits. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test. See appendix B, PG2–141, for the exception to this requirement.

4.2.8 ACCELERATION TEST, COMPONENT QUALIFICATION

4.2.8.1 PURPOSE

This test demonstrates the capability of the component to withstand or, if appropriate, to operate in the design–level acceleration environment.

4.2.8.2 TEST DESCRIPTION

The component shall be mounted to a test fixture through the normal mounting points of the component. The component shall be tested in each of three mutually perpendicular axes. The specified accelerations apply to the geometric center of the test item.

4.2.8.3 TEST LEVELS AND DURATION

A. Acceleration Level. The test acceleration level shall be, at least, at the design level in each direction for each of the three orthogonal axes.

B. Duration. The test duration shall be five minutes each axis in each direction.

4.2.8.4 SUPPLEMENTARY REQUIREMENTS

A functional test shall be conducted before the acceleration test and after completion of the test. Electrical components, if operated during ascent, shall be powered during the test, and their parameters monitored for failures or intermittences. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test.

4.2.9 HUMIDITY TEST, COMPONENT QUALIFICATION

4.2.9.1 PURPOSE

This test demonstrates that the component is capable of surviving, without excessive degradation, the maximum predicted humidity environment to which the component may be exposed during fabrication, test, shipment, storage, preparation for launch, ascent, on–orbit operation, descent, and ferry flight.

4.2.9.2 TEST DESCRIPTION AND LEVELS

The component shall be installed in the chamber and tested in accordance with the following conditions:

A. Pretest Conditions. Chamber temperature shall be at room ambient conditions with uncontrolled humidity.

B. Cycle 1. The temperature shall be increased to +95 degrees F (+35 degrees C) over a one hour period; then the humidity shall be increased to not less than 95 percent over a one hour period with the temperature maintained at +95 degrees F (+35 degrees C). These conditions shall be held for two hours. The temperature shall be reduced to +36 degrees F (+2 degrees C) over a two hour period with the relative humidity stabilized at not less than 95 percent. These conditions shall be held for two hours.

C. Cycle 2. The foregoing cycle shall be repeated except that the temperature shall be increased from +36 degrees F (+2 degrees C) to +95 degrees F (+35 degrees C) over a two hour period [moisture is not added to the chamber until +95 degrees F (+35 degrees C) is reached].

D. Cycle 3. The chamber temperature shall be increased to +95 degrees F (+35 degrees C) over a two hour period without adding any moisture to the chamber. The test component shall then be dried with air at room temperature and 50–percent maximum relative humidity by blowing air through the chamber for six hours. The volume of air used per minute shall be equal to one to three times the test chamber volume. A suitable container may be used in place of the test chamber for drying the test component.

E. Cycle 4. The component shall be placed in the test chamber and the temperature increased to +95 degrees F (+35 degrees C) and the relative humidity increased to 90 percent over a one hour period. These end conditions shall be maintained for at least one hour. The temperature shall be reduced to +36 degrees F (+2 degrees C) over a one hour period with the relative humidity stabilized at 90 percent and these conditions maintained for at least one hour. A drying cycle should follow (see Cycle 3).

4.2.9.3 SUPPLEMENTARY REQUIREMENTS

The component shall be functionally tested prior to the test and at the end of Cycle 3 (within two hours after the drying) and visually inspected for deterioration or damage. The component shall be functionally tested during the Cycle 4 periods of stability: after the one hour period to reach +95 degrees F (+35 degrees C) and 90–percent relative humidity, and again after the one hour period to reach the +36 degrees F (+2 degrees C) and 90–percent relative humidity condition. The component shall be visually inspected for deterioration or damage after removal from the chamber.

4.2.10 PRESSURE TEST, COMPONENT QUALIFICATION

4.2.10.1 PURPOSE

This test demonstrates that the design and fabrication of such items as pressure vessels, pressure lines, fittings, and valves provide an adequate margin such that structural failure or excessive deformation does not occur at the maximum expected operating pressure.

4.2.10.2 TEST DESCRIPTION

A. Proof Pressure. For such items as pressure vessels, pressure lines, and fittings, the temperature of the component shall be consistent with the critical–use temperature and subjected to a minimum of one cycle of proof pressure. A proof–pressure cycle shall consist of raising the internal pressure (hydrostatically or pneumatically, as applicable) to the proof pressure, maintaining it for five minutes, and then decreasing the pressure to ambient. Evidence of permanent set or distortion or failure of any kind shall indicate failure to pass the test. See appendix C, PG3–117, for the exception to this paragraph.

B. Proof Pressure for Valves. With the valves in the open and closed positions, the proof pressure shall be applied for a minimum of one cycle to the inlet port for five minutes. Following the five minute pressurization period, the inlet pressure shall be reduced to ambient conditions. The interior and exterior of the article shall be visually examined. Evidence of deformation shall indicate failure to pass the test. The test may be conducted at room ambient temperature.

C. Ultimate Pressure. For such items as pressure vessels, pressure lines, and fittings, the temperature of the component shall be consistent with the critical–use temperature, and the component shall be pressurized to design ultimate pressure or greater. The internal pressure shall be applied at a uniform rate such that stresses resulting from shock loading are not imposed. Ultimate testing shall not be performed on actual flight articles. See appendix B, PG2–80, for the exceptions to these paragraphs.

D. Ultimate Pressure for Valves. With the valve in the open or closed position, as applicable, the design ultimate pressure shall be applied to the inlet port for five minutes. Following the five minute pressurization period, the inlet pressure shall be reduced to ambient conditions. The exterior of the article shall be visually examined for indications of deformation or failure. The test may be conducted at room ambient temperature. Ultimate testing shall not be performed on actual flight articles.

4.2.10.3 TEST LEVELS

A. Temperature. The temperature shall be as specified in the test description. As an alternative, tests may be conducted at ambient room temperatures if the test pressures are suitably adjusted to account for temperature effects on strength and fracture toughness.

B. Proof Pressure. Proof pressure shall be as specified in SSP 30559, section 3.

C. Ultimate Pressure. Ultimate pressure shall be specified in SSP 30559, section 3. See appendix A, PG1–148, for the exception to this paragraph. See appendix C, PG3–158, for the exception to this paragraph.

See appendix D, GFE–10, GFE–53, GFE–98, and GFE–99, for the exceptions to these paragraphs.

4.2.10.4 SUPPLEMENTARY REQUIREMENTS

The component shall withstand proof pressure without leakage or detrimental deformation. Applicable safety standards shall be followed in conducting all tests.

See appendix D, GFE–53, GFE–98, and GFE–99, for the exceptions to this paragraph.

4.2.11 LEAK TEST, COMPONENT QUALIFICATION

4.2.11.1 PURPOSE

This test demonstrates the capability of sealed and pressurized components to meet the leakage rate requirements specified in the component development specification.

4.2.11.2 TEST DESCRIPTION AND ALTERNATIVES

Component leak tests shall be made prior to initiation of, and following the completion of, the following component qualification tests: 1) thermal vacuum and/or thermal cycle; 2) random, sinusoidal, or acoustic vibration; 3) pyrotechnic shock; and 4) humidity. The leak test method employed shall have sensitivity and accuracy consistent with the component specified maximum allowable leakage rate. The sensitivity of the leak test, in particular, shall be quantitatively less than the minimum leakage rate to be detected by a factor of at least two to ensure reliability of measurements. When temperature potentially affects the sealing materials or surfaces, an evaluation of the hardware design and operational characteristics shall be performed and, if technically warranted, the leak test shall be conducted at the minimum and maximum qualification temperature limits. If it is determined from the evaluation that a leak test at temperature limits is warranted on a component of a given level–of–assembly due solely to one or more lower tier components comprising the assembly, and it can be shown that all of those lower tier components receive an appropriate leak test at temperature limits as part of a lower level qualification test, then the higher level–of–assembly does not require leak testing at temperature limits. One of the following methods or another suitable leak test method in accordance with the criteria established in section 10 shall be used. Method I, II, III, IV, V, VIII, IX, X, or XI, as appropriate, shall be used for pressurized components. Method VI or VII shall be used for sealed components. Leak testing may be performed prior to component proof pressure testing only if approved by the responsible safety organization. In all cases, leak testing shall be conducted after the component proof pressure test.

A. Method I (Immersion, to be used only as a pass/fail test; this method does not provide a quantitative measurement of component leakage rate). This method may be used for total or local internal–to–external leak testing of pressurized components and systems. Internal gas pressure shall be applied across the pressure boundary for a minimum duration of 15 minutes before the test liquid contacts the external surface. Lighting in the area to be examined shall be no less than 1000 lux or lumen/m^2 (100 foot–candles) in brightness. Illumination shall be free from shadows over the surface area under inspection. The observer shall place his eyes within 60 cm (2 feet) of the surface to be examined. Mirrors or magnifying glasses may be used to improve visibility of indications. The component shall be completely immersed in a liquid. The critical side or sides of interest of the component shall be in a horizontal plane facing up. There shall be no observed leakage during immersion (as evidenced by one or more bubbles emanating from the component). NOTE: See appendix C, PG3–200, for exceptions existing to the previous Gross Leak Test requirements in SSP 41172, Revision T.

B. Method II (Vacuum Chamber). This method may be used for total internal–to–external leak testing of pressurized components and systems. The component shall be placed in a vacuum chamber (bell jar) and tested for leakage with a leak detector appropriate for the tracer gas used. The vacuum chamber system leak test sensitivity shall be quantified and documented with a standard leak not to exceed the component maximum allowable leakage requirement. The component shall be charged with a known concentration of a tracer gas to the required pressure. Pressure shall be maintained until stabilization of the leak detector output is achieved (stabilization shall be defined as three consecutive readings no less than 5 minutes apart with no more than a 10 percent variation in the leak detector output from one measurement to the next, including the first and last measurements). Calibration data and leak detector initial and final readings shall be recorded. The final component leakage rate shall be recorded along with a minimum of 3 data points within a 15 minute duration to demonstrate stabilization in accordance with the definition above.

C. Method III (Pressure Change). The pressure decay technique may be used for total internal–to–external leak testing of pressurized components and systems. To improve the accuracy of this technique, a reference vessel connected to the pressurized component or system may be used. If ambient temperature changes, the component and reference vessel volumetric change shall be taken into account. The pressure rise technique may be used for total external–to–internal leak testing of sealed components and systems. The component internal pressure, barometric pressure, and ambient temperature (or temperature of the component) shall be monitored for the required time to determine the actual pressure drop or rise and the corresponding leakage rate. The pressure gauge/transducer shall have accuracy and sensitivity adequate to measure the minimum required pressure change. The tolerance/error associated with the total internal volume of the component and test fixture under pressure used for the leakage rate calculation shall be taken into account as a maximum positive value.

D. Method IV (Chemical Indicator, to be used only as a pass/fail test; this method does not provide a quantitative measurement of component leakage rate). A suitable indicator such as a dilute solution of phenolphthalein or other suitable color–change indicator such as colorimetric in accordance with ASTM 1066.95, Revision (2000) shall be applied to all seams, terminals, and pinch tubes subject to leakage of the working fluid. A change in the color of the indicator shall be an indication of a leak. After testing, the indicator shall be removed (e.g., with distilled water).

E. Method V (Detector Probe, to be used only as a pass/fail test for individual joints (e.g., welds and mechanical fittings); this method does not provide a quantitative measurement of component leakage rate). This detector probe is a semiquantitative technique used to detect and locate internal–to–external leaks in pressurized components and systems, and shall not be considered quantitative. The component shall be charged with a known concentration of a tracer gas to the required pressure. Prior to examination, the test pressure shall be held for a minimum duration of 30 minutes. Prior to examination, the tracer gas background shall be measured and the leak test setup shall be calibrated for the test by passing the detector probe tip across the orifice of a calibrated leak (the magnitude of the calibrated leak shall be equal to or less than the maximum allowable leakage rate). The resulting leak detector output shall be at least 40 percent above the tracer gas background. After the required soak time, the detector probe tip shall be passed over the test surface at the same scanning rate and distance used during the system calibration. The system calibration will be repeated every 60 minutes and any time test conductors/operators are changed. Any leak detector output above the established tracer gas background with allowance made for atmospheric tracer gas variations and leak detector drift, that in the aggregate do not exceed 40 percent of the tracer gas background, indicates a leak.
NOTE: See appendix A, PG1–207, PG1–212, PG1–214, PG1–216, and PG1–252, for exceptions existing for components of pressurized fluid systems in SSP 41172, Revision T.

F. Method VI (Hood). This method may be used for total external–to–internal leak testing of sealed components and systems. The component internal volume shall be evacuated to a vacuum compatible with a tracer gas leak detector. The system sensitivity shall be determined by installing a standard leak at the furthest point from the leak detector. The external surfaces of the component shall be exposed to a verified concentration of a tracer gas. Pressure shall be maintained until stabilization of the leak detector output is achieved (stabilization shall be defined as three consecutive readings no less than 5 minutes apart with no more than a 10 percent variation in the leak detector output from one measurement to the next, including the first and last measurements). Calibration data and leak detector initial and final readings shall be recorded. The final component leakage rate shall be recorded along with a minimum of 3 data points within a 15 minute duration to demonstrate stabilization in accordance with the definition above.

G. Method VII (Tracer Probe, this method may be used to locate known external–to–internal leaks but shall not be used for verifying the specified allowable leakage rate of flight hardware). The component internal volume is evacuated to a vacuum compatible with a tracer gas leak detector. The tracer probe is connected to a source of 100 percent tracer gas with a valved opening at the other end for directing a stream of tracer gas over the component. Any indication of tracer gas above the background by the leak detector indicates a leak.

H. Method VIII (Accumulation). This method may be used for total internal–to–external leak testing of pressurized components and systems. The component shall be enclosed in a suitable enclosure. The enclosure shall be calibrated by placing a standard leak in the enclosure for a predetermined period of time. At the end of the time period, a detector probe shall be placed in the enclosure and the maximum leak detector response shall be recorded. The enclosure shall then be purged with nitrogen or air. The component shall be charged with a known concentration of a tracer gas to the required pressure. Prior to examination, the test pressure shall be held for a minimum duration of 30 minutes. The enclosure shall be purged with nitrogen or air and sealed. After the time period used for the calibration, the detector probe shall be placed in the enclosure. Calibration data and leak detector initial and final readings shall be recorded. The final component leakage rate shall also be recorded.

I. Method IX (Volumetric Displacement). This method may be used for total internal–to–internal leak testing of pressurized components such as valves, pressure regulators, or heat exchangers. One side of the component shall be pressurized to the required pressure while the other side across the internal barrier shall be sealed from the atmosphere and attached to a suitable device for the purposes of demonstrating volumetric displacement. This will be accomplished by either using a displacement of liquid or by moving a fluid meniscus along the graduations of the measuring device.

J. Method X (Leak Detector Direct Connection). This method may be used for total internal–to–internal leak testing of pressurized components such as valves, pressure regulators, or heat exchangers. One side of the component shall be charged with a known concentration of a tracer gas to the required pressure while the other side across the internal barrier shall be sealed from the atmosphere and attached to the leak detector. Pressure shall be maintained until stabilization of the leak detector output is achieved (stabilization shall be defined as three consecutive readings no less than 5 minutes apart with no more than a 10 percent variation in the leak detector output from one measurement to the next, including the first and last measurements). Calibration data and leak detector initial and final readings shall be recorded. The final component leakage rate shall be recorded along with a minimum of 3 data points within a 15 minute duration to demonstrate stabilization in accordance with the definition above.

K. Method XI (Local Vacuum Chamber). This method may be used for local internal–to–external leak testing of pressurized components and systems. The local vacuum chamber (bell jar) connected to the tracer gas leak detector shall be installed on the component area to undergo the leak test. The local vacuum chamber system sensitivity shall be quantified and documented with a standard leak not to exceed the component maximum allowable leakage rate requirement. The component shall be charged with a known concentration of a tracer gas to the required pressure. Pressure shall be maintained until stabilization of the leak detector output is achieved (stabilization shall be defined as three consecutive readings no less than 5 minutes apart with no more than a 10 percent variation in the leak detector output from one measurement to the next, including the first and last measurements). Calibration data and leak detector initial and final readings shall be recorded. The final component leakage rate shall be recorded along with a minimum of 3 data points within a 15 minute duration to demonstrate stabilization in accordance with the definition above.

NOTE: See appendix A, PG1–192, PG1–246, PG1–266, and PG1–268, for exceptions existing prior to the current requirements. See appendix B, PG2–68, PG2–107, PG2–114, PG2–116, and PG2–118, for exceptions existing prior to the current requirements. See appendix C, PG3–177, and PG3–183, for exceptions existing prior to the current requirements.

4.2.11.3 TEST LEVELS AND DURATION

The leak tests shall be performed with the component pressurized at the maximum design pressure and then at the minimum design pressure if the seals are dependent upon pressure for proper sealing. Regardless of the method used, the test duration shall be sufficient to detect any out–of–specification leakage.

NOTE: See appendix A, PG1–192, PG1–201, PG1–210, PG1–250, PG1–264, and PG1–268, for exceptions to the previous Fine Leak Test requirements in SSP 41172, Revision T. See appendix B, PG2–101, for an exception existing to the previous Gross Leak Test requirements in SSP 41172, Revision T. See appendix B, PG2–92, PG2–96, and PG2–103, for exceptions to the previous Fine Leak Test requirements in SSP 41172, Revision T. See appendix C, PG3–128, PG3–198, and PG3–228, for exceptions to the previous Fine Leak Test requirements in SSP 41172, Revision T.

NOTE: See appendix C, PG3–136, PG3–155, PG3–163, and PG3–202, for exceptions existing to the previous pressurized systems and components leak requirements in SSP 41172, Revision T.

See appendix A, PG1–248, PG1–258, and PG1–262, for the exceptions to these paragraphs. See appendix B, PG2–150 for an exception to these paragraphs. See appendix C, PG3–124, PG3–126, PG3–134, PG3–140, PG3–213, PG3–220, and PG3–226, for the exceptions to these paragraphs. See appendix D, GFE–10 and GFE–28, for the exceptions to these paragraphs.

4.2.11.4 SUPPLEMENTARY REQUIREMENTS

Component leak tests are considered adjunctive to the component qualification environmental tests in that their results are part of the success criteria for these tests.

4.2.12 ELECTROMAGNETIC COMPATIBILITY TEST, COMPONENT QUALIFICATION

4.2.12.1 PURPOSE

This test demonstrates that the electromagnetic interference characteristics (emission and susceptibility) of the component under normal operating conditions does not result in malfunction of the component and that the component does not emit, radiate, or conduct interference which results in malfunction of other system components.

4.2.12.2 TEST DESCRIPTION

The test shall be in accordance with the requirements of SSP 30237. An evaluation shall be made of each component to determine which tests shall be conducted in accordance with SSP 30238.

4.2.12.3 TEST LEVELS AND DURATION

Test levels and duration shall be in accordance with SSP 30237. See appendix A, PG1–170, for the exception to this paragraph. See appendix D, GFE–11, for the exception to these paragraphs.

4.2.12.4 SUPPLEMENTARY REQUIREMENTS

Electrical functional tests in accordance with paragraph 4.2.1 shall be conducted prior to and after the susceptibility test. The electrical functional test shall be sufficiently thorough to ensure that no unacceptable performance degradation or failures of components or circuitry critical to Electromagnetic Interference (EMI)/Electromagnetic Compatibility (EMC) characteristics has occurred during performance of the EMI/EMC test sequences. See appendix A, PG1–170, for the exception to this paragraph.

4.2.13 LIFE TEST, COMPONENT QUALIFICATION

4.2.13.1 PURPOSE

This test increases confidence that components which may have a wearout, drift, or fatigue–type failure mode have the capability to withstand the maximum duration or cycles of operation to which they are expected to operate during repeated ground testing and in flight without degradation of their function outside of allowable limits. For structural components, reference SSP 30559.

4.2.13.2 TEST DESCRIPTION

The test hardware shall be set up to operate in conditions that simulate the flight conditions to which they would be subjected. These environmental conditions shall be selected for consistency with end–use requirements and the significant life characteristics of the particular component. Typical environments are ambient, thermal, thermal vacuum, and various combinations of these. The hardware shall be selected at random from production articles and may include a qualification article. The test shall be designed to demonstrate the ability of the component to withstand the maximum operating time and the maximum number of operational cycles predicted during its service life with a suitable margin. For components having a relatively low–percentage duty cycle, it shall be acceptable to compress the operational duty cycle into a tolerable total test duration. For components that operate continuously in orbit or at very high–percentage duty cycles, accelerated test techniques may be employed if such an approach can be shown to be valid.

4.2.13.3 TEST LEVELS AND DURATION

A. Pressure. Ambient pressure shall be used except for unsealed articles where degradation due to a vacuum environment may be anticipated. In those cases, a pressure of less than 0.0001 Torr (0.0133 Pa) shall be used.

B. Environmental Levels. The maximum predicted environmental levels shall be used. For accelerated life tests, environmental levels may be selected that are more severe than flight levels, provided the higher stresses can be correlated with life at the predicted use stresses and do not introduce additional failure mechanisms.

C. Duration. The total operating time or number of operational cycles for a component life test shall be twice that predicted during the service life, including ground testing, to demonstrate an adequate margin. See appendix B, PG2–95, for the exception to this paragraph.

D. Functional Duty Cycle. Complete functional tests shall be conducted before the test begins, after each 168 hours of operation, and during the last two hours of the test. An abbreviated functional test shall be conducted periodically to ascertain that the component is functioning within specification limits.

4.2.13.4 SUPPLEMENTARY REQUIREMENTS

For statistical–type life tests, the duration is dependent upon the number of samples, confidence, and reliability to be demonstrated.

4.2.14 CORONA/ARCING TEST, COMPONENT QUALIFICATION

4.2.14.1 PURPOSE

The purpose of this test is to ensure no detrimental corona/arcing occurs in unsealed electrical/electronic components which are required to operate during ascent/descent or during a depressurization/repressurization event. Electrical/electronic components utilizing a sealed chassis design or components which are powered under space vacuum conditions only do not require corona/arcing testing. This test should be performed as part of the thermal vacuum or depress/repress qualification test.

4.2.14.2 TEST DESCRIPTION

The component shall be placed in a vacuum chamber at ambient conditions. With the component powered–up and monitored, the chamber pressure shall then be reduced to the specified low pressure level. The time for change of pressure between 20 Torr and 0.001 Torr (or vice versa) shall be at least 10 minutes to allow sufficient time in the region of critical pressure. The worst–case design and operating condition(s) (either by normal component design functions or externally induced transient conditions) producing the most likely corona/arcing potential shall be duplicated in the region of critical pressure. Full functional testing of the component in the critical pressure region shall not be required unless necessary to adequately demonstrate absence of corona/arcing. This corona/arcing monitoring may be performed during either depressurization or repressurization of the vacuum chamber. The chamber shall then be returned to ambient pressure.

4.2.14.3 TEST LEVELS AND DURATION

A. Pressure. The pressure shall be reduced from ambient to below 0.001 Torr.

B. Temperature. Ambient temperature shall be used.

4.2.14.4 SUPPLEMENTARY REQUIREMENTS

An assessment shall be made of the component design and operational characteristics to establish proper corona monitoring techniques. MSFC–STD–531 may be used for guidance.

4.3 STRUCTURAL QUALIFICATION TESTS

4.3.1 STATIC STRUCTURAL LOAD TEST

4.3.1.1 PURPOSE

This test demonstrates the adequacy of the structure to meet requirements of strength, stiffness, or both, with the required desired design margin when subjected to simulated critical environments, such as temperature, acceleration, pressure, and other relevant loads, predicted to occur during its service life.

4.3.1.2 TEST DESCRIPTION

The structural configuration, materials, and manufacturing processes employed in the qualification test articles shall be identical to those of flight articles. When structural items are rebuilt or reinforced to meet specific strength or rigidity requirements, all modifications shall be structurally identical to the changes incorporated in flight articles. The support and load application fixture shall consist of an adequate replication of the adjacent structural section to provide boundary conditions which simulate those existing in the flight article. Static loads representing the limit load and the ultimate load shall be applied to the structure, and measurements of the strain and deformation shall be recorded. Strain and deformation shall be measured before loading, after removal of the limit/ultimate loads, and at several intermediate levels up to limit/ultimate load for post–test diagnostic purposes. The test conditions shall include the combined effects of acceleration, pressure, preloads, and temperature. These effects can be simulated in the test conditions as long as the failure modes and design margins are enveloped by the simulations. For example, temperature effects, such as material degradation and additive thermal stresses, often can be accounted for by increasing mechanical loads. Analysis of flight profiles shall be used to determine the proper sequencing or simultaneity for application of thermal stresses. When prior loading histories affect the structural adequacy of the test articles, these shall be included in the test requirements. The final test may be taken to failure to substantiate the capability to accommodate internal load redistribution, to provide data for any subsequent design modification effort, and to provide data for use in any weight–reduction programs. Failures at limit load shall include detrimental deformation (as defined in SSP 30559), and at ultimate load shall include rupture or collapse.

4.3.1.3 TEST LEVELS AND DURATION

A. Static Loads. The loads shall be increased until the specified test loads are reached. In cases where a load or other environment has a relieving, stabilizing, or otherwise beneficial effect on the structural capability, the minimum, rather than the maximum, design load shall be used, and it shall not be increased by any factor of safety.

B. Temperature. Critical flight temperature–load combinations shall be used to determine the expected worst–case stress anticipated in flight.

C. Duration of Loading. The duration of loading shall be sufficient to record test data such as stress, strain, deformation, and temperature.

See appendix B, PG2–142, for the exception to this requirement.

4.3.1.4 SUPPLEMENTARY REQUIREMENTS

Pretest analysis shall be conducted to identify the locations of minimum design margins and associated failure modes which correspond to the selected critical test–load conditions. This analysis shall be used to locate instrumentation, to determine the sequence of loading conditions, and to afford early indications of anomalous occurrences during the test. This analysis shall also form the basis for judging the adequacy of the test loads. Internal loads resulting from the limit–test conditions shall envelope all critical internal loads expected in flight. Structural design margins are defined in SSP 30559. See appendix B, PG2–142, for the exception to this requirement.

4.3.2 MODAL SURVEY

A modal survey shall be conducted to define or verify an analytically derived dynamic model for use in flight loading event simulations and for use in examinations of postboost configuration elastic effects upon control precision and stability. This test is conducted on a flight–quality structure as augmented by mass–simulated components. The data obtained shall be adequate to define orthogonal mode shapes, mode frequencies, and mode damping ratios of all modes which occur within the frequency range of interest.

4.4 FLIGHT ELEMENT QUALIFICATION TEST

The flight element qualification test baseline consists of all the required tests specified in Table 4–2. Flight element tests include assembly elements (i.e., Pre–Integrated Truss, Node, Photovoltaic Module) where the flight element is not launched as a single entity. Where it is impractical to test flight elements as a single entity, testing of major assemblies that constitute the flight element may be utilized with the appropriate analyses, simulations, and/or simulators to satisfy this requirement. If the flight element is controlled by on–board data processing, the flight software will be resident in the on–board computer for these tests. The verification of the operational requirements shall be demonstrated. See appendix B, PG2–128 and PG2–129, for the exceptions to this requirement.

TABLE 4–2 FLIGHT ELEMENT QUALIFICATION TESTS
(PAGE 1 OF 1)

Tests	Notes
Functional	(1) (4)
EMC	(4)
Acoustic Vibration	(2) (3)
Pressure/Leak	
Model Survey	(3)
Static Structural Load	(3)

Notes:
(1) Electrical and mechanical functional tests shall be conducted prior to and following the acoustic vibration test, and following the pressure/leak test.
(2) Random vibration may be conducted in place of acoustic vibration test for flight elements.
(3) Can be satisfied by test on a dedicated nonfunctional structural test article. This includes Pre–integrated Truss segments and Photovoltaic Module launch configurations.
(4) Does not apply to structures.

4.4.1 FUNCTIONAL TEST, FLIGHT ELEMENT QUALIFICATION

4.4.1.1 PURPOSE

This test verifies that mechanical and electrical performance of the flight element meets the specification requirements, verifies compatibility with ground support equipment, and validates all test techniques and software algorithms used in computer–assisted commanding and data processing.

4.4.1.2 MECHANICAL FUNCTIONAL TEST

Mechanical devices, valves, deployables, and separable entities shall be functionally tested with the flight element in the ascent, orbital, or recovery configuration appropriate to the function. Alignment checks shall be made where appropriate. The maximum and minimum limits of acceptable performance shall be determined with respect to mechanics, time, and other applicable requirements. For each mechanical operation, such as appendage deployments, tests shall demonstrate positive margins of strength, torque margins, and that they function at conditions above and below specified operational limits. Where operation in a 1–G environment cannot be performed, a suitable ground–test fixture may be utilized to permit operation and evaluation of the devices. Fit checks shall be made of the flight element physical interfaces with other flight elements and the launch vehicle by means of master gauges or interface assemblies. The most adverse tolerance accumulation shall be considered in these fit checks.

4.4.1.3 ELECTRICAL FUNCTIONAL TEST

The flight element shall be in its flight configuration with all components and subsystems connected except pyrotechnic devices. This test shall verify the integrity of all electrical circuits in the flight element, including redundant paths, by the application of an initiating stimulus and the confirmation of the successful completion of the event. The test shall be designed to operate all components, primary and redundant, and all commands shall be exercised. The operation of all thermal control components, such as heaters and thermostats, shall be verified by test. The test shall demonstrate that all commands having preconditioning requirements (such as enable, disable, specific equipment configuration, specific command sequence, etc.) cannot be executed unless the preconditions are satisfied. Equipment performance parameters (such as power, voltage, gain, frequency, command, and data rates) shall be varied over specification ranges to demonstrate the performance margins. Autonomous functions shall be verified to occur when the conditions exist for which they are designed. The flight element main bus shall be continuously monitored by a power transient monitor system. All telemetry monitors shall be verified, and pyrotechnic circuits shall be energized and monitored. A segment of this test shall operate the flight element through an ascent and mission profile with all events occurring in actual flight sequence.

4.4.1.4 SUPPLEMENTARY REQUIREMENTS

Mechanical and electrical functional tests shall be conducted prior to and after each of the flight element environmental tests to detect equipment anomalies and to assure that performance meets the requirements of the specification. These tests do not require the mission profile sequence. Data analysis to verify the adequacy of the testing and the validity of the data shall be completed before disconnection from a particular environmental test configuration so that any required retesting can be readily accomplished.

4.4.2 ELECTROMAGNETIC COMPATIBILITY TEST, FLIGHT ELEMENT QUALIFICATION

4.4.2.1 PURPOSE

This test demonstrates the EMC of the element and ensures the element has adequate margins.

4.4.2.2 TEST DESCRIPTION

The test shall be defined and conducted in accordance with the requirements of SSP 30243. An evaluation shall be made of each system to determine which tests shall be performed as the baseline requirements.

4.4.3 ACOUSTIC VIBRATION TEST, FLIGHT ELEMENT QUALIFICATION

4.4.3.1 PURPOSE

This test demonstrates the ability of the flight element to withstand or to operate in the maximum expected acoustic environment. This test also verifies the adequacy of component vibration qualification criteria.

4.4.3.2 TEST DESCRIPTION

The flight element or structural test article shall be installed in a reverberant acoustic cell capable of generating desired sound pressure levels. It shall be mounted on a flight–type support structure or reasonable simulation thereof. The mechanical configuration of the flight element shall be as it is during ascent. Dynamic instrumentation shall be installed to measure vibration responses at attachment points of critical and representative components.

4.4.3.3 TEST LEVELS AND DURATION

The sound pressure level shall be at least the maximum expected flight level and spectrum, but not less than 141 dB overall (whose spectrum is defined by NSTS 21000–IDD–ISS, paragraph 4.1.1.5). Exposure test time shall be at least three times the expected flight exposure time to the maximum flight environment or three times the acceptance test duration if that is greater but not less than three minutes. See appendix A, PG1–13 and PG1–203, for the exceptions to this paragraph.

4.4.3.4 SUPPLEMENTARY REQUIREMENTS

Functional tests are required before and after the environmental exposure.

4.4.4 PRESSURE/LEAK TEST, FLIGHT ELEMENT QUALIFICATION

4.4.4.1 PURPOSE

This test demonstrates the capability of pressurized structure and fluid subsystems to meet the flow, pressure, and leakage rate requirements specified in the flight element specification.

4.4.4.2 TEST DESCRIPTION

The requirements of the flight element including flow, leakage, and regulation shall be measured while operating applicable valves, pumps, and motors. The flow checks shall verify that the plumbing configurations are adequate. Checks for subsystem cleanliness, moisture levels, and pH shall be made as applicable. Where sealed or pressurized subsystems are assembled with other than brazed or welded connections, the specified torque values for these connections shall be verified prior to leak tests. In addition to the high–pressure test, propellant tanks and thruster valves shall be tested for leakage under propellant–servicing conditions. The system shall be evacuated to the internal pressure normally used for propellant loading and the systems pressure monitored for any indication of leakage.

A suitable leak test method in accordance with section 10 herein shall be used.

4.4.4.3 TEST LEVELS AND DURATION

The flight element shall be pressurized to proof pressure and held until all strain and deflection data are recorded, and then shall be reduced to the maximum design pressure. The proof pressure shall be as specified in SSP 30559, Section 3. Inspection for leakage after the proof test shall be at no less than 14.7 psid. The duration of the evacuated propulsion system leak test shall not exceed the time that this condition is normally experienced during propellant loading. These element–level internal proof pressure and leak tests are only applicable to flight elements with pressurized internal volumes.

All flight element pressurized fluid systems shall be proof pressure tested at the final system assembly level. All flight element pressurized fluid systems shall be leak tested at the final system assembly level at the system maximum design pressure after successful system proof pressure testing. Where it is impossible or impractical to perform system–level proof and leak testing in the final assembly configuration due to design or manufacturing limitations, proof pressure and leak testing at the highest assembly level practical shall be required. In this case, the proof factor shall be appropriate for the level of assembly during the test, and the overall fluid system proof and leak test strategy shall require approval by the ISS Program.

See appendix B, PG2–150, for an exception to these paragraphs.

4.4.4.4 SUPPLEMENTARY REQUIREMENTS

Applicable safety standards shall be followed in conducting all tests. Leak tests shall be conducted only after satisfactory proof pressure tests have been completed. Leak detection and measurement procedures may require vacuum chambers, enclosure of the entire flight element or localized areas, or other special techniques to achieve the required accuracy.

5.0 ACCEPTANCE TESTING

5.1 COMPONENT ACCEPTANCE TESTS

The component acceptance tests that are a Program requirement are designated in Table 5–1 according to test category and component category. Where components fall into more than one category, the required tests for each category shall be applied. Subsequent paragraphs describe in detail the requirements for each test category. The word ”required” means that, as a minimum, the component is required to be tested to the specific environment. Acceptance test articles shall be subjected to environmental test levels, cycles, and durations within the range of the design qualification test levels cycles and durations.

TABLE 5–1 COMPONENT ACCEPTANCE TESTS
(PAGE 1 OF 1)

Test	Electronic or Electrical Equipment	Antennas	Moving Mechanical Assembly	Solar Panel	Batteries	Fluid or Propulsion Equipment	Pressure Vessels	Thermal Equipment	Optical Equipment
Functional (1)	R	R	R	R	R	R	R	R	R
Thermal Vacuum (8)	R(4)	–	R(6)	–	R	R	–	R	–
Thermal Cycling	R(4)	–	–	–	–	R	–	–	R
Random Vibration	R	R(7)	R(7)	–	–	R(5)	–	R(5)	R(7)
Acoustic Vibration	–	R(3)	R(3)	R	–	–	–	–	–
Pressure	–	–	–	–	R(2)	R	R	–	–
Leak	R(2)	–	–	–	R(2)	R	R	R	R(2)
Burn–In	R	–	–	–	–	–	–	–	–
Oxygen Compatibility	–	–	–	–	–	R(10)	–	–	–
Corona (11) (12)	R	–	–	–	–	–	–	–	–

LEGEND: R = REQUIRED – The ISS requires as a minimum that the article be tested to detect material and workmanship defects.

Notes:

(1) Functional tests shall be conducted prior to and following environmental test.
(2) Required only on sealed or pressurized equipment.
(3) Either random vibration or acoustic vibration test required with the other optional.
(4) Minimum 100 degrees F (55.6 degrees C) sweep required.
(5) Only maximum predicted flight spectrum and level minus 6 dB required.
(6) Only components with close tolerance requiring precise adjustment, or that cannot be inspected effectively, require an acceptance thermal vacuum test.
(7) Only components with close tolerances requiring precise adjustment, or that cannot be inspected effectively, require an acceptance random vibration test.
(8) For components which operate in pressurized environment only, thermal vacuum testing is optional.
(9) When a proven technique of acceptance by inspection without vibration testing has been demonstrated on previous space programs, items are not required to undergo random vibration or acoustic vibration acceptance tests.
(10) Only required for components wetted with pure oxygen.
(11) Corona testing is not required for components with a sealed chassis or components which are powered on and operating under space vacuum conditions only.
(12) See Table 5–a1 for component voltage criteria dictating corona testing.

TABLE 5–a1 CORONA TEST VOLTAGE CRITERIA

Peak Voltage (V) (1)	Transient Duration (microseconds)	Corona Testing Required
< 150	N/A	N
150 <= V < 190	<250	N
150 <= V < 190	>= 250	Y
>= 190	N/A	Y

Notes:
(1) Applies to both steady–state and transient conditions. Also applies to input voltages, internal voltages, and output voltages.

5.1.1 FUNCTIONAL TEST, COMPONENT ACCEPTANCE

5.1.1.1 PURPOSE

This test verifies that the electrical and mechanical performance of the component meets the specified operational requirements of the component.

5.1.1.2 TEST DESCRIPTION

Electrical tests shall include application of expected voltages, impedance, frequencies, pulses, and waveforms at the electrical interfaces of the component, including all redundant circuits. Mechanical tests shall include application of torque, load, and motion as appropriate. These parameters shall be varied throughout their specification ranges and the sequences expected during its life cycle, and the component output shall be measured to verify the component performance to specification requirements. Functional performance shall also include electrical continuity, stability, response time, alignment, pressure, leakage, or other special functional tests related to a particular component configuration. See appendix D, GFE–114, for the exception to this paragraph.

5.1.1.3 SUPPLEMENTARY REQUIREMENTS

A functional test shall be conducted prior to each of the environmental tests to assure that performance meets the requirements of the particular specification. The same functional test shall be conducted after each environmental test. Functional tests shall not be required after application of ultimate loads during a structural loads test. See appendix A, PG1–255, for the exception to this paragraph. See appendix C, PG3–196, for the exception to this paragraph.

5.1.2 THERMAL VACUUM TEST, COMPONENT ACCEPTANCE

5.1.2.1 PURPOSE

This test detects material and workmanship defects prior to installation into a flight element by subjecting the article to a thermal vacuum environment.

5.1.2.2 TEST DESCRIPTION

The component shall be mounted in a vacuum chamber on a thermally controlled heat sink or in a manner to simulate the flight environment.

With the chamber at the test–pressure level, radio frequency equipment shall be monitored to assure that multipacting does not occur. A temperature cycle begins with the chamber and component at ambient temperature. The temperature of the chamber is reduced to bring the component to the specified low acceptance level and stabilized. Temperature stability has been achieved when the rate of change is no more than 5.4 degrees F per hour (3 degrees C per hour).

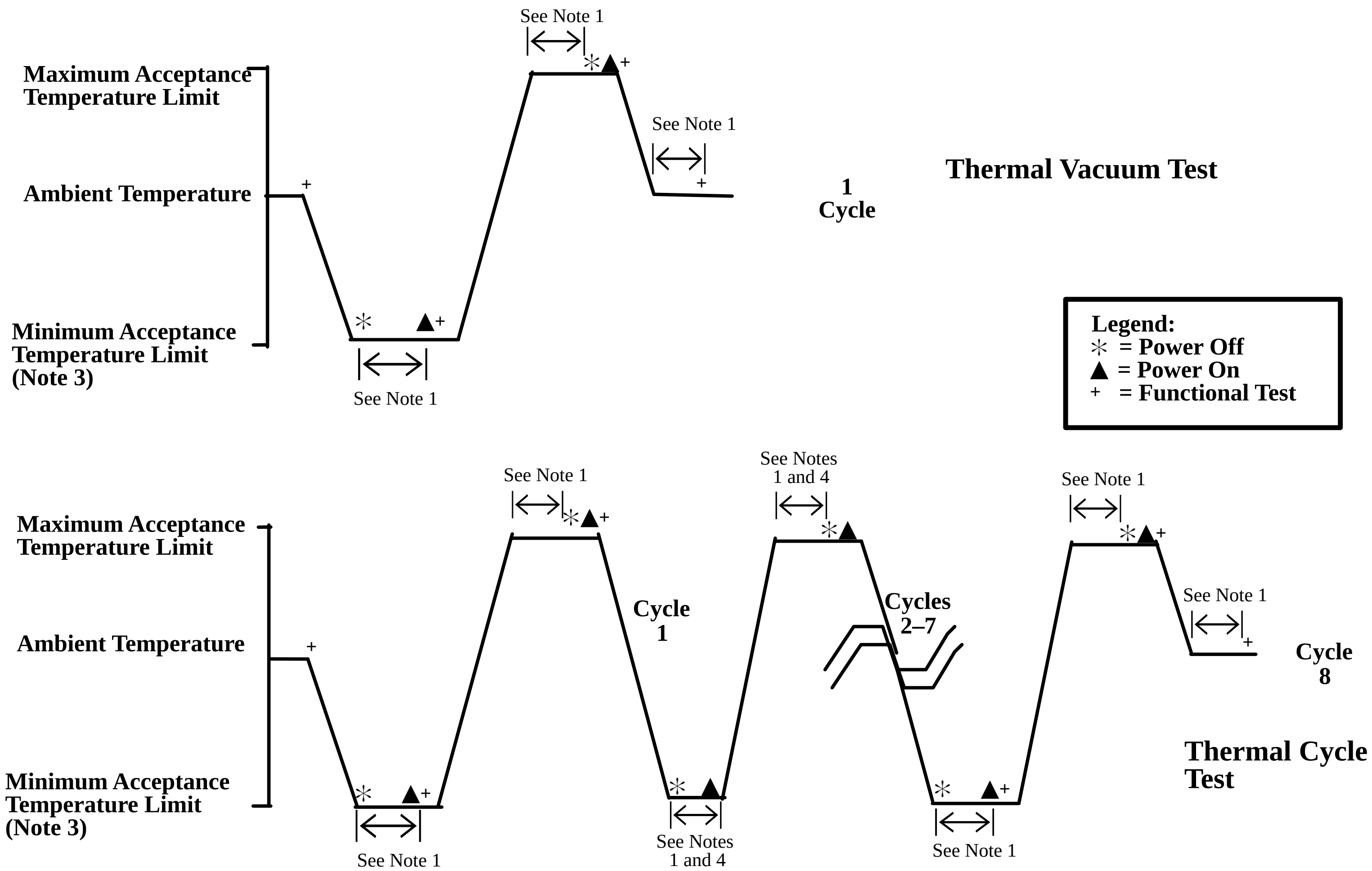
Components that operate on orbit shall be turned off, then started after a soak period sufficient to ensure the component internal temperature has stabilized at the specified temperature, and then functionally tested. With the component operating, the temperature of the chamber is increased to bring the component acceptance temperature to the upper temperature level. After the component temperature has stabilized at the specified level, the component shall be turned off, then started after electrical circuits have been discharged. The temperature of the chamber and component shall then be reduced to ambient conditions. This constitutes one complete temperature cycle.

See appendix A, PG1–77, for the exception to this paragraph. See appendix B, PG2–46, for the exception to this paragraph. See appendix C, PG3–218, for the exception to this paragraph.

5.1.2.3 TEST LEVELS AND DURATION

A. Pressure. The pressure shall be reduced from atmospheric to below 0.0001 Torr (0.0133 Pa). See appendix A, PG1–191, for the exception to this paragraph. See appendix B, PG2–110 and PG2–144, for the exceptions to this paragraph. See appendix D, GFE–76, for the exception to this paragraph.

B. Temperature (See Figure 5–1). The component shall be at the maximum acceptance limit during the hot portion of the cycle and at the minimum acceptance limit during the cold portion of the cycle. For the components identified in Table 5–1 with note 4 there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) wherever possible. When the component’s required non–operating temperature range exceeds its required operating temperature range, the component shall be exposed to one cycle of the non–operating maximum and minimum temperatures. This cycle may be combined with the required operating cycle in accordance with Figure 5–1a. If performed separately, it shall precede the operating cycle (Figure 5–1b). The component is not required to operate during exposure to non–operating temperatures. This non–operating temperature cycle is not required during acceptance thermal vacuum testing if it is performed during acceptance thermal cycle testing in accordance with 5.1.3.3B. See appendix A, PG1–157 and PG1–180, for the exceptions to this paragraph. See appendix B, PG2–41, PG2–58, PG2–83, and PG2–137, for the exceptions to this paragraph. See appendix C, PG3–190, for the exception to this paragraph.



Note 1: The minimum dwell period at each temperature extreme shall be the time required for the component to reach internal thermal equilibrium as determined from the first thermal vacuum qualification cycle or as determined by development testing or analysis, but in no case less than one hour.

Note 2: The dwell period at the maximum operating acceptance temperature limit may be performed with power on or off (preferred method – power on dwell – is shown).

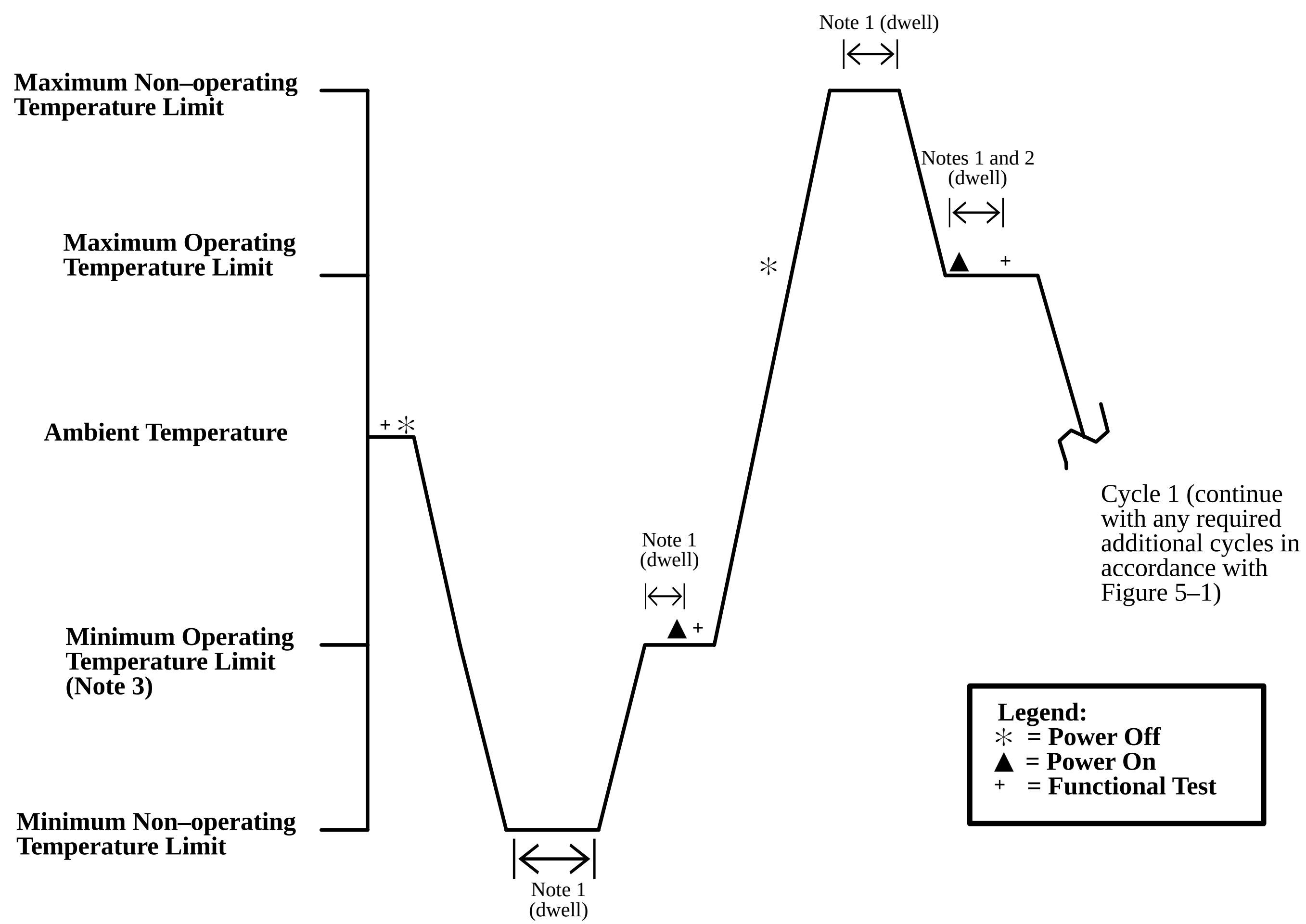
Note 3: The minimum operating temperature for acceptance thermal vacuum and thermal cycle testing shall be below 30 degrees F whenever possible.

Note 4: Power on/off cycles required at each temperature extreme for thermal cycles 2 through 7.

Note 5: The minimum sweep between the minimum and maximum acceptance limits for electrical and electronic components shall not be less than 100 degrees F.

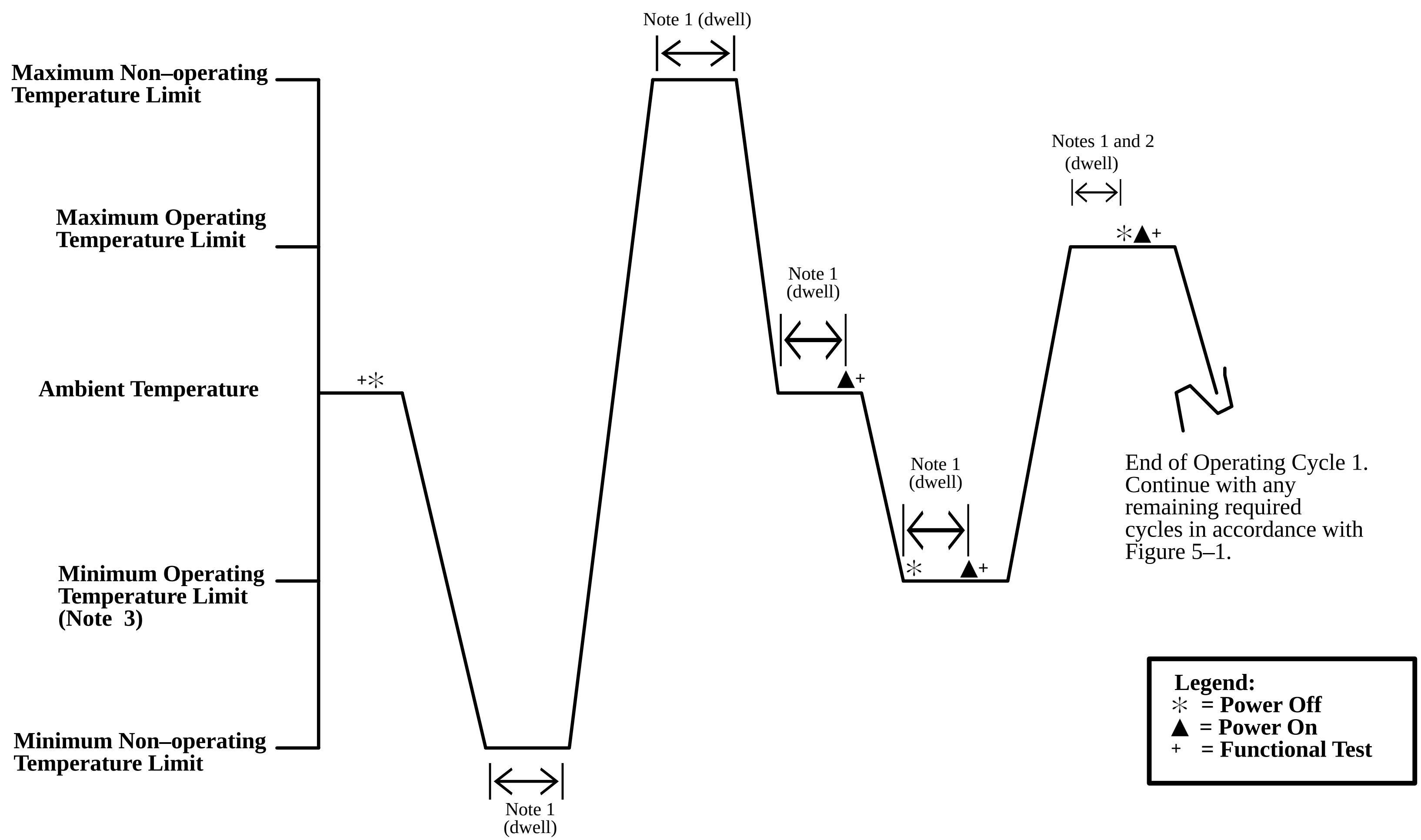
Note 6: For acceptance thermal cycle testing, the temperature transition rate between the minimum and maximum temperature limits shall be no less than 1.0 degree F (0.56 degree C) per minute.

FIGURE 5–1 COMPONENT THERMAL VACUUM AND THERMAL CYCLE ACCEPTANCE TESTS



Note 1: The minimum dwell period at each temperature extreme shall be the time required for the component to reach internal thermal equilibrium as determined from the first thermal vacuum qualification cycle or as determined by development testing or analysis, but in no case less than one hour.
Note 2: The dwell period at the maximum operating acceptance temperature limit may be performed with power on or off (preferred method – power on dwell – is shown).
Note 3: The minimum operating acceptance temperature limit shall be below 30 degrees F whenever possible.
Note 4: The minimum sweep between the minimum and maximum acceptance limits for electrical and electronic components shall not be less than 100 degrees F.
Note 5: For acceptance thermal cycle testing, the temperature transition rate between the minimum and maximum temperature limits shall be no less than 1.0 degree F (0.56 degree C) per minute.

FIGURE 5–1a COMBINED OPERATING/NON–OPERATING THERMAL VACUUM/ CYCLES



Note 1: The minimum dwell period at each temperature extreme shall be the time required for the component to reach internal thermal equilibrium as determined from the first thermal vacuum qualification cycle or as determined by development testing or analysis, but in no case less than one hour.
Note 2: The dwell period at the maximum operating acceptance temperature limit may be performed with power on or off (preferred method – power on dwell – is shown).
Note 3: The minimum operating acceptance temperature limit shall be below 30 degrees F wherever possible.
Note 4: The minimum sweep between the minimum and maximum acceptance limits for electrical and electronic components shall not be less than 100 degrees F.
Note 5: For acceptance thermal cycle testing, the temperature transition rate between the minimum and maximum operating temperature limits shall be no less 1.0 degree F (0.56 degree C) per minute.

FIGURE 5–1b SEPARATE OPERATING/NON–OPERATING THERMAL VACUUM/ CYCLES

C. Duration. A minimum of one temperature cycle shall be used. The component shall undergo a dwell period of at least one hour or a time sufficient for the component to reach internal thermal equilibrium as established by qualification testing, whichever is greater, at both the high and low temperature extremes with power off and then turned on. Dwell periods at the maximum predicted temperature may be performed with power on or off. See appendix A, PG1–159, for the exception to this paragraph. See appendix B, PG2–58 and PG2–149, for the exceptions to this paragraph.

NOTE: See appendix A, PG1–77 and PG1–83, for the exceptions existing prior to the current requirements.

See appendix A, PG1–21, PG1–30, PG1–76, PG1–94, PG1–149, PG1–195, PG1–220, PG1–232, and PG1–241, for the exceptions to these paragraphs. See appendix B, PG2–62, PG2–71, PG2–79, PG2–84, PG2–85, PG2–88, PG2–91, and PG2–127, for the exceptions to these paragraphs. See appendix C, PG3–51, PG3–102, and PG3–118, for the exceptions to these paragraphs. See appendix D, GFE–82 and GFE–90, for the exceptions to these paragraphs.

5.1.2.4 SUPPLEMENTARY REQUIREMENTS

Functional tests shall be conducted at the maximum and minimum predicted temperature levels during the first and last operating cycles after the dwell and after return of the component to ambient temperature. During the remainder of the test within the components operating temperature range, electrical and electronic components, including redundant circuits and paths, shall be monitored for failures and intermittences. Components with rotating equipment that use air as a lubricant should not be spinning when the atmosphere is removed. See appendix A, PG1–21, PG1–30, PG1–76, PG1–94, PG1–149, PG1–169, PG1–187, PG1–195, PG1–220, PG1–232, and PG1–241, for the exceptions to this paragraph. See appendix B, PG2–58 PG2–62, PG2–71, PG2–79, PG2–84, PG2–85, PG2–88, PG2–91, PG2–125, and PG2–127, for the exceptions to this paragraph. See appendix C, PG3–51, PG3–61, PG3–102, and PG3–118, for the exceptions to this paragraph. See appendix D, GFE–82, GFE–90, and GFE–91, for the exceptions to this paragraph.

5.1.3 THERMAL CYCLING TEST, COMPONENT ACCEPTANCE

5.1.3.1 PURPOSE

This test detects material and workmanship defects prior to installation of the component into a flight element by subjecting the component to thermal cycling.

5.1.3.2 TEST DESCRIPTION

A thermal cycle begins with the component at ambient temperature. With the component operating (power on) and while parameters are being monitored, the chamber temperature shall be reduced to bring the component to the specified low acceptance temperature level as measured at a representative location on the component such as the mounting point on the baseplate for conduction–dominated internal designs or at a representative location(s) on the case for radiation–controlled designs. The component shall be powered off and stabilized at the cold temperature. The component shall then be started after a dwell period sufficient to ensure that component internal thermal equilibrium has been achieved. With the component operating, the chamber temperature shall be increased to bring the component to the upper acceptance temperature level. After the component temperature has stabilized at the specified level, the component may be powered off or remain with power on. The component shall then undergo a dwell period sufficient to achieve internal thermal equilibrium. The component shall then be powered off (if not already so) and then powered back on after electrical circuits have discharged. The temperature of the chamber and component shall then be reduced to ambient conditions. This constitutes one thermal cycle. See appendix B, PG2–58 and PG2–147, for the exceptions to this paragraph. See appendix C, PG3–217, for the exception to this paragraph.

5.1.3.3 TEST LEVELS AND DURATION

A. Pressure. Ambient pressure shall be used. When unsealed components are being tested, the chamber shall be flooded with dry air or nitrogen to preclude condensation on and within the component at low temperature. See appendix B, PG2–56, for the exception to this paragraph.

B. Temperature (See Figure 5–1). The component shall be at the maximum acceptance limit during the hot portion of the cycle and at the minimum acceptance limit during the cold portion of the cycle. For components identified in Table 5–1 with note 4 there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) wherever possible. When the component's required non–operating temperature range exceeds its required operating temperature range, the component shall be exposed to one cycle of the non–operating maximum and minimum temperatures. This cycle may be combined with the required operating cycle in accordance with Figure 5–1a. If performed separately, it shall precede the operating cycle (Figure 5–1b). The component is not required to operate during exposure to non–operating temperatures. See appendix A, PG1–140, PG1–157, and PG1–180, for the exceptions to this paragraph. See appendix B, PG2–42, PG2–43, PG2–58, PG2–83, and PG2–137, for the exceptions to this paragraph. See appendix C, PG3–85, PG3–92, PG3–94, PG3–99, PG3–101, PG3–108, PG3–110, PG3–131, PG3–147, PG3–149, and PG3–194, for the exceptions to this paragraph. See appendix D, GFE–39, GFE–62, GFE–63, GFE–68, GFE–97, GFE–106, and GFE–109, for the exceptions to this paragraph.

C. Duration. The minimum number of temperature cycles shall be eight. This test may be performed in thermal vacuum and combined with the component acceptance thermal vacuum test provided that the temperature limits, number of cycles, rate of temperature change, and dwell times conform to this test. Each operating cycle shall have a 1 hour minimum dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on. The dwell time at the high and low levels shall be long enough to obtain internal thermal equilibrium. The transitions between the minimum and maximum operating temperature limits shall be at a rate no less than 1.0 degree F (0.56 degrees C) per minute. See appendix A, PG1–100, PG1–132, PG1–135, and PG1–159, for the exceptions to this paragraph. See appendix B, PG2–51, PG2–53, PG2–58, and PG2–149, for the exceptions to this paragraph. See appendix C, PG3–55, PG3–154, and PG3–162, for the exceptions to this paragraph. See appendix D, GFE–12, GFE–18, GFE–51, GFE–88, and GFE–106, for the exceptions to this paragraph.

See appendix A, PG1–149 and PG1–204, for the exceptions to these paragraphs. See appendix B, PG2–84 and PG2–85, for the exceptions to these paragraphs. See appendix C, PG3–45, PG3–49, PG3–58, PG3–63, and PG3–119, for the exceptions to these paragraphs. See appendix D, GFE–05, GFE–25, GFE–33, GFE–44, GFE–92, and GFE–112, for the exceptions to these paragraphs.

5.1.3.4 SUPPLEMENTARY REQUIREMENTS

Functional tests shall be conducted during the first and last operating thermal cycles after the dwell at the maximum and minimum predicted operating temperatures and after return of the component to ambient. Dwell periods at the high temperature extreme may be performed with power on or off. During the remainder of the test within the components operating temperature range, electrical components shall be cycled through various operational modes and parameters monitored for failures and intermittences. See appendix A, PG1–101, PG1–149, PG1–204, and PG1–228, for the exceptions to this paragraph. See appendix B, PG2–58, PG2–84, PG2–85, and PG2–125, for the exceptions to this paragraph. See appendix C, PG3–45, PG3–49, PG3–62, PG3–63, PG3–119, and PG3–191, for the exceptions to this paragraph. See appendix D, GFE–05, GFE–16, GFE–25, GFE–33, GFE–40, GFE–44, GFE–92, and GFE–112, for the exceptions to this paragraph.

5.1.4 RANDOM VIBRATION TEST, COMPONENT ACCEPTANCE

5.1.4.1 PURPOSE

This test detects material and workmanship defects prior to installation of the component into a flight system by subjecting the article to a dynamic vibration environment. Components receiving random vibration as an acceptance test environment are identified in Table 5–1.

5.1.4.2 TEST DESCRIPTION

The component shall be mounted to a rigid fixture through the normal mounting point of the component. The component shall be tested in each of three mutually perpendicular axes. Valves shall be pressurized to operating pressure for this test and monitored for internal pressure decay if pressurized during ascent. See appendix A, PG1–243, for the exception to this paragraph. See appendix B, PG2–106, for the exception to this paragraph.

5.1.4.3 TEST LEVELS AND DURATION

The test duration in each of the three orthogonal axes shall not be less than one minute per axis. The acceptance test spectrum input may be adjusted in the components resonant frequency zone(s) to reduce the component resultant level to within the test level spectrum. Component random vibration test levels and spectrums shall be the envelope of the following:

A. The maximum predicted flight level and spectrum minus 6 dB, but not less than a level derived from a 135 dB overall acoustic environment (whose spectrum is defined by NSTS 21000–IDD–ISS, paragraph 4.1.1.5). See appendix A, PG1–185 and PG1–188, for the exceptions to this requirement. See appendix B, PG2–66, for the exception to this requirement. See appendix D, GFE–79, for the exception to this requirement.

B. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime contractor. See appendix A, PG1–137 and PG1–243, for the exceptions to this requirement. See appendix B, PG2–48 and PG2–49, for the exceptions to this requirement. See appendix C, PG3–42, PG3–91, PG3–98, PG3–107, PG3–146, and PG3–214, for the exceptions to this requirement. See appendix D, GFE–54, for the exception to this requirement.

See appendix A, PG1–21, PG1–95, PG1–149, PG1–171, PG1–174, PG1–237, and PG1–240, for the exceptions to these paragraphs. See appendix B, PG2–45, PG2–75, PG2–112, and PG2–140, for the exceptions to these paragraphs. See appendix C, PG3–60, PG3–111, PG3–120, and PG3–222, for the exceptions to these paragraphs. See appendix D, GFE–06, GFE–27, GFE–35, GFE–45, GFE–65, GFE–81, GFE–83, GFE–86, and GFE–93, for the exceptions to these paragraphs.

5.1.4.4 SUPPLEMENTARY REQUIREMENTS

Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test. See appendix A, PG1–21, PG1–95, PG1–149, PG1–171, PG1–174, PG1– 230, and PG1– 244, for the exceptions to this paragraph. See appendix B, PG2–55, PG2–112, PG2–122, and PG2–140, for the exceptions to this paragraph. See appendix C, PG3–53, PG3–60, PG3–66, PG3–67, PG3–74, PG3–75, PG3–76, PG3–87, PG3–89, PG3–96, PG3–105, PG3–111, PG3–120, PG3–123, PG3–133, PG3–139, PG3–144, PG3–160, PG3–174, PG3–180, PG3–206, PG3–209, PG3–222, and PG3–224, for the exceptions to this paragraph. See appendix D, GFE–06, GFE–22, GFE–27, GFE–35, GFE–45, GFE–55, GFE–65, GFE–78, GFE–81, GFE–83, GFE–86, and GFE–93, for the exceptions to these paragraphs.

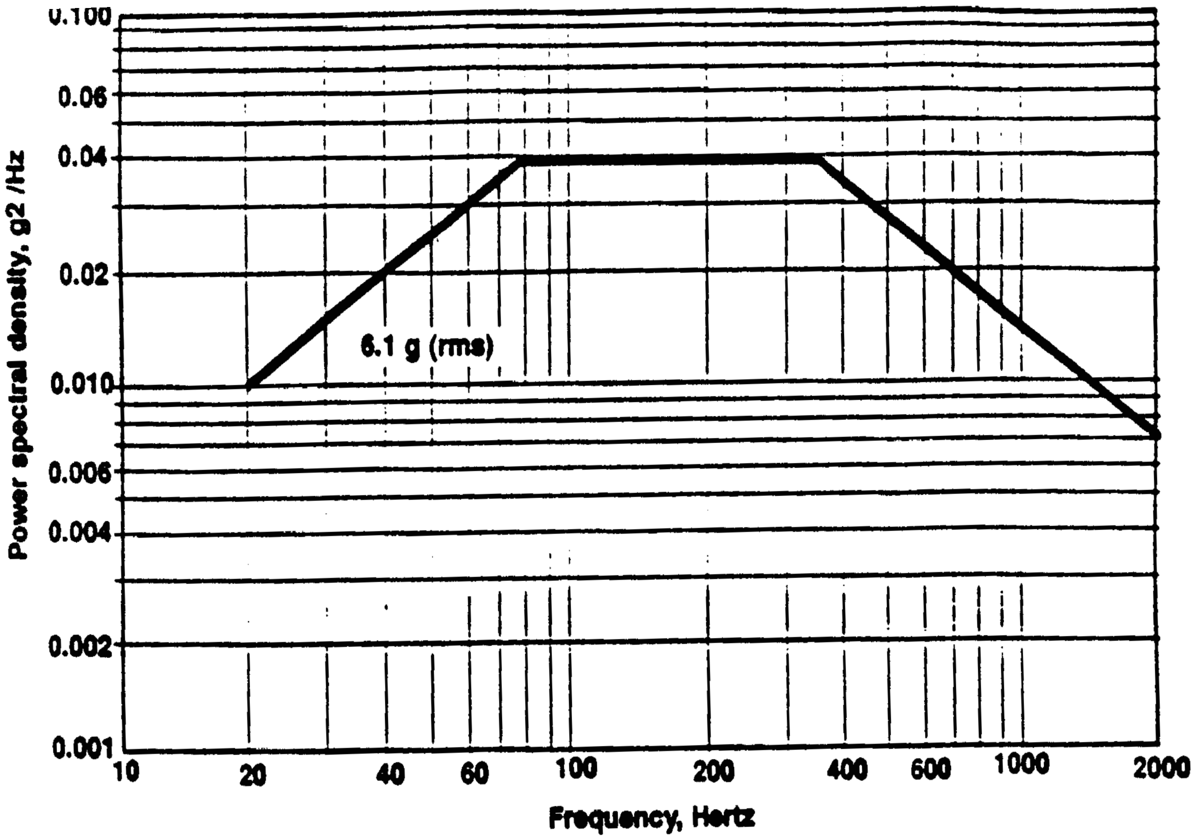
5.1.5 ACOUSTIC VIBRATION TEST, COMPONENT ACCEPTANCE

5.1.5.1 PURPOSE

This test may detect material and workmanship defects in large surface area–to–weight ratio components and assemblies that may be sensitive to acoustic excitation.

5.1.5.2 TEST DESCRIPTION

The component shall be installed in a reverberant acoustic cell capable of generating desired sound pressure levels. A uniform sound energy density throughout the chamber is desired. The configuration of the component during launch, such as deployed or stowed, shall be as it is during subjection to the flight dynamic environment. The preferred method of testing shall be with the component mounted on flight–type support structure and with ground–handling equipment removed.



Frequency Range (Hz)	Minimum Power Spectral Density (PS)
20	0.01 G^2/Hz
20–80	3 dB/Octave Slope
80–350	0.04 G^2/Hz
350–2000	–3 dB/Octave Slope
2000	0.007 G^2/Hz
Overall	6.1 grms

FIGURE 5–2 COMPONENT RANDOM VIBRATION WORKMANSHIP SCREENING TEST LEVEL

5.1.5.3 TEST LEVELS AND DURATION

The component shall be exposed to sound pressure levels equal to the maximum predicted levels minus 6 dB, but not less than a level derived from a 135 dB overall acoustic environment (whose spectrum is as defined by NSTS 21000–IDD–ISS, Table 4.1.1.5–1). The duration shall not be less than one minute. See appendix A, PG1–184, for the exception to this paragraph.

5.1.5.4 SUPPLEMENTARY REQUIREMENTS

A functional test shall be conducted before and following the acoustic test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. Components characterized by large ratios of surface area to volume, such as large antennas and solar arrays, cannot be tested in a manner that suitably simulates imposition of the service dynamic environment by employing mechanical vibration. For such component configurations, acoustic testing shall be required. When acoustic component testing is required, random vibration testing shall not be required. See appendix A, PG1–184, for the exception to this paragraph.

5.1.6 PRESSURE TEST, COMPONENT ACCEPTANCE

5.1.6.1 PURPOSE

This test detects material and workmanship defects that could result in failure of the pressure vessel or valves in usage.

5.1.6.2 TEST DESCRIPTION

A. Proof Pressure. For such items as pressure vessels, pressure lines, and fittings, the temperature of the component shall be consistent with the critical–use temperature and subjected to one cycle of proof pressure. A proof–pressure cycle shall consist of raising the internal pressure (hydrostatically or pneumatically, as applicable) to the proof pressure, maintaining it for 5 minutes, and then decreasing the pressure to ambient. Evidence of permanent set or distortion or failure of any kind shall indicate failure to pass the test.

B. Proof Pressure for Valves. With the valves in the open and closed positions, the proof pressure shall be applied for one cycle to the inlet port for 5 minutes. Following the 5 minute pressurization period, the inlet pressure shall be reduced to ambient conditions. The interior and exterior of the article shall be visually examined. Evidence of deformation shall indicate failure to pass the test. The test may be conducted at room ambient or elevated temperature.

5.1.6.3 TEST LEVELS

A. Temperature. The temperature shall be as specified in the test description. As an alternative, tests may be conducted at ambient room temperatures if the test pressures are suitably adjusted to account for temperature effects on strength and fracture toughness.

B. Proof Pressure. Proof pressure shall be as specified in SSP 30559, section 3.

See appendix A, PG1–149 and PG1–261, for the exceptions to these paragraphs. See appendix C, PG3–196, for the exception to these paragraphs. See appendix D, GFE–13 and GFE–110, for the exceptions to these paragraphs.

5.1.6.4 SUPPLEMENTARY REQUIREMENTS

Applicable safety standards shall be followed in conducting all tests. See appendix C, PG3–196, for the exception to this paragraph.

5.1.7 LEAK TEST, COMPONENT ACCEPTANCE

5.1.7.1 PURPOSE

This test demonstrates the capability of sealed and pressurized components to meet the leakage rate requirements specified in the component development specification.

5.1.7.2 TEST DESCRIPTION AND ALTERNATIVES

Component leak tests shall be made prior to initiation of, and following the completion of, component thermal vacuum, thermal cycle, and random or acoustic vibration acceptance tests. The leak test method employed shall have sensitivity and accuracy consistent with the components specified maximum allowable leakage rate. The sensitivity of the leak test, in particular, shall be quantitatively less than the minimum leakage rate to be detected by a factor of at least two to ensure reliability of measurements. When temperature potentially affects the sealing materials or surfaces, an evaluation of the hardware design and operational characteristics shall be performed and, if technically warranted, the leak test shall be conducted at the minimum and maximum acceptance temperature limits. If it is determined from the evaluation that a leak test at temperature limits is warranted on a component of a given level–of–assembly due solely to one or more lower tier components comprising the assembly, and it can be shown that all of those lower tier components receive an appropriate leak test at temperature limits as part of a lower level acceptance test, then the higher level–of–assembly does not require leak testing at temperature limits. Leak testing may be performed prior to component proof pressure testing only if approved by the responsible safety organization. In all cases, leak testing shall be conducted after the component proof pressure test. One of the following methods or another suitable leak test method in accordance with the criteria established in section 10 shall be used:

A. Method I (Immersion, to be used only as a pass/fail test; this method does not provide a quantitative measurement of component leakage rate). This method may be used for total or local internal–to–external leak testing of pressurized components and systems. Internal gas pressure shall be applied across the pressure boundary for a minimum duration of 15 minutes before the test liquid contacts the external surface. Lighting in the area to be examined shall be no less than 1000 lux or lumen/m^2 (100 foot–candles) in brightness. Illumination shall be free from shadows over the surface area under inspection. The observer shall place his eyes within 60 cm (2 feet) of the surface to be examined. Mirrors or magnifying glasses may be used to improve visibility of indications. The component shall be completely immersed in a liquid. The critical side or sides of interest of the component shall be in a horizontal plane facing up. There shall be no observed leakage during immersion (as evidenced by one or more bubbles emanating from the component).

B. Method II (Vacuum Chamber). This method may be used for total internal–to–external leak testing of pressurized components and systems. The component shall be placed in a vacuum chamber (bell jar) and tested for leakage with a leak detector appropriate for the tracer gas used. The vacuum chamber system leak test sensitivity shall be quantified and documented with a standard leak not to exceed the component maximum allowable leakage requirement. The component shall be charged with a known concentration of a tracer gas to the required pressure. Pressure shall be maintained until stabilization of the leak detector output is achieved (stabilization shall be defined as three consecutive readings no less than 5 minutes apart with no more than a 10 percent variation in the leak detector output from one measurement to the next, including the first and last measurements). Calibration data and leak detector initial and final readings shall be recorded. The final component leakage rate shall be recorded along with a minimum of 3 data points within a 15 minute duration to demonstrate stabilization in accordance with the definition above.

C. Method III (Pressure Change). The pressure decay technique may be used for total internal–to–external leak testing of pressurized components and systems. To improve the accuracy of this technique, a reference vessel connected to the pressurized component or system may be used. If ambient temperature changes, the component and reference vessel volumetric change shall be taken into account. The pressure rise technique may be used for total external–to–internal leak testing of sealed components and systems. The component internal pressure, barometric pressure, and ambient temperature (or temperature of the component) shall be monitored for the required time to determine the actual pressure drop or rise and the corresponding leakage rate. The pressure gauge/transducer shall have accuracy and sensitivity adequate to measure the minimum required pressure change. The tolerance/error associated with the total internal volume of the component and test fixture under pressure used for the leakage rate calculation shall be taken into account as a maximum positive value.

D. Method IV (Chemical Indicator, to be used only as a pass/fail test; this method does not provide a quantitative measurement of component leakage rate). A suitable indicator such as a dilute solution of phenolphthalein or other suitable color–change indicator such as colorimetric in accordance with ASTM 1066.95, Revision (2000) shall be applied to all seams, terminals, and pinch tubes subject to leakage of the working fluid. A change in the color of the indicator shall be an indication of a leak. After testing, the indicator shall be removed (e.g., with distilled water).

E. Method V (Detector Probe, to be used only as a pass/fail test for individual joints (e.g., welds and mechanical fittings); this method does not provide a quantitative measurement of component leakage rate). This detector probe is a semiquantitative technique used to detect and locate internal–to–external leaks in pressurized components and systems, and shall not be considered quantitative. The component shall be charged with a known concentration of a tracer gas to the required pressure. Prior to examination, the test pressure shall be held for a minimum duration of 30 minutes. Prior to examination, the tracer gas background shall be measured and the leak test setup shall be calibrated for the test by passing the detector probe tip across the orifice of a calibrated leak (the magnitude of the calibrated leak shall be equal to or less than the maximum allowable leakage rate). The resulting leak detector output shall be at least 40 percent above the tracer gas background. After the required soak time, the detector probe tip shall be passed over the test surface at the same scanning rate and distance used during the system calibration. The system calibration will be repeated every 60 minutes and any time test conductors/operators are changed. Any leak detector output above the established tracer gas background with allowance made for atmospheric tracer gas variations and leak detector drift, that in the aggregate do not exceed 40 percent of the tracer gas background, indicates a leak.

F. Method VI (Hood). This method may be used for total external–to–internal leak testing of sealed components and systems. The component internal volume shall be evacuated to a vacuum compatible with a tracer gas leak detector. The system sensitivity shall be determined by installing a standard leak at the furthest point from the leak detector. The external surfaces of the component shall be exposed to a verified concentration of a tracer gas. Pressure shall be maintained until stabilization of the leak detector output is achieved (stabilization shall be defined as three consecutive readings no less than 5 minutes apart with no more than a 10 percent variation in the leak detector output from one measurement to the next, including the first and last measurements). Calibration data and leak detector initial and final readings shall be recorded. The final component leakage rate shall be recorded along with a minimum of 3 data points within a 15 minute duration to demonstrate stabilization in accordance with the definition above.

G. Method VII (Tracer Probe, this method may be used to locate known external–to–internal leaks but shall not be used for verifying the specified allowable leakage rate of flight hardware). The component internal volume is evacuated to a vacuum compatible with a tracer gas leak detector. The tracer probe is connected to a source of 100 percent tracer gas with a valved opening at the other end for directing a stream of tracer gas over the component. Any indication of tracer gas above the background by the leak detector indicates a leak.

H. Method VIII (Accumulation). This method may be used for total internal–to–external leak testing of pressurized components and systems. The component shall be enclosed in a suitable enclosure. The enclosure shall be calibrated by placing a standard leak in the enclosure for a predetermined period of time. At the end of the time period, a detector probe shall be placed in the enclosure and the maximum leak detector response shall be recorded. The enclosure shall then be purged with nitrogen or air. The component shall be charged with a known concentration of a tracer gas to the required pressure. Prior to examination, the test pressure shall be held for a minimum duration of 30 minutes. The enclosure shall be purged with nitrogen or air and sealed. After the time period used for the calibration, the detector probe shall be placed in the enclosure. Calibration data and leak detector initial and final readings shall be recorded. The final component leakage rate shall also be recorded.

I. Method IX (Volumetric Displacement). This method may be used for total internal–to–internal leak testing of pressurized components such as valves, pressure regulators, or heat exchangers. One side of the component shall be pressurized to the required pressure while the other side across the internal barrier shall be sealed from the atmosphere and attached to a suitable device for the purposes of demonstrating volumetric displacement. This will be accomplished by either using a displacement of liquid or by moving a fluid meniscus along the graduations of the measuring device.

J. Method X (Leak Detector Direct Connection). This method may be used for total internal–to–internal leak testing of pressurized components such as valves, pressure regulators, or heat exchangers. One side of the component shall be charged with a known concentration of a tracer gas to the required pressure while the other side across the internal barrier shall be sealed from the atmosphere and attached to the leak detector. Pressure shall be maintained until stabilization of the leak detector output is achieved (stabilization shall be defined as three consecutive readings no less than 5 minutes apart with no more than a 10 percent variation in the leak detector output from one measurement to the next, including the first and last measurements). Calibration data and leak detector initial and final readings shall be recorded. The final component leakage rate shall be recorded along with a minimum of 3 data points within a 15 minute duration to demonstrate stabilization in accordance with the definition above.

K. Method XI (Local Vacuum Chamber). This method may be used for local internal–to–external leak testing of pressurized components and systems. The local vacuum chamber (bell jar) connected to the tracer gas leak detector shall be installed on the component area to undergo the leak test. The local vacuum chamber system sensitivity shall be quantified and documented with a standard leak not to exceed the component maximum allowable leakage rate requirement. The component shall be charged with a known concentration of a tracer gas to the required pressure. Pressure shall be maintained until stabilization of the leak detector output is achieved (stabilization shall be defined as three consecutive readings no less than 5 minutes apart with no more than a 10 percent variation in the leak detector output from one measurement to the next, including the first and last measurements). Calibration data and leak detector initial and final readings shall be recorded. The final component leakage rate shall be recorded along with a minimum of 3 data points within a 15 minute duration to demonstrate stabilization in accordance with the definition above.

NOTE: See appendix A, PG1–193, PG1–208, PG1–213, PG1–215, PG1–217, PG1–224, PG1–225, PG1–226, PG1–247, PG1–253, PG1–267, and PG1–269, for exceptions existing prior to the current requirements. See appendix B, PG2–69, PG2–77, PG2–108, PG2–115, PG2–117, and PG2–119, for exceptions existing prior to the current requirements. See appendix C, PG3–176, PG3–182, and PG3–201, for exceptions existing prior to the current requirements.

NOTE: See appendix C, PG3–175 and PG3–181 for the exceptions existing to the previous pressurized systems and components leak requirements in SSP 41172, Revision T (Method V).

5.1.7.3 TEST LEVELS AND DURATION

The leak tests shall be performed with the component pressurized at the maximum design pressure and then at the minimum design pressure if the seals are dependent upon pressure for proper sealing. Regardless of the method used, the test duration shall be sufficient to detect any significant leakage.

NOTE: See appendix B, PG2–102, for exceptions existing prior to the current requirements.

See appendix A, PG1–193, PG1–202, PG1–211, PG1–224, PG1–225, PG1–226, PG1–251, PG1–265, and PG1–269, for exceptions existing to the previous Fine Leak Test requirements in SSP 41172, Revision T. See appendix B, PG2–93, PG2–94, PG2–97, and PG2–104, for exceptions existing to the previous Fine Leak Test requirements in SSP 41172, Revision T. See appendix C, PG3–129, PG3–199, and PG3–229, for exceptions existing to the previous Fine Leak Test requirements in SSP 41172, Revision T.

NOTE: See appendix C, PG3–137, PG3–156, and PG3–164, for exceptions existing to the previous pressurized systems and components leak requirements in SSP 41172, Revision T.

See appendix A, PG1–149, PG1–249, PG1–259, and PG1–263, for the exceptions to these paragraphs. See appendix B, PG2–151, for an exception to these paragraphs. See appendix C, PG3–125, PG3–127, PG3–135, PG3–141, PG3–196, PG3–211, PG3–212, PG3–219, and PG3–227, for the exceptions to these paragraphs. See appendix D, GFE–13 and GFE–29, for the exceptions to these paragraphs.

5.1.7.4 SUPPLEMENTARY REQUIREMENTS

None.

5.1.8 BURN–IN TEST, COMPONENT ACCEPTANCE

5.1.8.1 PURPOSE

The purpose of the burn–in test is to detect material and workmanship defects which can result in early component failure.

5.1.8.2 TEST DESCRIPTION

This test shall be used to operationally stress electronic and electrical components to precipitate early life failures. The test may be performed at ambient temperature, elevated temperature, or under temperature cycling conditions, and the component shall be operating (power on) and parameters monitored for the duration of this test. See appendix C, PG3–216, for the exception to this paragraph.

5.1.8.3 TEST LEVELS AND DURATION

A. Pressure. This test may be conducted at ambient pressure or under vacuum conditions.

B. Temperature. The test may be conducted at constant ambient or elevated temperature, or under temperature cycling conditions. Temperatures shall be based on the control temperature sensor mounted to the component baseplate. See appendix B, PG3–215, for the exception to this paragraph.

(1) If an accelerated burn–in approach is selected by testing at a constant elevated temperature, the temperature shall not exceed that corresponding to the high operational temperature level during acceptance thermal cycle testing. When an accelerated constant elevated temperature test is selected, the following equation shall be used to determine the time acceleration factor:

$$F = \exp[(Ea/K)(1/Ta - 1/Tbi)]$$

where F = Time Acceleration Factor
Ea = Activation Energy (eV)
K = Boltzmann's constant (8.625E–05 eV/K)
Ta = ambient temperature (degrees K)
(For the purposes of this application, ambient temperature shall be considered to be 295.8 degrees K).
Tbi = elevated burn–in temperature (degrees K).

(2) When burn–in under temperature cycling conditions is chosen, the minimum and maximum test temperatures shall correspond to the minimum and maximum operational temperatures during acceptance thermal cycle testing as specified in 5.1.3, but the sweep range shall not be less than 100 degrees F. Each temperature transition between temperature extremes shall be at an average rate of 9 degrees F per minute (5 degrees C per minute) or greater. A minimum dwell period sufficient to achieve component internal thermal equilibrium, but not less than one hour, shall be performed for each cycle at both the high and low temperature extremes.

C. Duration. For constant temperature burn–in (either at ambient or accelerated via elevated temperature), the total operating time shall be equivalent to 300 hours at ambient temperature. Any operating time accumulated during other acceptance testing may be included in this 300 hour total. For the elevated temperature burn–in approach, the remaining time required to reach ambient 300 hours equivalency shall be determined by using the time acceleration factor equation above. For example, if after all acceptance testing (except burn–in) has been completed, the component has a total accumulated operating time of 60 hours, then an additional 240 hours (at ambient temperature) would be required to complete burn–in. This required remaining test time may be reduced by performing accelerated burn–in at an elevated temperature; for example 120 degrees F. Using the equation above (assume for this example Ea = 0.6), one can calculate a time acceleration factor of 6.7. Therefore, instead of performing an ambient temperature burn–in for 240 hours, an accelerated burn–in at 120 degrees F may be performed for 36 (240/6.7) hours. See appendix A, PG1–156, for the exception to this paragraph.

When burn–in under temperature cycling is conducted, a minimum of ten temperature cycles shall be conducted. These cycles are in addition to those required by the thermal cycle acceptance test as specified in 5.1.3. The total accumulated operating time on the component shall be a minimum of 100 hours. This total includes any operating time recorded during the acceptance test program. After completion of all other acceptance testing and the ten additional burn–in temperature cycles as specified in this paragraph, any additional operating time required to reach the 100 hour total may be accomplished at ambient temperature or may be accelerated by testing at a constant elevated temperature as defined above.

See appendix A, PG1–233, for the exception to these paragraphs. See appendix D, GFE–01, GFE–07, GFE–46, and GFE–107, for the exceptions to these paragraphs.

NOTE: See appendix B, PG2–47, for the exception existing prior to the current requirements.

See appendix D, GFE–36, for the exception to these paragraphs.

5.1.8.4 SUPPLEMENTARY REQUIREMENTS

The accelerated constant elevated temperature burn–in approach shall only be applicable to the time remaining, after all other acceptance testing, to achieve the total operating time requirement (300 hours or 100 hours as applicable). The Time Acceleration Factor equation shall not be applied to the time spent at elevated temperature during any acceptance thermal vacuum or thermal cycle (including the thermal cycle burn–in of this paragraph) testing.

For the accelerated constant elevated temperature burn–in approach, the activation energy for the component shall be calculated using weighted averaging. If the quantity (Q_i) of a given type of part internal to the component (resistor, capacitor, diode, etc.) has a specific activation energy (Ea_i), the component activation energy (Ea) shall be determined by weighted averaging using the following equation:

$$Ea = \frac{\sum (Q_i)(Ea_i)}{\sum Q_i}$$

For example, consider a component comprised of four different parts with the following activation energies:

Part	Quantity of Internal Items of Part (Q_i)	Part Activation Energy (Ea_i)
A	4	0.6
B	1	0.3
C	2	0.4
D	3	0.8

The component activation energy shall be determined by a weighted average technique as follows:

$$Ea = \frac{4(0.6) + 1(0.3) + 2(0.4) + 3(0.8)}{10} = 0.59$$

Typical activation energies for various parts are:

Part	Ea
Bipolar digital ICs	0.8
Bipolar linear ICs	0.7
MOS digital ICs	0.6
Resistors	0.56
Capacitors	0.6
Transistors/diodes	0.96
Transformers/inductors	0.5

For Commercial Off–the–Shelf (COTS) components where no parts list is available, the activation energy (Ea) to be used in the acceleration factor equation shall be 0.3. For COTS components where a parts list is available, the activation energy shall be that corresponding to the part(s) in the box having the lowest activation energy.

Functional tests in accordance with 5.1.1 shall be performed prior to and after completion of this burn–in test.

See appendix D, GFE–36, for the exception to these paragraphs.

5.1.9 OXYGEN COMPATIBILITY TEST, COMPONENT ACCEPTANCE

The oxygen system components shall undergo Oxygen Compatibility Acceptance Test as noted in Table 5–1a.

TABLE 5–1a OXYGEN COMPONENTS REQUIRING ACCEPTANCE TESTING

Component	Testing
Hard Line (rigid metal tubing)	
Metal Flex Hose	
Metal Flex Hose (>/=3,000 psia)	X
Metal Fluid Fitting with all metal seals	
Self–Sealing Quick Disconnect	X
Valve	X
Pressure Relief Valve	X
Temperature Sensor	X
Pressure Sensor	X
Nonmetal Lining Flex Hose	X
Fluid Fitting with nonmetal seals	X
Pressure Regulator	X
Metal Pressure Vessels	

5.1.9.1 PURPOSE

This test detects material and workmanship defects that could result in ignition of the component when pressurized with oxygen.

5.1.9.2 TEST DESCRIPTION

Oxygen system components shall be exposed to oxygen at Maximum Design Pressure (MDP) as defined in SSP 30559. Functional test, other than leakage, shall be conducted while the component is pressurized with oxygen at MDP (functional tests include opening and closing a valve, connecting and disconnecting a quick disconnect, etc.). Cleanliness shall be maintained to the level specified in the component specification. Hydrocarbon detection analysis shall be performed as specified in MSFC–PROC–404 prior to the Oxygen Compatibility Acceptance Test for components exposed to nonoxygen compatible solvents as an assembly. Total hydrocarbon count shall not exceed 5 parts per million.

Each component shall be subjected to 10 oxygen pressurization cycles from ambient pressure (10 to 15 psia) to MDP within 100 milliseconds. The component shall be maintained at MDP for at least 30 seconds following each pressurization cycle. Each component shall be subjected to oxygen flow in both the forward and reverse flow directions, where reversible flow is within the operational capability of the component. See appendix D, GFE–108, for the exception to this requirement.

Visual inspection shall be performed after conduct of the Oxygen Compatibility Acceptance Test and shall be verified to the level specified in the component specification. If disassembly of the component listed in Table 5–1a occurs after the Oxygen Compatibility Acceptance Test, the Oxygen Compatibility Acceptance Test must be redone in full. Functional test and leak test (as specified in the component specification) shall be conducted after the Oxygen Compatibility Acceptance Test.

5.1.10 CORONA/ARCING TEST, COMPONENT ACCEPTANCE

5.1.10.1 PURPOSE

The purpose of this test is to ensure no detrimental corona/arcing occurs in unsealed electrical/electronic components which are required to operate during ascent/descent or during a depressurization/repressurization event. Electrical/electronic components utilizing a sealed chassis design or components which are powered under space vacuum conditions only do not require corona/arcing testing. For external components, this test should be performed as part of the thermal vacuum acceptance test.

5.1.10.2 TEST DESCRIPTION

The component shall be placed in a vacuum chamber at ambient conditions. With the component powered–up and monitored, the chamber pressure shall then be reduced to the specified low pressure level. The time for change of pressure between 20 Torr and 0.001 Torr (or vice versa) shall be at least 10 minutes to allow sufficient time in the region of critical pressure. The worst–case design and operating condition(s) (either by normal component design functions or externally induced transient conditions) producing the most likely corona/arcing potential shall be duplicated in the region of critical pressure. Full functional testing of the component in the critical pressure region shall not be required unless necessary to adequately demonstrate absence of corona/arcing. This corona/arcing monitoring may be performed during either depressurization or repressurization of the vacuum chamber. The chamber shall then be returned to ambient pressure.

5.1.10.3 TEST LEVELS AND DURATION

A. Pressure. The pressure shall be reduced from ambient to below 0.001 Torr.

B. Temperature. Ambient temperature shall be used.

5.1.10.4 SUPPLEMENTARY REQUIREMENTS

An assessment shall be made of the component design and operational characteristics to establish proper corona monitoring techniques. MSFC–STD–531 may be used for guidance.

5.2 FLIGHT ELEMENT ACCEPTANCE TESTS

The flight element acceptance test baseline consists of all the required tests specified in Table 5–2. Flight element tests include assembly elements (i.e. PIT, Node, PVM) where the flight element is not launched as a single entity. Where it is impractical to test flight elements as a single entity, testing of major assemblies that constitute the flight element may be utilized with the appropriate analyses, simulations, and/or simulators to satisfy this requirement. If the flight element is controlled by on–board data processing, the flight software shall be resident in the on–board computer for these tests. The verification of the operational requirements shall be demonstrated. Functional performance requirements shall also be verified.

TABLE 5–2 FLIGHT ELEMENT ACCEPTANCE TESTS

Tests	Notes
Functional	(1)
Toxic Off–Gassing	(2)
Acoustic Noise	(2)
EMC	–
Pressure/Leak	
Mass Properties	–

Notes:
(1) Electrical and mechanical functional tests shall be conducted following pressure/leak test.
(2) Applies only to pressurized elements and racks.

5.2.1 FUNCTIONAL TEST, FLIGHT ELEMENT ACCEPTANCE

5.2.1.1 PURPOSE

This test verifies that the electrical and mechanical performance of the flight element meets the performance requirements of the specifications and detects any anomalous condition.

5.2.1.2 MECHANICAL FUNCTIONAL TEST

Mechanical devices, valves, deployables, and separable entities shall be functionally tested with the flight element in the ascent, orbital, or recovery configuration appropriate to the function. Alignment checks shall be made where appropriate. The maximum and minimum limits of acceptable performance shall be determined with respect to mechanics, time, and other applicable requirements. For each mechanical operation, such as appendage deployments, tests shall demonstrate positive margins of strength, torque margins, and that they function at conditions above and below specified operational limits. Where operation in a 1–G environment cannot be performed, a suitable ground–test fixture may be utilized to permit operation and evaluation of the devices. Fit checks shall be made of the flight element physical interfaces with other flight elements and the launch vehicle by means of master gauges or interface assemblies. The most adverse tolerance accumulation shall be considered in these fit checks. Tests are necessary only at nominal performance requirements and ambient environment, unless otherwise specified in the applicable Prime Item Development Specification.

5.2.1.3 ELECTRICAL FUNCTIONAL TEST

The flight element shall be in its flight configuration with all components and subsystems connected except pyrotechnic devices. This test shall verify the integrity of all electrical circuits in the flight element, including redundant paths, by the application of an initiating stimulus and the confirmation of the successful completion of the event. The test shall be designed to operate all components, primary and redundant, and all commands shall be exercised. The operation of all thermal control components, such as heaters and thermostats, shall be verified by test. The test shall demonstrate that all commands having preconditioning requirements (such as enable, disable, specific equipment configuration, specific command sequence, etc.) cannot be executed unless the preconditions are satisfied. Equipment performance parameters (such as power, voltage, gain, frequency, command, and data rates) shall be varied over specification ranges to demonstrate the performance margins. Autonomous functions shall be verified to occur when the conditions exist for which they are designed. The flight element main bus shall be continuously monitored by a power transient monitor system. All telemetry monitors shall be verified, and pyrotechnic circuits shall be energized and monitored. A segment of this test shall operate the flight element through an ascent and mission profile with all events occurring in actual flight sequence. Tests are necessary only at nominal performance requirements and ambient environment, unless otherwise specified in the applicable Prime Item Development Specification.

5.2.1.4 SUPPLEMENTARY REQUIREMENTS

Mechanical and electrical functional tests shall be conducted prior to and after each of the flight element environmental tests to detect equipment anomalies and to assure that performance meets the requirements of the specification. These tests do not require the mission profile sequence. Data analysis to verify the adequacy of the testing and the validity of the data shall be completed before disconnection from a particular environmental test configuration so that any required retesting can be readily accomplished.

5.2.2 ELECTROMAGNETIC COMPATIBILITY TEST, FLIGHT ELEMENT ACCEPTANCE

Limited EMC acceptance testing shall be accomplished on flight elements to check on EMC compliance indicated during flight element EMC qualification testing (see 4.4.2) and to verify that changes have not occurred on successive production equipment. The limited tests shall include measurements of power bus ripple, peak transients, and monitoring of critical circuit parameters.

5.2.3 PRESSURE/LEAK TEST, FLIGHT ELEMENT ACCEPTANCE

5.2.3.1 PURPOSE

This test demonstrates the capability of pressurized structure and fluid subsystems to meet the flow, pressure, and leakage rate requirements specified in the flight element specification.

5.2.3.2 TEST DESCRIPTION

The requirements of the flight element including flow, leakage, and regulation shall be measured while operating applicable valves, pumps, and motors. The flow checks shall verify that the plumbing configurations are adequate. Checks for subsystem cleanliness, moisture levels, and pH shall be made as applicable. Where sealed or pressurized subsystems are assembled with other than brazed or welded connections, the specified torque values for these connections shall be verified prior to leak tests. In addition to the high–pressure test, propellant tanks and thruster valves shall be tested for leakage under propellant–servicing conditions. The system shall be evacuated to the internal pressure normally used for propellant loading and the systems pressure monitored for any indication of leakage.

A suitable leak test method in accordance with section 10 herein shall be used.

5.2.3.3 TEST LEVELS AND DURATION

The flight element shall be pressurized to proof pressure and held until all strain and deflection data are recorded, and then shall be reduced to the maximum design pressure. The proof pressure shall be as specified in SSP 30559, Section 3. Inspection for leakage after the proof test shall be at no less than 14.7 psid. The duration of the evacuated propulsion system leak test shall not exceed the time that this condition is normally experienced during propellant loading. These element–level internal proof pressure and leak tests are only applicable to flight elements with pressurized internal volumes.

All flight element pressurized fluid systems shall be proof pressure tested at the final system assembly level. All flight element pressurized fluid systems shall be leak tested at the final system assembly level at the system maximum design pressure after successful system proof pressure testing. Where it is impossible or impractical to perform system–level proof and leak testing in the final assembly configuration due to design or manufacturing limitations, proof pressure and leak testing at the highest assembly level practical shall be required. In this case, the proof factor shall be appropriate for the level of assembly during the test, and the overall fluid system proof and leak test strategy shall require approval by the ISS Program.

See appendix B, PG2–151, for an exception to these paragraphs.

5.2.3.4 SUPPLEMENTARY REQUIREMENTS

Applicable safety standards shall be followed in conducting all tests. Leak tests shall be conducted only after satisfactory proof pressure tests have been completed. Leak detection and measurement procedures may require vacuum chambers, enclosure of the entire flight element or localized areas, or other special techniques to achieve the required accuracy.

5.2.4 TOXIC OFF–GASSING TEST, FLIGHT ELEMENT ACCEPTANCE

5.2.4.1 PURPOSE

The purpose of the Toxic Off–Gassing test is to demonstrate that the flight element does not emit toxic vapors that may build up to harmful levels for personnel. This test is applicable to habitable pressurized elements which are sealed and have no atmospheric scrubbing capability prior to crew entry.

5.2.4.2 TEST DESCRIPTION

Air samples shall be taken from representative locations inside the element and shall be taken in pairs. When possible, all samples shall be taken from an external sampling port. A baseline pair of samples, marking the beginning of the test, shall be obtained immediately prior to element hatch closure (internal volume sealed). Provisions for air circulation without scrubbing shall be provided. When an external sampling port is available, a second pair of samples shall be obtained midway through the test. A final pair of samples shall be obtained at the end of the test and prior to any purge air being introduced. When samples are taken from an external sampling port, internal atmospheric circulation without scrubbing shall be provided to ensure homogeneous air samples. A standard ambient atmosphere shall be verified prior to taking the first samples. When no external sampling port is available, two pairs of samples shall be taken; one immediately prior to element hatch closure and one immediately upon reentry into the element at the end of the test. All samples shall be taken from the same general location.

5.2.4.3 TEST LEVELS AND DURATION

The test duration shall be, at a minimum, one–fifth the elapsed time between final element closeout and on–orbit crew entry. Three pairs of samples shall be taken whenever possible. The time between taking the two samples of a given pair shall not exceed 5 minutes. The time from hatch opening (breaking of seal) to obtaining the final pair of test samples shall not exceed 15 seconds.

5.2.4.4 SUPPLEMENTARY REQUIREMENTS

The flight element shall be, to the greatest extent possible, in its fully outfitted configuration for this test. If more than 25 percent (by mass) of the hardware is missing from the element at the time of the test, than the estimated mass of the missing hardware shall be documented and provided with the air samples. Systems, subsystems, and components of the flight element shall be powered to the greatest extent practicable during the conduct of the test. Canisters of 350 or 500 ml volume, with passivated internal surfaces, shall be used to obtain samples. Each canister shall contain a minimum of 3 surrogate standards for assessing the accuracy of the sampling and analysis process. The canisters shall be cleaned and proofed to 5 parts per billion for each potential contaminant that could be present when the sample is taken. If a sample line is required to obtain a sample, the line shall be made of an inert material and thoroughly purged prior to the air sample being withdrawn. Test duration shall be recorded and be to the nearest hour for tests lasting five days or longer, or to the nearest minute for shorter duration tests.

Samples shall be returned for analysis to the NASA/JSC Toxicology Laboratory within three days of their acquisition. As a minimum, analytical methods shall be at least equivalent to the standards promulgated for U.S. Environmental Protection Agency method TO14; however, the list of compounds shall include all of those on the target list for flight sample analysis. Any contaminants found in the samples shall be compared to their 7–day Spacecraft Maximum Allowable Concentrations as contained in JSC 20584. T values shall be calculated for each test point during the test period.

5.2.5 ACOUSTIC NOISE GENERATION TEST FLIGHT ELEMENT ACCEPTANCE

5.2.5.1 PURPOSE

The purpose of the Acoustic Noise test is to demonstrate that the flight hardware does not produce acoustic noise levels that are detrimental to the crew health and safety.

5.2.5.2 TEST DESCRIPTION

The systems and subsystems of the crew module shall be operated to duplicate the mission profiles that provide the most severe acoustic noise environments. Acoustic noise measurements shall be made in a manner to duplicate flight conditions in accordance with the sections on Instrumentation and Measurement in MIL–STD–1474.

5.2.6 MASS PROPERTIES, FLIGHT ELEMENT ACCEPTANCE

5.2.6.1 PURPOSE

The purpose of the Mass Properties Test is to record weight and center of gravity data.

5.2.6.2 TEST DESCRIPTION

Each integrated subelement shall be weight verified by actual measurement with an accuracy within +/– 0.2 percent of actual measured weight. Each integrated subelement shall be two axis (minimum) center of gravity verified by actual measurement within +/– 0.5 inches as measured from coordinate origin.

5.2.6.3 SUPPLEMENTARY REQUIREMENTS

After test completion an error analysis shall be conducted to document the accuracy of the measurements.

6.0 PROTOFLIGHT TEST

6.1 USE OF QUALIFICATION ASSEMBLIES FOR FLIGHT (PROTOFLIGHT)

When the qualification assemblies/components are planned for flight use, the assembly/component qualification test program shall be modified from that specified for dedicated qualification articles to reduce stress levels. Subsequent assemblies/components shall be subjected to identical protoflight tests. The flight element in which these qualification assemblies (protoflight) are installed shall be acceptance or protoflight tested in accordance with this document.

See appendix A, PG1–97, PG1–102, PG1–105, PG1–108, PG1–111, PG1–113, PG1–116, PG1–119, PG1–122, PG1–125, PG1–128, and PG1–130, for the exceptions to this paragraph. See appendix B, PG2–89, for the exception to this paragraph.

6.1.1 ASSEMBLY/COMPONENTS PROTOFLIGHT TESTS

When there is no dedicated qualification test article and all production articles are intended for flight usage, the test shall be the same (as defined in paragraph 4.2 for component qualification) with the following exceptions:

A. For the thermal vacuum tests, the temperature extremes shall be 10 degrees F (5.6 degrees C) beyond the minimum and maximum predicted temperatures. The minimum number of cycles shall be one. For electrical/electronic components, the minimum operational temperature sweep shall be 100 degrees F (55.6 degrees C). See appendix A, PG1–141, PG1–206, and PG1–239, for the exceptions to this paragraph. See appendix B, PG2–113, for the exception to this paragraph. See appendix E, BOE–01 for the exception to this paragraph.

B. For the thermal cycling test, the temperature extremes shall be 10 degrees F (5.6 degrees C) beyond the maximum and minimum predicted temperatures. The minimum number of cycles shall be eight. For electrical/electronic components, the minimum operational temperature sweep shall be 100 degrees F (55.6 degrees C). See appendix A, PG1–141, PG1–153, PG1–154, PG1–160, and PG1–260, for the exceptions to this paragraph. See appendix B, PG2–113, PG2–130, PG2–131, PG2–135, PG2–136, and PG2–145, for the exception to this paragraph. See appendix D, GFE–31, GFE–69, GFE–85, and GFE–94, for the exceptions to this paragraph.

C. For the acoustic vibration test, the test level shall be the maximum predicted flight level, but not less than a level derived from an acoustic environment of 141 dB overall, (whose spectrum is defined by NSTS 21000–IDD–ISS, Table 4.1.1.5–1). The duration of the test shall be limited to one minute. See appendix A, PG1–86, PG1–91, PG1–92, PG1–155, and PG1–167, for the exceptions to this paragraph. See appendix C, PG3–103 and PG3–171, for the exceptions to this paragraph.

D. For the random vibration test, the test level and spectrum shall be the envelope of the following:

   (1) The maximum predicted flight level, but no less than a level derived from an acoustic environment of 141 dB overall, (whose spectrum is defined by NSTS 21000–IDD–ISS, Table 4.1.1.5–1), and;

   (2) The minimum workmanship screening level and spectrum as defined by Figure 5–2.

The test duration shall be limited to one minute in each of the three orthogonal axes. See appendix E, BOE–02 for the exception to this paragraph.

E. For the pyrotechnic shock test, the shock spectrum shall be 3 dB greater than the maximum predicted level. See appendix A, PG1–162, for the exception to this paragraph.

F. For the pressure test, only proof pressure tests per 4.2.10.2 shall be conducted.

Protoflight electrical and electronic components shall also undergo a burn–in test as defined in paragraph 6.1.2 below.

See appendix A, PG1–84, PG1–96, PG1–106, PG1–109, PG1–114, PG1–117, PG1–120, PG1–123, PG1–126, PG1–131, PG1–134, PG1–142, PG1–143, PG1–144, PG1–145, PG1–146, PG1–150, PG1–151, PG1–161, PG1–163, PG1–164, PG1–165, PG1–166, PG1–175, PG1–176, PG1–177, PG1–178, PG1–179, PG1–197, PG1–198, PG1–199, PG1–200, PG1–238, PG1–271, and PG1–272, for the exceptions to these paragraphs.

See appendix C, PG3–41, PG3–56, PG3–113, PG3–114, PG3–150, PG3–151, PG3–165, PG3–166, PG3–167, PG3–168, PG3–169, PG3–170, PG3–184, PG3–185, PG3–186, PG3–187, PG3–188, PG3–189, PG3–203, and PG3–204, for the exceptions to these paragraphs.

See appendix D, GFE–30, GFE–47, GFE–48, GFE–70, GFE–71, GFE–72, GFE–73, GFE–74, GFE–84, and GFE–95, for the exceptions to these paragraphs.

6.1.2 ASSEMBLY/COMPONENTS PROTOFLIGHT BURN–IN TESTS

6.1.2.1 PURPOSE

The purpose of the burn–in test is to detect material and workmanship defects which can result in early component failure.

6.1.2.2 TEST DESCRIPTION

This test shall be used to operationally stress electronic and electrical components to precipitate early life failures. The test may be performed at ambient temperature, elevated temperature, or under temperature cycling conditions, and the component shall be operating (power on) and parameters monitored for the duration of this test.

6.1.2.3 TEST LEVELS AND DURATION

A. Pressure. This test may be conducted at ambient pressure or under vacuum conditions.

B. Temperature. The test may be conducted at constant ambient or elevated temperature, or under temperature cycling conditions. Temperatures shall be based on the control temperature sensor mounted to the component baseplate.

(1) If an accelerated burn–in approach is selected by testing at a constant elevated temperature, the temperature shall not exceed that corresponding to the high operational temperature level during protoflight thermal cycle testing. When an accelerated constant elevated temperature test is selected, the following equation shall be used to determine the time acceleration factor:

$$F = \exp[(Ea/K)(1/Ta - 1/Tbi)]$$

where F = Time Acceleration Factor
Ea = Activation Energy (eV)
K = Boltzmann's constant (8.625E–05 eV/K)
Ta = ambient temperature (degrees K)
(For the purposes of this application, ambient temperature shall be considered to be 295.8 degrees K).
Tbi = elevated burn–in temperature (degrees K).

(2) When burn–in under temperature cycling conditions is chosen, the minimum and maximum test temperatures shall correspond to the minimum and maximum operational temperatures during protoflight thermal cycle testing as specified in 6.1.1B, but the sweep range shall not be less than 100 degrees F. Each temperature transition between temperature extremes shall be at an average rate of 9 degrees F per minute (5 degrees C per minute) or greater. A minimum dwell period sufficient to achieve component internal thermal equilibrium, but not less than one hour, shall be performed for each cycle at both the high and low temperature extremes.

C. Duration. For constant temperature burn–in (either at ambient or accelerated via elevated temperature), the total operating time shall be equivalent to 300 hours at ambient temperature. Any operating time accumulated during other protoflight testing may be included in this 300 hour total. For the elevated temperature burn–in approach, the remaining time required to reach ambient 300–hours equivalency shall be determined by using the time acceleration factor equation above. For example, if after all protoflight testing (except burn–in) has been completed, the component has a total accumulated operating time of 60 hours, then an additional 240 hours (at ambient temperature) would be required to complete burn–in. This required remaining test time may be reduced by performing accelerated burn–in at an elevated temperature; for example 120 degrees F. Using the equation above (assume for this example Ea = 0.6), one can calculate a time acceleration factor of 6.7. Therefore, instead of performing an ambient temperature burn–in for 240 hours, an accelerated burn–in at 120 degrees F may be performed for 36 (240/6.7) hours.

When burn–in under temperature cycling is conducted, a minimum of ten temperature cycles shall be conducted. These cycles are in addition to those required by the thermal cycle protoflight test as specified in 6.1.1B. The total accumulated operating time on the component shall be a minimum of 100 hours. This total includes any operating time recorded during the protoflight test program. After completion of all other protoflight testing and the ten additional burn–in temperature cycles as specified in this paragraph, any additional operating time required to reach the 100–hour total may be accomplished at ambient temperature or may be accelerated by testing at a constant elevated temperature as defined above.

6.1.2.4 SUPPLEMENTARY REQUIREMENTS

The accelerated constant elevated temperature burn–in approach shall only be applicable to the time remaining, after all other protoflight testing, to achieve the total operating time requirement (300 hours or 100 hours as applicable). The Time Acceleration Factor equation shall not be applied to the time spent at elevated temperature during any protoflight thermal vacuum or thermal cycle (including the thermal cycle burn–in of this paragraph) testing.

For the accelerated constant elevated temperature burn–in approach, the activation energy for the component shall be calculated using weighted averaging. If the quantity (Qi) of a given type of part internal to the component (resistor, capacitor, diode, etc.) has a specific activation energy (Eai), the component activation energy (Ea) shall be determined by weighted averaging using the following equation:

$$Ea = \frac{\sum (Q_i)(Ea_i)}{\sum Q_i}$$

For example, consider a component comprised of four different parts with the following activation energies:

Part	Quantity of Internal Items of Part (Q_i)	Part Activation Energy (Ea_i)
A	4	0.6
B	1	0.3
C	2	0.4
D	3	0.8

The component activation energy shall be determined by a weighted average technique as follows:

$$Ea = \frac{4(0.6) + 1(0.3) + 2(0.4) + 3(0.8)}{10} = 0.59$$

Typical activation energies for various parts are:

Part	Ea
Bipolar digital ICs	0.8
Bipolar linear ICs	0.7
MOS digital ICs	0.6
Resistors	0.56
Capacitors	0.6
Transistors/diodes	0.96
Transformers/inductors	0.5

For COTS components where no parts list is available, the activation energy (Ea) to be used in the acceleration factor equation shall be 0.3. For COTS components where a parts list is available, the activation energy shall be that corresponding to the part(s) in the box having the lowest activation energy.

Functional tests shall be performed prior to and after completion of this burn–in test.

6.1.3 PROTOFLIGHT ASSEMBLY CERTIFICATION FOR FLIGHT

Upon completion of the protoflight test program, the assembly test history shall be reviewed for excessive test time and potential fatigue–type failures to determine if the article is acceptable for flight or if refurbishment is required. Protoflight testing of Space Station structures and assemblies shall be in accordance with SSP 30559.

6.2 USE OF THE FLIGHT (PROTOFLIGHT) ELEMENT FOR QUALIFICATION

When the flight element is used for the qualification tests (protoflight), the flight element qualification test levels and durations shall be reduced as defined below. The components installed in this flight element shall be qualified to protoflight or qualification levels, as applicable, and shall be acceptance tested.

6.2.1 FLIGHT ELEMENT PROTOFLIGHT TESTS

If the flight element acceptance tests and qualification tests are to be combined, the acceptance tests required by this document are waived, and each flight element shall be tested to the qualification test baseline.

For the flight element acoustic vibration qualification test, the test level shall be the maximum predicted level plus but not less than 141 dB overall, (whose spectrum is defined by NSTS 21000–IDD–ISS, paragraph 4.1.1.5). The duration of the test shall be limited to one minute.

6.2.2 FLIGHT ELEMENT CERTIFICATION FOR FLIGHT

Upon completion of the protoflight test program, the test article test history shall be reviewed for excessive test time and potential fatigue–type failure to determine whether the article is acceptable for flight or whether refurbishment is required. If significant modifications are incorporated or numerous components are refurbished or replaced with new components subsequent to protoflight testing, the acceptance tests specified in 5.2 shall be required prior to launch certification.

7.0 USE OF THE QUALIFICATION TEST ARTICLE FOR FLIGHT

When a dedicated test article used for qualification testing is subsequently planned for flight use, components shall be replaced based on a detailed post–qualification inspection and analysis with components that have passed the acceptance tests. The test article is certified for flight when it satisfactorily completes the specified acceptance tests.

8.0 RETEST

8.1 GENERAL

Retest is the repeat of all or part of previously conducted tests because of events such as a failure, design or manufacturing process changes, change in manufacturing source or facility, or changes in predicted service environments. Discrepancies may occur at any point in the qualification or acceptance test sequences or other times during the component's service life.

A failure analysis shall be conducted to determine the cause of all failures and to determine if there are any generic or lot–related problems that could affect other tests or flight articles. If it is determined that the component failed (as opposed to test equipment, etc.), the reason for failure shall be determined and remedial/corrective action taken. Hardware failures shall be documented and dispositioned in accordance with SSP 30223.

When a discrepancy occurs, the test shall be interrupted and a determination shall be made as to whether the discrepancy is because of a failure of the component under test or a failure of the system performing the test (test setup, software, or equipment). If the component under test did not initially fail, it is possible that it could have been overstressed by a failure of the test equipment. After a determination is made that no overstress of the test item exists, the test may be continued after repairs of the system performing the tests are completed. If the test item has failed, either originally or because of overstress, test activities shall resume only after a preliminary failure analysis determines the cause and remedial or corrective action has been taken. Failure analysis, remedial/corrective action(s) taken, and resumption of test activities shall require the approval of the responsible System Problem Review Team.

Final failure analysis may be a continuing function because initial evaluations are sometimes inconclusive and further action may be required, particularly if the failure represents a generic or lot–related problem. For long–term corrective action, a determination shall be made if the failure could have been, and therefore should have been, detected at a lower level of assembly or in an earlier test.

The degree of retest shall be determined on a case–by–case basis and shall take into account the following factors:

A. the results of the failure analysis which may indicate that more thorough functional testing is required

B. the specifics of any manufacturing process or design changes and their potential impact to the component's performance or reliability

C. changes in predicted service environments including transportation and storage

D. the degree of disassembly/reassembly required to achieve rework/redesign objectives

E. hardware criticality and redundancy

Retesting may reduce previously baselined test durations (e.g., reduced number of thermal cycles or only one axis of random vibration versus three, etc.) when deemed reasonable and technically justifiable. Previously baselined test levels (e.g., minimum/maximum test temperatures, vibration amplitudes, etc.) shall not be reduced without prior approval of the ISS Program.

When acceptance random vibration retesting is contemplated, the procedures of 8.2 below shall be followed to ensure that there is adequate remaining life in the component for retest. When retest decisions are dictated by a hardware failure, the cognizant System Problem Review Team shall be responsible for ensuring that the procedures defined below are followed and that there is sufficient remaining life in the item prior to performing acceptance random vibration retesting. Otherwise, the responsible contractor and NASA hardware owners shall be responsible for ensuring the procedures are followed.

8.2 SPECIAL CONSIDERATIONS FOR RANDOM VIBRATION RETESTING

Paragraph 4.1 requires that qualification test levels and duration be sufficient to accommodate acceptance retesting. While 4.1 does not require any specific number of acceptance retests for components, it must be recognized that flight equipment may require additional acceptance random vibration testing (in addition to the initial acceptance random vibration test) throughout its service life. Therefore, a methodology to establish remaining vibration life in components that have undergone multiple acceptance random vibration tests is needed to demonstrate that sufficient useful life remains in the component for retest and subsequent service operations.

Two approaches are provided for assessing remaining random vibration life. Method I shall be used when appropriate response accelerometer data are available from the qualification random vibration test. Caution must be exercised when using this approach by ensuring that the available response accelerometer data are representative of fatigue critical locations in the component. For example, if an electronic component with internal circuit boards has response data from chassis locations only, these data are insufficient for use of Method I. A common “rule–of–thumb” used for design of electronics for vibration environments is to maintain at least one octave separation between any chassis resonance and any circuit board resonance (the so–called “octave rule”). Therefore, use of Method I in this instance would only establish life based on chassis resonance and provide improper life for the critical internal circuit boards.

When appropriate response data are unavailable, Method II shall be used. This approach utilizes the composite root mean square (rms) acceleration input level of the applicable environments for determining remaining life.

If appropriate response data are available for some, but not all, fatigue critical locations in the equipment, both Method I (for locations where response data are available) and Method II shall be used with the more conservative result establishing the life of the equipment.

Regardless of the approach used, particular attention shall be paid to the selection of the fatigue exponent b. The value of this exponent has significant impact on the results of the life calculations and therefore must be selected carefully. Use of a value greater than 4 shall be documented and shall require technical justification based on the equipment’s materials and design. When available data indicate a value less than 4 should be used, then this lower value shall be used.

In general, this policy assumes that service life random vibration environments other than acceptance testing and launch (e.g., transportation, on–orbit, and entry/landing) are significantly lower than the acceptance and launch levels and are therefore negligible. However, all service life environments shall be evaluated, and if a service environment other than the acceptance and launch environments is considered to also contribute to fatigue damage accumulation, then the environment shall be considered in the manner defined herein for flight and included in the calculation for remaining acceptance test time available.

When test environments are defined herein, nominal test levels and duration as documented in the acceptance and qualification test procedures may be used. However, if flight equipment is subjected to random vibration testing in a facility or using test equipment (e.g., shaker tables and/or test fixtures) different from that for which qualification testing was conducted, as–run accelerometer and test time data from all tests shall be used in place of specified nominal test levels and duration. As–run test data shall also be used anytime response limiting has been used during qualification or acceptance testing.

When analysis for retest capability per the methods defined below shows that there is insufficient life remaining, engineering shall recommend an appropriate course of action such as not performing the vibration test, performing the vibration test and accepting the life risk, extending the demonstrated life by performing additional qualification vibration testing on the original qualification test article, or the acquisition of additional spares. This recommendation shall be presented to, and require the approval of, the ISS T&VCP. When additional qualification testing is performed to extend the demonstrated life, an assessment shall be made and documented as to any rework or refurbishment of the qualification test article to ensure that the additional qualification testing will validly extend the demonstrated life of the equipment.

8.2.1 METHOD I

When component response data are available from the qualification random vibration test, the following method shall be used to establish remaining life based on input acceleration spectral density (power spectral density) at the resonant frequency(ies) of the component.

Step 1:

Determine the component resonant response frequency(ies) from the available response data from the qualification test.

For each resonant frequency *i* (based on peak resonant response), determine the input acceleration spectral density for each of the following environments: flight, acceptance, and qualification.

Step 2:

Convert one flight exposure to equivalent time at the acceptance test level (one flight is 30 seconds of exposure) by the following relationship:

$$t_{ae} = t_f \left(\frac{W_{fi}(f_i)}{W_{ai}(f_i)} \right)^{\frac{b}{2}} \qquad \text{where,} \qquad [8.1]$$

t_{ae}	is the equivalent acceptance test time
t_f	is the exposure time in one flight (30 seconds)
$W_{fi}(f_i)$	is the flight input acceleration spectral density at resonant frequency *i*
$W_{ai}(f_i)$	is the input acceleration spectral density during acceptance testing at resonant frequency *i*
b	is a fatigue exponent

Step 3:

Establish the total acceptance test duration (per axis) for which the equipment has been qualified:

$$t_a = \frac{t_q}{4}\left(\frac{W_{qi}(f_i)}{W_{ai}(f_i)}\right)^{\frac{b}{2}} \qquad \text{where,} \qquad [8.2]$$

t_a	is the qualified acceptance random vibration test time in each axis
t_q	is the qualification random vibration test time in each axis
$W_{qi}(f_i)$	is the input acceleration spectral density during qualification testing at resonant frequency *i*
4	is a fatigue scatter factor

Step 4:

Compute the remaining allowable acceptance random vibration test time in each axis:

$$t_{ar} = t_a - [(t_{ae})(F) + t_{au}] \qquad \text{where,} \qquad [8.3]$$

t_{ar}	is the acceptance test time remaining in each axis
t_a	is the qualified total acceptance test time in each axis from equation [8.2]
t_{ae}	is the equivalent acceptance test time in each axis of one flight from equation [8.1]
F	is the number of required flights
t_{au}	is the acceptance test time in each axis already expended

For a component that has more than one critical resonant response frequency, the above procedure shall be followed for each resonance and the result with the lowest remaining acceptance test time shall be used for establishing the remaining life of the component.

Example:

A component has the following data available:

f_1 = 150 Hz	first critical resonant response frequency of the component
f_2 = 560 Hz	second critical resonant response frequency of the component
$W_{f1}(f_1)$ = 0.04 g^2/Hz	flight input acceleration spectral density at 150 Hz
W_{f2} (f_2) = 0.008 g^2/Hz	flight input acceleration spectral density at 560 Hz

$W_{a1}(f_1)$= 0.04 g^2/Hz	acceptance test input acceleration spectral density at 150 Hz
$W_{a2}(f_2)$= 0.025 g^2/Hz	acceptance test input acceleration spectral density at 560 Hz
$W_{q1}(f_1)$= 0.08 g^2/Hz	qualification test input acceleration spectral density at 150 Hz
$W_{q2}(f_2)$= 0.05 g^2/Hz	qualification test input acceleration spectral density at 560 Hz

Acceptance test duration (per axis) is 60 seconds

Qualification test duration (per axis) is 180 seconds

Flight duration (one flight) is 30 seconds

b=4

Number of required flights = 2

For the first resonant frequency (150 Hz):

From Equation [8.1],

$$t_{ae} = 30\left(\frac{0.04}{0.04}\right)^2 \quad = 30 \text{ seconds}$$

Therefore, one flight is equivalent to 30 seconds of acceptance testing.

Next, the total qualified acceptance test time per axis is (from equation [8.2]):

$$t_a = \frac{180}{4}\left(\frac{0.08}{0.04}\right)^2 \quad = 180 \text{ seconds}$$

Assume that the component has already undergone two random vibration acceptance tests; the remaining acceptance test time available from equation [8.3] while maintaining a capability for two launches is:

$$t_{ar} = 180 - [(30)(2) + 120] = 0 \text{ seconds}$$

Therefore, there is no capability for acceptance vibration retesting.

The same method can be used to calculate remaining time for the 560 Hz resonance but it is unnecessary since there is no remaining life based on the 150 Hz resonance.

8.2.2 METHOD II

Step 1:

Convert one flight exposure to equivalent time at the acceptance test level by the following relationship:

$$t_{ae} = t_f \left(\frac{G_f}{G_a} \right)^b \qquad \text{where,} \qquad [8.4]$$

t_{ae}	is the equivalent acceptance test time for one flight
t_f	is the exposure time of one flight (30 seconds)
G_f	is the root mean square (rms) acceleration level (grms) of the maximum predicted flight environment
G_a	is the rms acceleration level (grms) of the acceptance test environment in each axis

Step 2:

Establish the total acceptance test duration (per axis) for which the equipment has been qualified:

$$t_a = \frac{t_q}{4} \left(\frac{G_q}{G_a} \right)^b \qquad \text{where,} \qquad [8.5]$$

t_a	is the qualified time in each axis for acceptance vibration testing
t_q	is the time of the qualification random vibration test in each axis
G_q	is the rms acceleration level (grms) of the qualification random vibration test environment

Step 3:

Compute remaining allowable acceptance random vibration test time in each axis:

$$t_{ar} = t_a - [(t_{ae})(F) + t_{au}] \qquad \text{where,} \qquad [8.6]$$

t_{ar}	is the acceptance test time remaining in each axis
t_a	is the qualified total acceptance test time in each axis from equation [8.5]
t_{ae}	is the total equivalent acceptance test time for all service environments in each axis from equation [8.4]
F	is the number of required flights for the equipment
t_{au}	is the acceptance test time in each axis already expended

Example:

A rack–mounted internal component has the following qualification, acceptance, and flight random vibration environments:

Qualification = 8.6 grms (all axes) for 180 seconds per axis

Acceptance = 6.1 grms (all axes) for 60 seconds per axis

Flight = 4.3 grms (all axes) for 30 seconds per flight; three launches required

$b = 4$

The equivalent acceptance test time for one flight (from equation [8.4]) is:

$$t_{ae} = 30 \left(\frac{4.3}{6.1}\right)^4 \qquad = 7.5 \text{ seconds}$$

Therefore, one flight is equivalent to 7.5 seconds of acceptance testing.

Next, calculate the total qualified acceptance test time from equation [8.5]

$$t_a = \frac{180}{4} \left(\frac{8.6}{6.1}\right)^4 \qquad = 178 \text{ seconds}$$

Assume that the component has undergone two random vibration acceptance tests (all axes); compute the remaining acceptance test time available from equation [8.6] while maintaining the three launch capability:

$$t_{ar} = 178 - [(7.5)(3) + 120] = 35 \text{ seconds}$$

Therefore, another full acceptance vibration test cannot be performed in any axis.

9.0 NONCRITICAL COMPONENT TESTING

This section establishes the program requirements for testing noncritical Space Station hardware. This section does not apply to hardware classified as criticality 1 or 2 as defined by SSP 30234. This section is only applicable to items classified as criticality 3 based on an integrated FMEA assessment from the Reliability and Maintainability Panel or its designee. The special provisions in this section for testing noncritical hardware were established to enable a lower cost and faster development approach, where the lower criticality of the function allows so. This section defines the minimum and mandatory tests and hazard control screens that are required for noncritical hardware with guidelines for additional testing applied on a case by case basis for reliability screening as deemed appropriate for the hardware. Noncritical component qualification and acceptance test requirements and guidelines are defined in Tables 9–1 and 9–2, respectively.

TABLE 9–1 NONCRITICAL HARDWARE QUALIFICATION REQUIREMENTS (PAGE 1 OF 1)

Test	Electronic or Electrical Equipment	Antennas	Moving Mechanical Assembly	Solar Panel	Batteries	Fluid or Propulasion Equipment	Pressure Vessels	Thermal Equipment	Optical Equipment
Functional (1)	R	R	R	R	R	R	R	R	R
Thermal Vacuum (2) 1.5 cycles	R(5)	R	R	R	R	R		R	R
Thermal Cycling 6 Cycles (5)	G				H			G	
Depress/ Repress	G		G			G	G		G
Random Vibration (7)	G H	G	G		G H	G	G	G	G
Acoustic Vibration		G							
Shock	G		G		G H	G		G	G
Pressure					H(6)	H	H		
Leak (6)	H(6)				H(6)	H	H	H	H
EMC	R(3) H(4) G(4)								

Legend:
R = Required
H = Hazard Screen
G = Reliability Screen

Note: Test listed as guidelines (G) do not alleviate the responsibility to verify the criticality 3 equipment will meet the requirements for its intended use on ISS.

(1) Functional tests shall be conducted prior to and following each environmental test.
(2) For external hardware only. In performing the 1.5 cycles, there shall be two maximum temperature dwells and one minimum temperature dwell. A 12 hour dwell is required on first cycle.
(3) Emissions only.
(4) Susceptibility only.
(5) Minimum 140 degrees F sweep required.
(6) For sealed or pressurized equipment.
(7) Vibration testing shall be in accordance with 4.2.5.

TABLE 9–2 NONCRITICAL HARDWARE ACCEPTANCE REQUIREMENTS (PAGE 1 OF 1)

Test	Electronic or Electrical Equipment	Antennas	Moving Mechanical Assembly	Solar Panel	Batteries	Fluid or Propulsion Equipment	Pressure Vessels	Thermal Equipment	Optical Equipment
Functional (1)	R	R	R	R	R	R		R	R
Thermal Vacuum (2)(4)(6) 1 cycle	R		R		R	R		R	R
Thermal Cycling 3 Cycles (4)	G					G			
Random Vibration	G	G	G			G		G	G
Acoustic Vibration		G	G	G					
Pressure					G	G	G		
Leak	G(7)				G(7)	G	G	G	G
Burn–in	G(3)								

Legend:
R = Required
H = Hazard Screen
G = Reliability Screen

Notes:
(1) Functional tests shall be conducted prior to and following each environmental test.
(2) For external hardware only.
(3) Burn–in shall be 300 hours ambient or equivalent as specified in section 5.
(4) Minimum 100 degrees F sweep required.
(5) Vibration testing shall be in accordance with 5.1.4.
(6) Thermal cycling can be used in lieu of thermal vacuum if qualification test or analysis shows no vacuum sensitive components or materials are present.
(7) For sealed or pressurized equipment.

9.1 DESCRIPTION

Testing requirements for noncritical hardware are established on a case–by–case basis for each deliverable component by the provider and the ISS Program Office of Primary Responsibility (OPR) for the component. Tailoring of test requirements for specific noncritical components shall be established when specifications for design are defined, and are modified as necessary during design review activities. Tailored requirements shall be developed by including required tests specified in 9.1.1 and 9.1.2 and adding tests from the guidelines specified in 9.1.3 that are appropriate for the component consistent with its intended life, cost, complexity, and maintenance strategy. Test levels and procedures shall be the same as section 4 for component qualification tests and section 5 for component acceptance tests with reduced cycles and duration as indicated in Tables 9–1 and 9–2.

The requirements for noncritical hardware in this section shall not be applied retroactively for items currently requiring testing to the requirements proscribed elsewhere in this document without the consent of the NASA ISS Program OPR.

9.1.1 REQUIRED TESTS

Test identified with a (R) in Tables 9–1 and 9–2 shall be the minimum and mandatory test requirements for noncritical hardware and shall be relieved only by an ISS Program approved exception, deviation, or waiver as appropriate.

9.1.2 HAZARD SCREENING

Tests identified as hazard screens (H) in Tables 9–1 and 9–2 shall be required except when an approved integrated hazard report indicates that adequate controls are in place to mitigate the hazard in lieu of the test. The Safety Review Panel shall be the designated approving authority for integrated hazard reports.

9.1.3 RELIABILITY SCREENING GUIDELINES

Tests from these guidelines are added as needed to the required minimum tests and hazard control screens to develop a tailored test requirement suite for qualification and acceptance of a specific noncritical hardware component by agreement between the hardware provider and the NASA ISS Program OPR for the item. The following are examples of factors which should be considered to determine whether tests should be added to the suite from the guidelines to strengthen the overall reliability screening of the component:

A. Rationale for minimizing addition of reliability testing:

(1) Commercial off the shelf hardware with a proven track record;

(2) Proven in–service history in a similar environment;

(3) Readily available spares and ease of maintenance by crew;

(4) High volume, vendor screened items.

B. Rationale for increasing reliability testing:

(1) Relative size of hardware investment;

(2) Operational impact associated with noncritical hardware failures;

(3) Hardware with an analytically derived (calculated) Mean Time Between Failure.

The full suite of tests identified as reliability screening guidelines (G) in Tables 9–1 and 9–2 are the recommended suite of tests for complex, expensive hardware with failure impact that is difficult to maintain or replace.

10.0 NOTES

The listing in this glossary amplifies selected terms used in this document. The meanings reflect the particular use of these terms in this document.

10.1 GOVERNMENT–FURNISHED PROPERTY

The contracting officer should arrange to furnish the property listed in accordance with the contract.

10.2 DEFINITIONS

ACCEPTANCE. A process which demonstrates that an item was manufactured as designed with adequate workmanship, performs in accordance with specification requirements, and is acceptable for delivery.

ACCEPTANCE LIMITS. The maximum predicted levels documented in an element, subsystem, or component specification or as otherwise specified herein.

AMBIENT ENVIRONMENT. The ambient environment for a ground test is defined as normal room conditions (i.e., a temperature of 78 +/– 18 degrees F; atmospheric pressure of 101 +2/–23 kilo–pascals (29.9 + 0.6/–6.8 inches Hg); and a relative humidity of 50 +/– 30 percent).

ANALYSIS. A verification method utilizing techniques and tools such as math models, compilation, similarity assessments, and validation of records to confirm that requirements to be verified have been satisfied.

ANTENNAS. Mechanical assemblies utilized to radiate or collect Radio Frequency (RF) energy to or from the surrounding environment.

ASSEMBLY. The process of joining together two or more parts/components; placing in orbit and arranging in orbital configuration the various elements of the International Space Station (multi unit spacecraft); placing on the ISS and assembling in flight configuration the various elements of a large spacecraft; multiple software components that function together; the third level of software structure, which corresponds to the level of multiple Orbital Replaceable Units; and a hardware structure containing two or more components.

BACKGROUND. In leak testing, the steady or fluctuating output of the leak detector caused by the presence of residual tracer gas or other substance to which the detecting element responds.

BATTERIES. Electrochemical energy storage devices used to store electrical power for use when solar array output is not sufficient to support station needs.

COMPONENT. A component is an assembly of parts that constitute a functional article viewed as an entity for purposes of analysis, manufacturing, maintenance, or record keeping; the smallest entity specified for a distributed system. Examples are hydraulic actuators, valves, batteries, electrical harnesses, individual electronic assemblies, and Orbital Replaceable Units.

DETECTOR PROBE. In leak testing, a device used to collect tracer gas from an area of the test component and feed it to the leak detector at the reduced pressure required. Also called a sniffing probe.

DEVIATION. A specific authorization, granted before the fact, to depart from a particular baseline requirement for a limited application.

DRIFT. In leak testing, the relatively slow change in the background output level of the leak detector due to the electronics rather than a change in the level of the tracer gas.

ELECTROMAGNETIC COMPATIBILITY. The condition that prevails when various electronic devices are performing their functions according to design in a common electromagnetic environment.

ELECTROMAGNETIC INTERFERENCE. Electromagnetic energy that interrupts, obstructs, or otherwise degrades or limits the effective performance of electrical equipment.

ELECTRONIC OR ELECTRICAL EQUIPMENT. Electrical power equipment for conversion, distribution, switching, or electrical power consuming equipment as defined in SSP 30482.

ENVIRONMENTS. The International Space Station environments include design environmental conditions experienced during the life cycle of the components, assemblies, and supporting elements. The life cycle includes manufacturing, test, storage, transportation, launch, orbital operation, and landing. The environments include vibration, shock, acoustic noise, acoustic vibration, toxic offgassing, acceleration, electromagnetic, electrostatic, temperature, humidity, reduced atmosphere, pressure, radiation, orbital density and composition, orbital debris, meteoroids, magnetic and gravitational fields, plasma, and contamination.

ENVIRONMENTAL DESIGN MARGINS. An environmental design margin for an item is an increase in the environmental range used for the design (and for the qualification testing) of an item to reduce the risk of an operational failure. It may include increases in the maximum predicted levels, decreases in the minimum predicted levels, and increases in the time exposure to the extreme predicted levels. The environmental design margin is intended to perform the following:

A. To accommodate differences among qualification and flight units due to variations in parts, materials, processes, manufacturing, testing, and degradation during usage;

B. To incorporate the allowable test conditions tolerances;

C. To avoid qualification test levels that are less severe than the acceptance test ranges or operating ranges;

D. To help prevent fatigue failures because of repeated testing and operational use.

EXTERNAL–TO–INTERNAL TOTAL LEAKAGE. The combined leakage rate of a tracer gas through all the existing leaks from outside to inside of a component being tested.

FLIGHT ELEMENT. The ISS is composed of flight elements as defined in Section 3.7 of the On–Orbit Segment specifications, SSP 41160 thru SSP 41167. Element as used in this document means flight element.

FLIGHT HARDWARE. All identifiable ISS equipment, including assembly elements, flight elements, ORUs, and distributed systems, that will undergo acceptance or protoflight testing and be certified for flight.

FLIGHT SOFTWARE. Flight software is the body of operational software on the ISS at a given point in time. For qualification, acceptance, or protoflight tests of hardware using the flight software, the software may be modified to the extent necessary to conduct the tests.

FLUID OR PROPULSION EQUIPMENT. Hydromechanical equipment used to control, regulate, dispense, distribute, or expel fluids or propellants.

FUNCTIONALITY. Proof that an item functions, as specified, by sampling the operating envelope while in the specified environment.

INTERNAL–TO–EXTERNAL TOTAL LEAKAGE. The combined leakage rate of a tracer gas through all the existing leaks from inside of a component being tested to outside.

INTERNAL–TO–INTERNAL TOTAL LEAKAGE. The combined leakage rate of a tracer gas through all the existing leaks in a component internal barrier.

LEAK. A hole or void in the wall of an enclosure capable of passing liquid or gas from one side of the wall to the other under action of pressure or concentration differential existing across the wall, independent of the quantity of fluid flowing.

LEAK DETECTOR. A device for detecting, locating, and/or measuring leakage.

LEAKAGE RATE. The flow rate of a liquid or gas through a leak at a given temperature as a result of a specified pressure difference across the leak. Standard conditions for gases are 77 degrees F (25 degrees C) and 100 kPa. Leakage rates are expressed in various units such as pascal cubic meters per second or standard cubic centimeters per second.

LIFE–CYCLE ENVIRONMENTS. The full set of environments experienced by an article both operating and nonoperating from factory acceptance to disposal.

LIMIT LOAD. The maximum load expected on the structure during its service life, including fabrication, ground handling and transportation, transport to and from orbit including abort conditions, and on–orbit operations. As used in this document for structural test requirements, it shall be construed as the limit load for which the structure must be certified.

MAXIMUM DESIGN PRESSURE. The MDP for a pressurized system or component is the highest pressure defined by the maximum relief pressure, maximum regulator pressure, maximum temperature, and transient pressure excursions.

MAXIMUM PREDICTED LEVEL. The maximum expected level of an environmental parameter (temperature, pressure, vibration, etc.) which a component, assembly, or flight element will be subjected during its service life and at which the component is required to operate as documented in the component specification.

MINIMUM PREDICTED LEVEL. The minimum expected level of an environmental parameter (temperature, pressure, vibration, etc.) which a component, assembly, or flight element will be subjected during its service life and at which the component is required to operate as documented in the component specification.

MOVING MECHANICAL ASSEMBLIES. A mechanical or electromechanical device that controls the movement of one mechanical part of a vehicle relative to another part. Examples are gimbals, actuators, separation mechanisms, capture mechanisms, valves, pumps, motors, latches, clutches, springs, damper bearings, and instrumentation that are an integral part of the mechanical assembly.

MULTIPACTING. The resonant back and forth flow of secondary electrons in a vacuum between two surfaces separated by a distance such that the electron transit time is an odd integral multiple of one half the period of the alternating voltage impressed on the surfaces. Multipacting requires an electron impacting one surface to initiate the action and requires the secondary emission of one or more electrons at each surface to sustain the action.

NONOPERATING TEMPERATURES. The maximum and minimum temperatures to which the component may be exposed in a nonoperational state as documented in the component specification. The component is required to meet all specification requirements at operational environmental extremes after exposure to the required nonoperational environments.

NOMINAL TEST VALUE. This value is the expected or planned for value within the acceptance tolerance band.

OPERATIONAL MODES. All combinations of operational configurations or conditions that can occur during the service life.

OPTICAL EQUIPMENT. Sensors or devices requiring or utilizing any portion of the light or energy spectrum.

ORBITAL REPLACEABLE UNIT. The designated level of system hardware that is permitted to be removed and replaced on location under orbital conditions.

PART. A part is a single piece, or two or more pieces joined together, that are not normally subject to disassembly without destruction or impairment of the design use.

PASSIVE THERMAL EQUIPMENT. Components in the thermal system that have no moving parts, no electrical components, or thermal sensors.

PRESSURE VESSEL. A container designed primarily for pressurized storage of gases or liquids and:

A. Contains stored energy of 14,240 foot–pounds (19,307 joules) or greater based on adiabatic expansion of a perfect gas;

B. Contains a gas or liquid in excess of 15 psia (103.4 kPa) which will create a hazard if released; or

C. Stores a gas which will experience an MDP greater than 100 psi (689.5 kPa).

PRESSURIZED COMPONENT. A component designed to retain its leak tightness at both standard atmospheric and positive differential internal pressure.

PROOF PRESSURE. The pressure to which pressurized components, assemblies, or elements are subjected to fulfill the acceptance requirements to give evidence of satisfactory workmanship and materials quality. Proof pressure is the product of the maximum operating pressure and the appropriate proof factor of safety required for screening maximum allowable flaws based on fracture mechanics analysis.

PROTOFLIGHT. A test program intended to combine the objectives of the qualification and acceptance test programs; i.e. design confidence for use in the service environments and adequate workmanship/quality. All protoflight components, assemblies, and flight elements are intended for subsequent flight use. The protoflight approach uses reduced test levels, cycles, and/or duration from the standard qualification test requirements, to allow the protoflight tested hardware to be used for flight. A Protoflight approach is a high technical risk approach compared to a full qualification test program due to there being no demonstrated flight duration capability (i.e., number of cycles; or time of operation or exposure to a service environment) and, in some cases, lower demonstrated margins over the service environment extremes.

PROTOFLIGHT HARDWARE. Flight hardware utilized for qualification testing in lieu of a dedicated test article. The approach includes the use of reduced test levels and/or durations and post–test hardware refurbishment, where required, to allow tested hardware to be used subsequently for flight.

QUALIFICATION. Qualification is the process that proves the design, manufacturing, and assembly of the hardware and software complies with the design requirements when subjected to environmental conditions.

QUALIFICATION TEST ARTICLE. A qualification test article is a flight article modified to the extent necessary to conduct the qualification test.

SEALED COMPONENT. A component designed to retain its leak tightness at both standard atmospheric and negative differential internal pressure.

SENSITIVITY OF LEAK TEST. The smallest leakage rate that an instrument, method, or system is capable of detecting under specified conditions.

SERVICE LIFE. The service life is the total life expectancy of the item. The service life starts at the completion of assembly of the item and continues through all acceptance testing, handling, storage, transportation, launch operations, orbital operations, refurbishment, retesting, reentry or recovery from orbit, and reuse that may be required or specified for the item.

SOLAR PANEL. A collection of Photovoltaic cells mounted on structural material and used to convert solar energy to electrical power.

SPECIFICATION. Statement of particulars such as performance, characteristics, requirements, and configuration for a given element of hardware/software.

STANDARD LEAK. A device that permits a tracer gas to be introduced into a leak detector or leak testing system at a known rate to facilitate calibration of the leak detector.

SUBASSEMBLY. Two or more components joined together as a unit package capable of disassembly and component replacement. A subassembly may or may not be an ORU but will have verification requirements.

SUBSYSTEM. A specific set of hardware and/or software functional entities and their associated interconnections that perform a single category of functions (e.g., data storage and retrieval subsystem, video subsystem). The functional level immediately below the system level.

SUITABLE LEAK TEST METHODS. A suitable leak test method shall exhibit, as a minimum, the following:

Calibration – A suitable leak test method shall be defined as one that establishes a calibration of the leak test setup such that the calibration method is commensurate with the allowable leakage rate to be detected. For leak test standard tools such as graduated flasks, columns, and pipettes purchased at standard scientific suppliers, the calibration of the graduations shall be accepted. Tracer gas leak standards shall bear a calibration certification sticker from metrology or the vendor and shall be within the prescribed dates and, if equipped with a pressure gage, within the appropriate pressure range.

Characterization – A suitable leak test method shall demonstrate the appropriate time duration to establish confidence in the ability to detect leakage rates above background and establish a stable time period to allow for permeation, multiple leak paths, etc. The time duration established to calibrate the leak test setup to demonstrate the ability to detect leakage will be accepted as the (one) time constant. To demonstrate leakage rate stability during the performance of the actual leak test, the leak test setup shall require continuous monitoring until the measured leakage rate result exhibits less than 10 percent variation for a duration of three time constants.

Documentation – Appropriate documentation shall include as a minimum: the operator, the inspector, the method of calibration and sensitivity of the leak test setup, a detailed sketch/ description of the leak test setup, number of test data points measured, and the respective time intervals of the measurements. The test report and procedures shall also include information on any training and/or certification of the inspectors and technicians.

TEMPERATURE CYCLE. A transition from some initial temperature condition to temperature stabilization at one extreme and then to temperature stabilization at the opposite extreme and returning to the initial temperature condition.

TEST. A method of verification wherein requirements are verified by measurement during or after the controlled application of functional and environmental stimuli. These measurements may require the use of laboratory equipment, recorded data, procedures, test support items, or services.

ACCEPTANCE TESTS are those formal tests conducted to assure that the component or end item meets specified requirements and is of adequate quality. Acceptance tests include performance demonstrations and environmental exposures to screen out manufacturing defects, workmanship errors, incipient failures, and other performance anomalies not readily detectable by normal inspection techniques or through ambient functional tests.

PROOF TESTS are generally conducted on pressure vessels, miscellaneous structural components, structural assemblies, or mechanisms to ensure confidence in the manufactured article.

PROTOFLIGHT TESTS are tests intended to demonstrate both design confidence for use in the service environments as well as adequate workmanship/quality of the component or end item. Protoflight testing does not demonstrate any additional capability of the item to perform during service usage, and is, therefore, a high technical risk approach.

QUALIFICATION TESTS are tests conducted as part of the verification program to demonstrate that the design and performance requirements can be realized under specified conditions.

TEST CONDITION TOLERANCES. The test condition tolerances allowed by this document are applied to the nominal test values specified.

TEST DISCREPANCY. A test discrepancy is a functional or structural anomaly which occurs during testing and which indicates a possible deviation from specification requirements for the test item. A test discrepancy may be a momentary, nonrepeatable, or permanent failure to respond in the predicted manner to a specified combination of test environment and functional test stimuli. Test discrepancies may be because of a failure of the test unit or of some other cause such as the test setup, test instrumentation, supplied power, the test procedures, or to the computer software used.

THERMAL EQUILIBRIUM. Thermal equilibrium is achieved when the component internal part with the largest temperature constant is with 5.4 degrees F (3 degrees C) of its equilibrium temperature, as determined by extrapolation of test temperatures and/or previous analysis/test data, and its rate of change is less than 5.4 degrees F per hour (3 degrees C per hour).

THERMAL EQUIPMENT. Mechanical or hydromechanical equipment used to collect or dispose of thermal energy.

THERMAL STABILITY. Thermal stability is achieved when the rate of change of the control temperature, as determined by the control temperature sensor, is no more than 5.4 degrees F per hour (3 degrees C per hour).

TRACER GAS. A gas which, after passing through a leak, can be detected by a specific leak detector and thus disclose the presence of a leak.

ULTIMATE LOAD. The maximum load is the product of the limit load and the ultimate factor of safety. The structure shall not rupture or collapse at any load less than or equal to the ultimate load. As used in this document for structural test requirements, it shall be construed as the ultimate load for which the structure must be certified.

ULTIMATE PRESSURE. The pressure to which pressure vessels are subjected to fulfill the qualification requirements to give evidence of satisfactory design quality. Ultimate pressure is the product of the maximum pressure and the appropriate ultimate factor of safety.

WAIVER. A written authorization to accept designated items which, during production or after having been submitted for inspection, are found to depart from specified requirements, but nevertheless are considered suitable for use ”as is”.

10.3 ABBREVIATIONS AND ACRONYMS

AAA	Avionics Air Assembly
ACBM	Active Common Berthing Mechanism
ACBSP	Assembly Contingency Baseband Signal Processor
ACRFG	Assembly Contingency RF Group
ACS	Atmosphere Conditioning System
ALQT	Assembly Level Qualification Test
APCU	Assembly Power Converter Unit
AR	Atmosphere Revitalization
AT	Acceptance Test
ATCS	Active Thermal Control System
ATP	Acceptance Test Procedure
AVT	Acceptance Vibration Test
BBA	Baseplate Ballast Assembly
BCA	Battery Charger Assembly
BCDU	Battery Charge/Discharge Unit
BGA	Beta Gimbal Assembly
BMRRM	Bearing/Motor Roller Ring Module
BSCCM	Battery Signal Conditioner and Control Module
C	Celsius
CAT	computer aided test
CBM	Common Berthing Mechanism
CCA	Circuit Card Assemblies
CCAA	Common Cabin Air Assembly
CDRA	Carbon Dioxide Removal Assembly
CETA	Crew and Equipment Translation Aids
CEU	Control Electronics Unit
CLA	Capture Latch Assembly
cm	centimeters
CMG	Control Moment Gyro
CTP	Command and Telemetry Processor
COTS	Commercial Off–the–Shelf
CSA–CP	Compound Specific Analyzer-Combustion Products

dB	Decibels
dc	direct current
DDCU	DC–to–DC Converter Unit
DDP	Design Decision Package
DLA	Drive Lock Assembly
EA	Electronic Assembly
ECU	Electronic Control Unit
EDDA	Extravehicular Mobility Unit Don/Doff Assembly
EEE	Electrical, Electronic, and Electromechanical
EL	elevation
EM	Engineering Model
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EPCS	Early Portable Computer System
EVA	Extravehicular Activity
F	Fahrenheit
FC	firmware controller
FDIR	Failure Detection, Isolation, and Recovery
FHRC	Flexible Hose Rotary Coupler
FLAP	Fluid Line Anchor Patch
FMEA	Failure Modes and Effects Analysis
FQDC	Fluid Quick Disconnect Coupling
FRACA	Failure Reporting and Corrective Action
g	gravity
GFE	Government Furnished Equipment
GHe	Gaseous Helium
GOx	Gaseous Oxygen
grms	gravity root mean square
Hab	Habitation
HRFM	High Rate Frame Multiplexer
HRM	High Rate Modulator
Hg	Mercury
HRS	Heat Rejection System
Hz	Hertz
IEA	Integrated Equipment Assembly
IMCA	Integrated Motor/Controller Assembly
IMV	Intermodule Ventilation
ISA	Internal Sampling Adapter
ISS	International Space Station
ITCS	Internal Thermal Control System
IUA	
IVA	Intravehicular Activity
JSC	Johnson Space Center
K	Kelvin
kPa	kilo Pascal

lbm	pounds mass
LDU	Linear Drive Unit
LED	Light Emitting Diode
LHA	Lamp Housing Assembly
LMMS	Lockheed–Martin Missiles and Space
LMSC	Lockheed Missiles and Space Corporation
LNA	Low Noise Amplifier
LTU	Load Transfer Unit
LVS	
MBA	Motorized Bolt Assembly
MDA	Motor Drive Assembly
MDM	Multiplexer/Demultiplexer
MDP	Maximum Design Pressure
ml	milliliter
MOP	Maximum Operating Pressure
MPEV	Manual Pressure Equalization Valve
MRK	Moisture Removal Kit
MT	Mobile Transporter
MTBF	Mean Time Between Failure
MUA	Material Usage Agreement
N2	Nitrogen
NH3	Ammonia
NIA	Nitrogen Interface Assembly
NIV	Nitrogen Isolation Valve
NPRV	Negative Pressure Relief Valve
NPV	Non Propulsive Vent
OASPL	Overall Sound Pressure Level
OPR	Office of Primary Responsibility
ORU	Orbital Replaceable Unit
OWV	Overboard Water Vent
Pa	Pascals
PAS	Passive Attachment System
PCBM	Passive Common Berthing Mechanism
PCS	Portable Computer System
PCU	Plasma Contactor Unit
PCVP	Pump and Control Valve Package
PDA	Preliminary Design Audit
PDGF	Power Data Grapple Fixture
PDTA	Power/Data Transfer Assembly
PFCS	Pump Flow Control System
PGSC	Payload Ground Support Computer
PIA	Preinstalled Assembly
PIT	Pre–integrated Truss
PG	Product Group
PMA	Pressurized Mating Adapter
psi	pounds per square inch
psia	pounds per square inch absolute
psid	pounds per square inch differential
psig	pounds per square inch gauge

PVM	Photovoltaic Module
PVR	Photovoltaic Radiator
QAVT	Qualification for Acceptance Test
QD	Quick Disconnect
OIV	Oxygen Isolation Valve
QTP	Qualification Test Procedure
RBVM	Radiator Beam Valve Module
RF	Radio Frequency
RFCA	Rack Flow Control Assembly
RFPDB	Radio Frequency Power Distribution Box
RGA	Rate Gyro Assembly
RJMC	Rotary Joint Motor Controller
RMO	Remote Manual Operator
RPCM	Remote Power Control Module
RSP	Respiratory Support Pack
rss	root sum square
RSU	Roller Suspension Unit (appendix A)
RSU	Remote Sensor Unit (appendix D)
RTAS	Rocketdyne Truss Attachment System
RTD	Resistance Temperature Detector
SARJ	Solar Alpha Rotary Joint
SAW	Solar Array Wing
sccs	standard cubic centimeters per second
SDMS	Structural Dynamic Measurement System
SDS	Sample Distribution System
SFU	Squib Fire Unit
SGANT	Space–to–Ground Antenna
SGTRC	Space–to–Ground Transmitter/Receiver Controller
SPCE	Servicing and Performance Checkout Equipment
SPDA	Secondary Power Distribution Assembly
SPF	Space Power Facility
SRCA	System Remote On/Off Control Assembl
SSBA	Space Station Buffer Amplifiers
SSFP	Space Station Freedom Program
SSPA	Solid State Power Amplifier
STA	Structural Test Article
T&VCP	Test and Verification Control Panel
T/C	Thermal Cycling
TCCS	Trace Contaminant Control Subassembly
TCCV	Temperature Control Check Valve
TDRSS	Tracking and Data Relay Satellite System
TDM	Tracking Modulator Driver
Torr	unit of measurement
TRRJ	Thermal Radiator Rotary Joint
TUS	Trailing Umbilical System
T/V	Thermal Vacuum
TVCIC	TV Camera Interface Converter
UHF	ultra high frequency
UMA	Umbilical Mechanism Assembly

UOP	Utility Outlet Panel
USL	United States Laboratory
UTA	Utility Transfer Assembly
UUT	Unit Under Test
V	Volt
VBSP	Video Baseband Signal Processor
VSW	External Video Switch
VTR	Videotape Recorder
WIS	Wireless Information System
XEL	cross elevation
XPNDR	Transponder

APPENDIX A PG–1 APPROVED EXCEPTIONS

The following is a list of exceptions to this document taken by Product Group (PG) 1. The exceptions to this document in no way eliminates the Contractor's responsibility for showing compliance to the sections 3.2 through 3.7 of the applicable specification.

PG1–01:

Mobile Transporter

SSP 41172 REQUIREMENT:

Random Vibration or Acoustic test required for qualification of components or flight elements. See Tables 4–1 and 4–2.

EXCEPTION:

No vibration or acoustic testing is planned for the protoflight Mobile Transporter (MT) assembly. This test was deleted during cost convergence.

RATIONALE:

This is a high risk approach that has been considered with, and accepted by Boeing during cost convergence. Some MT components are vibration tested at the component level. Other components (end stop unit, wire harness) are not qualified at component level. Current approach is to enhance MT Structural Test Article (STA) and confirm Orbital Replaceable Unit (ORU) levels during a Segment S0/MT acoustic test.

PG1–03:

Mobile Transporter components

SSP 41172 REQUIREMENT:

Thermal vacuum test required for qualification and acceptance of mechanical components per Table 5–1.

EXCEPTION:

No thermal vacuum acceptance testing is planned for MT components (Linear Drive Unit (LDU), Load Transfer Unit (LTU), and RSU), only thermal cycle.

RATIONALE:

The moving mechanical assembly MT components are subjected to thermal vacuum qualification tests at the component level (LDU, Roller Suspension Unit (RSU), LTU) to qualify the design. The Integrated Motor/Controller Assemblies (IMCA) are thermal vacuum acceptance tested to check for workmanship at the supplier prior to their assembly into the MT ORUs. The mechanical hardware associated with the ORUs do not require thermal vacuum to detect workmanship, only inspection, vibration, and thermal cycle tests.

PG1–04:

Mobile Transporter

SSP 41172 REQUIREMENT:

EMI/EMC test required for component and flight element qualification. See Tables 4–1 and 4–2.

EXCEPTION:

No planned EMI/EMC tests performed by Astro at component level or MT assembly level.

RATIONALE:

MT ORUs use IMCAs as the electromechanical device. These items have already been qualified prior to their delivery to Astro. The MT assembly will be part of the MT/Trailing Umbilical System (TUS)/S0 EMC test instead of a stand alone EMC test.

PG1–05:

Thermal Radiator Rotary Joint (TRRJ) Assembly

SSP 41172 REQUIREMENT:

Paragraph 6.1, Use of qualification assemblies for flight (protoflight).

Second sentence ”Subsequent assemblies to the first assembly subjected to protoflight tests shall be subjected to identical protoflight tests.”

EXCEPTION:

Acoustic and shock testing will be performed for the second TRRJ (protoflight unit for P1) at the segment level, during the S1 STA vibro–acoustic testing. Acoustic and shock testing for the first TRRJ (protoflight unit for S1) will be performed while mounted in a test fixture. The level and duration will be similar to those achieved during the S1 STA testing. Therefore, the two TRRJ units will not be subjected to identical testing.

RATIONALE:

In order to cut cost, the TRRJ went to a protoflight program. Additional cost cuts deleted the TRRJ S1 STA simulator. Therefore, due to cost and schedule issues the first and second TRRJ units will not be acoustic and shock tested in the identical manner.

PG1–06:

Bearing Assembly (part of TRRJ, S1/P1)

SSP 41172 REQUIREMENT:

Thermal Vacuum test required for qualification and acceptance of components of ”Moving Mechanical Assemblies”. See Tables 4–1 and 5–1.

EXCEPTION:

No Thermal Vacuum testing is planned for the protoflight TRRJ Bearing Assembly. This test was deleted during cost convergence.

RATIONALE:

The thermal vacuum test of the bearing assembly will take place at the TRRJ assembly level.

PG1–07:

This exception has been superceded by PG1–260.

PG1–08:

Power/Data Transfer Assembly (PDTA); part of TRRJ, S1/P1

SSP 41172 REQUIREMENT:

EMI/EMC test required for component qualification. Testing will be in accordance with test requirements of SSP 30237. See SSP 30237, Table 3.2–1, column 1C.

EXCEPTION:

No test cases CS01 and CS02 as defined in SSP 30237 will be performed.

RATIONALE:

The PDTA will be tested at the S1 Segment level.

PG1–10:

Rotary Joint Motor Controller (RJMC); part of TRRJ, S1/P1

SSP 41172 REQUIREMENT:

Thermal Vacuum test required for qualification and acceptance of components of ”Electronic or electrical equipment”. See Tables 4–1 and 5–1.

EXCEPTION:

Thermal vacuum testing for the RJMC quality and acceptance unit have been deleted during cost convergence.

RATIONALE:

Deleted during cost convergence. Thermal cycle testing will be sufficient.

PG1–11:

TRRJ Assembly

SSP 41172 REQUIREMENT:

EMI/EMC test required for component qualification. Testing will be in accordance with test requirements of SSP 30237. See SSP 30237, Table 3.2–1, column 1C.

EXCEPTION:

No EMI/EMC testing will be performed at the TRRJ assembly level.

RATIONALE:

EMI/EMC testing will be done on the RJMC and the Drive Lock Assembly (DLA) and not on the TRRJ assembly. Since these two items are the main electronic packages on the TRRJ assembly, some EMC testing will also be performed at the S1 ITA level.

PG1–12:

Radiator Beam Launch Locks and On–orbit restraint

SSP 41172 REQUIREMENT:

Thermal Vacuum test required for qualification of moving mechanical assemblies. See Table 4–1.

EXCEPTION:

Thermal vacuum test are not performed on the qualification Radiator Beam Launch Locks and On–orbit Radiator Beam restraint.

RATIONALE:

Thermal tests only (not thermal/vacuum) will be performed on the Radiator Beam Launch Locks and On–orbit Restraints to certify the designs for operation at temperature extremes. To perform these tests under vacuum conditions would result in a substantial increase in the cost and complexity of the tests and was deemed unnecessary since the function and performance of the assemblies does not involve vacuum sensitive components.

PG1–13:

S1/P1 Structural Test Article Part Number 1T9000–501

SSP 41172 REQUIREMENT:

Paragraph 4.4.3, Acoustic Vibration Test, Component Qualification.
Paragraph 4.4.3.3, Test Levels And Duration. Exposure test time shall be at least three times the expected flight exposure time to the maximum flight environment or three times the acceptance test duration if that is greater but not less than three minutes.

EXCEPTION:

The S1/P1 STA acoustic test article will only be tested for a total duration of 60 seconds (one minute).

RATIONALE:

The S1/P1 STA acoustic test article consisted of simulated (mass, dynamic, or acoustic) ORUs/components, three flight HRS radiators, and a flight TRRJ. The S1/P1 qualification acoustic vibration test serves as the acceptance acoustic vibration test of the flight HRS radiators, a protoflight acoustic vibration test of the TRRJ, a qualification test of selected fluid lines and wire harness clamps, and an acoustic structural test article for validation of other ORU/component qualification random vibration test environments. Discussions between NASA/JSC and Boeing Huntington Beach personnel resulted in reducing the required test duration to one minute. The three–minute minimum test duration specified in SSP 41172 presumes that a one–minute acoustic vibration acceptance test is performed on the flight element. There is no acoustic vibration acceptance test of the S1 or P1 flight element; therefore, the three–minute requirement is not applicable. An one–minute test duration is sufficient to gather data to validate ORU/component qualification random vibration environments and is the required test duration for acceptance testing of the HRS radiators and protoflight testing of the TRRJ. One minute qualifies the fluid lines and wire harness clamps for two launches, and the UHF antenna deployment mechanism is now manifested to launch on the Spacelab pallet and is qualified by a component random vibration test. Therefore, a one–minute test is sufficient to achieve all of the S1/P1 acoustic vibration test objectives and does not result in unnecessary life expenditure of the flight hardware present on the test article.

PG1–14:

LVS Radiator Assembly ORU shock qualification test

SSP 41172 REQUIREMENT:

Paragraph 4.2.7.3, Test Levels and Exposure. The shock spectrum in each direction along each of the three orthogonal axes shall be at least the maximum predicted level plus 6 dB for that direction.

EXCEPTION:

The LVS Radiator ORU qualification may not reach the required 6 dB margin.

RATIONALE:

The plan to qualify the radiator ORU for the shock environment it produces is to increase the preload of the cinch mechanisms to as high as strength will allow without yielding to try and obtain margin since the ORU weighs over 2200 lbm. The amount of margin realized will not be known until the S1 segment test is performed where the radiator release mechanisms will be activated using nominal preloads.

PG1–15:

TRRJ Assembly and Fluid Lines

SSP 41172 REQUIREMENT:

Paragraph 6.1.1D, Assembly/Component Protoflight Tests.

For the pyrotechnic shock test, the shock spectrum shall be 3 dB greater than the maximum predicted level.

EXCEPTION:

The radiator ORU release mechanisms will be activated using nominal preloads which will not produce margin for the TRRJ or any other component.

RATIONALE:

The defined TRRJ protoflight test program limits the shock input to flight levels. The shock is generated directly by the flight radiator ORUs. The risk associated with not demonstrating margin, relative to the shock environment, was accepted as part of the cost convergence effort which deleted TRRJ assembly level shock tests which would have been performed at higher levels.

PG1–16:

Solar Alpha Rotary Joint (SARJ) Assemblies (Protoflight Units)

SSP 41172 REQUIREMENT:

Thermal Cycling Qualification Tests of Electrical and Electronic equipment. See Table 4–1.

EXCEPTION:

Thermal Cycle Qualification testing will not be performed on the SARJ Assemblies.

RATIONALE:

All Electrical and Electronic ORUs mounted on the SARJ Assembly will go through Thermal Cycle Qualification testing at the ORU level. The only Electrical and Electronic equipment mounted on the SARJ Assembly which will not see Thermal Cycle Qualification testing will be the wire harnesses. McDonnell Douglas Aerospace has determined that a Thermal Cycle Qualification test of the SARJ Assembly is not warranted to test only wire harnesses.

PG1–17:

SARJ Assemblies (Protoflight Units)

SSP 41172 REQUIREMENT:

Acoustic/Random Vibration Qualification Tests of Electrical and Electronic equipment and Moving Mechanical Assemblies. See Table 4–1.

EXCEPTION:

Acoustic/Random Vibration Qualification testing will not be performed on the SARJ Assemblies (Protoflight Units).

RATIONALE:

The SARJ assembly is primary structure as well as a mechanism, and as such is capable of taking loads far in excess of the vibratory loads. The SARJ assembly will be subjected to a static test.

All Electrical and Electronic ORUs mounted on the SARJ Assembly will go through Random Vibration Qualification testing at the ORU level. The only Electrical and Electronic equipment mounted on the SARJ Assembly which will not see Random Vibration Qualification testing will be the wire harnesses. McDonnell Douglas Aerospace will conduct one segment level Acoustic Vibration Qualification test on a representative S3/P3 segment to validate the predicted acoustic vibration levels to which the SARJ ORUs are qualified. Functional tests will not be performed since this is a non functional unit.

PG1–18:

SARJ Assemblies (Protoflight Units), RJMC, and DLA

SSP 41172 REQUIREMENT:

EMI/EMC Qualification Tests of Electrical and Electronic equipment**.** See Table 4–1.

EXCEPTION:

EMI/EMC Qualification testing will not be performed on the SARJ Assemblies (Protoflight Units).

RATIONALE:

EMI/EMC Qualification testing will be performed using an RJMC and a DLA in a screen room, not at the SARJ Assembly level. The Utility Transfer Assembly (UTA) will go through a separate EMI/EMC Qualification test.

EMI/EMC Acceptance testing will be performed at the SARJ integration test and at the segment level acceptance tests for S3 and P3.

PG1–19:

Trundle Bearing Assembly

SSP 41172 REQUIREMENT:

EMI/EMC Qualification Tests of Electrical and Electronic equipment. See Table 4–1.

EXCEPTION:

EMI/EMC Qualification testing will not be performed on the Trundle Bearing Assembly.

RATIONALE:

The only electronic or electrical equipment on the Trundle Bearing Assemblies are the resistor box assembly and limit switches which are for Failure Detection, Isolation, and Recovery (FDIR).

The power requirements for operating the Trundle Bearing Assembly are one half of a Watt, and McDonnell Douglas Aerospace determined that EMI/EMC Qualification testing was not required for the Trundle Bearing Assembly

EMI/EMC Acceptance testing will be performed at the SARJ integration test and at the segment level acceptance tests for S3 and P3.

PG1–20:

This exception has been deleted. It has been replaced by PG1–222.

PG1–21:

Solar Alpha Rotary Joint Launch Restraint Mechanism Part Number 1F83193

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Solar Alpha Rotary Joint Launch Restraint Mechanism will not undergo an Acceptance Thermal Vacuum Test.
The Solar Alpha Rotary Joint Launch Restraint Mechanism will not undergo an Acceptance Random Vibration Test.

RATIONALE:

The SARJ Launch Restraint is a simple clamp device that consists of only a few parts. The primary features are made up of the same materials with the same coefficient of thermal expansion. Due to the same materials used in the SARJ Launch Restraint Clamshell and the Trunnions, the coefficient of thermal expansion and contraction is the same; therefore, no thermally induced binding will occur. Detailed thermal analysis as documented in MDC 0H1298, SARJ Launch Restraint (SLR) Thermal/Tolerance Analysis, mandatory inspection of the primary features to insure drawing conformance, and successful ground installation (proper running torque measurements and final preload along with proper gap measurements) will insure successful on–orbit removal.

The SARJ Launch Restraint is a preloaded clamping device with a secondary anti–rotation (locking) feature. Testing during development and qualification has shown the launch vibration environment does not affect the SARJ Launch Restraint. Technical review of the hardware indicates that acceptance random vibration testing would not precipitate any workmanship defects. Thus, individual component–level acceptance testing is not required for acceptance screening.

PG1–22:

UTA and DLA

SSP 41172 REQUIREMENT:

Burn–in Acceptance Tests of Electrical and Electronic equipment. See Table 5–1.

EXCEPTION:

Burn–in Acceptance testing will not be performed on the UTA or DLA.

RATIONALE:

Burn–in Acceptance testing will not be performed on the UTA or DLA but subsequent SARJ Assembly tests, SARJ integration tests, and segment level acceptance tests should give us confidence that infancy failures have been eliminated.

PG1–23:

Drive Lock Assembly

SSP 41172 REQUIREMENT:

Thermal Cycling Acceptance Tests of Electrical and Electronic equipment. See Table 5–1.

EXCEPTION:

Thermal Cycling Acceptance testing will not be performed on the DLA.

RATIONALE:

The DLAs will be Thermal Vacuum Acceptance tested. The DLAs will also be mounted on the SARJ Assembly for Thermal Vacuum Protoflight testing. McDonnell Douglas Aerospace has determined that Thermal Cycle Acceptance tests of the DLA are not necessary to detect workmanship defects.

PG1–24:

Rotary Joint Motor Controller

SSP 41172 REQUIREMENT:

Thermal Vacuum Acceptance Tests of Electrical and Electronic equipment. See Table 4–1.

EXCEPTION:

Thermal Vacuum Acceptance testing will not be performed on the RJMC.

RATIONALE:

The RJMC will go through Thermal Cycling Acceptance testing at the component level. The RJMCs will go through Thermal Vacuum Protoflight testing at the SARJ Assembly level.

PG1–25:

Rotary Joint Motor Controller

SSP 41172 REQUIREMENT:

Burn–in Acceptance Tests of Electrical and Electronic equipment. See Table 5–1.

EXCEPTION:

Burn–in Acceptance testing will not be performed on the RJMC.

RATIONALE:

The RJMC will meet the 300 hour burn–in requirement at the SARJ Assembly Level. The 300 hours will be met in an ambient temperature environment.

PG1–26:

Trundle Bearing Assembly

SSP 41172 REQUIREMENT:

Burn–in Acceptance Tests of Electrical and Electronic equipment. See Table 5–1.

EXCEPTION:

Burn–in Acceptance testing will not be performed on the Trundle Bearing Assembly.

RATIONALE:

The only electronic or electrical equipment on the Trundle Bearing Assemblies are the resistor box assembly and limit switches which are for FDIR. Burn–in Acceptance testing will not be performed on the Trundle Bearing Assembly but on subsequent SARJ Assembly tests, SARJ integration tests, and segment level acceptance tests should give us confidence that infancy failures have been eliminated.

<u>PG1–27</u>:

S3 STA Static Loads Test

<u>SSP 41172 REQUIREMENT</u>:

Section 4.3.1.2, Static Structural Load Test Description. "Static loads representing the design–limit and design–ultimate load shall be applied to the structure..."

<u>EXCEPTION</u>:

This is an issue of clarification. Since the S3 STA is integrated with a flight Integrated Equipment Assembly (IEA) (supplied by Rocketdyne), the combined cargo element will only be subjected to 1.1 times limit load rather than ultimate loads (1.4 x limit).

<u>RATIONALE</u>:

This test approach still meets the requirements as specified in NSTS 14046B, where options for structural hardware verification are defined. As stated in NSTS 14046B, this option is acceptable as long as several critical payload elements and/or components are tested to their ultimate load.

<u>PG1–29</u>:

Resource Node

<u>SSP 41172 REQUIREMENT</u>:

Paragraph 4.4.3, Flight Element Acoustic Vibration Qualification Test.

<u>EXCEPTION</u>:

A flight Element Acoustic Vibration Test will not be performed on the Node.

<u>RATIONALE</u>:

Common Module Acoustic Test Data will be used to verify Node component qualification vibration levels. Because the Common Module acoustic test may not be run in time to provide data to verify Node component vibration levels, this approach is considered to be of moderate risk. However, components tested to minimum screening qualification levels should be acceptable for flight.

<u>PG1–30</u>:

Common Hardware, Mechanical Assemblies – includes capture latch, deployment mechanisms, segment to segment attach system, module to truss structure, umbilical mechanism assembly, trailing umbilical system, common attach system, ORU adapters, and MT/Crew and Equipment Translation Aids (CETA) energy absorber.

<u>SSP 41172 REQUIREMENT</u>:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A hot and cold functional test will be performed on the indicated Common Mechanical Systems hardware in lieu of a formal Acceptance Thermal Vacuum Test.

RATIONALE:

The hardware that will be excluded from this requirement are not expected to have any material or workmanship defect that will require thermal vacuum test.

All such Common Mechanical Systems hardware will be subjected to hot and cold thermal extreme functional tests to screen for manufacturing defects which adversely affect performance at the thermal extremes.

During the acceptance testing of each product, data from the qualification thermal vacuum and thermal cycling test will be used to validate the thermal interface on hardware which uses the IMCA.

The IMCA will be subjected to thermal vacuum acceptance test prior to installation on the hardware.

PG1–31:

Common Hardware and Mobile Transporter mechanical assemblies with electrical components – includes capture latch, segment to segment attach system, module to truss structure, umbilical mechanism assembly, trailing umbilical system, common attach system, linear drive unit, and load transfer unit.

SSP 41172 REQUIREMENT:

Section 4.2.5 Random Vibration Test, Component Qualification

Section 4.2.5.4 Supplementary Requirements.

Section 5.1.4 Random Vibration Test, Component Acceptance

Section 5.1.4.4 Supplementary Requirements.

EXCEPTION:

Point of clarification on the Supplementary Requirements.

Electromechanical hardware will not be energized and monitored during the random vibration test.

RATIONALE:

All hardware to be excluded from this requirement will have electrical items independently certified and acceptance tested.

PG1–32:

Flight Elements

SSP 41172 REQUIREMENT:

Paragraph 4.4.2, EMC test flight element.

EXCEPTION:

EMC test performed at the element level will be Acceptance, not Qualification tests. The test will ensure self and inter–compatibility of the flight electrical/electronic equipment installed in the element that is being tested, to the extent achievable or to the limits imposed by the fidelity of simulated interface elements or ORUs.

RATIONALE:

System verification cannot be accomplished because each element requires intrusive testing (signal injection) that may result in stressing flight hardware. The data derived from element acceptance testing will be provided to the Prime for input to the computer aided test data base for EMC system analysis for system qualification (Reference SS–VE–058).

PG1–33:

Components general

SSP 41172 REQUIREMENT:

Paragraph 4.2.12, EMC test, component qualification.

EXCEPTION:

General clarification

RATIONALE:

The term component as used in this paragraph is interpreted by McDonnell Douglas Aerospace to mean the test article or Unit Under Test (UUT). The UUT may be a single ORU (blackbox) or a collection of interconnected ORUs subjected to the SSP 30238 test requirements. The term component is not applicable for subassemblies or parts.

PG1–36:

Utility Rails (internal and external)

SSP 41172 REQUIREMENT:

Table 5–1 calls for Thermal Vacuum, Thermal Cycle and Burn–In on Electronic and Electrical equipment.

EXCEPTION:

Thermal Vacuum, Thermal Cycle and Burn–In acceptance tests are not performed on Utility Rails.

RATIONALE:

Utility Rails are a very simple electronic design with no active components and these environmental tests are not necessary to detect workmanship defects.

PG1–37:

Radiator assembly ORU

SSP 41172 REQUIREMENT:

Thermal vacuum acceptance test for moving mechanical assemblies, fluid equipment, and thermal equipment. See Table 5–1.

EXCEPTION:

Thermal cycle testing rather than thermal vacuum testing conducted at ORU level to verify workmanship

RATIONALE:

Size of deployed assembly precludes factory testing. Qualification of ORU at Plumbrook Space Power Facility (SPF).

PG1–38:

Heat exchanger, cold plate, NH3 tank, N2 tank, and pump module ORUs.

SSP 41172 REQUIREMENT:

Thermal vacuum and thermal cycle qualification tests for fluid equipment and thermal equipment. See Table 4–1.

EXCEPTION:

Thermal cycle qualification testing conducted at ORU level to verify design (The cold plate and heat exchanger ORUs are not tested to either environment).

RATIONALE:

Functional piece parts qualified to thermal vacuum environment. ORU level thermal vacuum exposure at system level (Thermal Test Article). All hardware conduction mounted, radiation not primary mode of heat transfer.

PG1–39:

Heat exchanger, cold plate, NH3 tank, N2 tank, and pump module ORUs

SSP 41172 REQUIREMENT:

Thermal vacuum acceptance tests for fluid equipment and thermal equipment. Thermal cycle acceptance test also for fluid equipment. See Table 5–1.

EXCEPTION:

No thermal vacuum or thermal cycle testing conducted at ORU level except for the Pump Module ORU that is thermal cycle tested.

RATIONALE:

Functional items that are a part of the ORU are accepted tested to the thermal cycle environment but not thermal vacuum. All hardware is conduction mounted, and radiation is not the primary mode of heat transfer. Thermal cycle tests will detect manufacturing defects. Structural integrity verified by random vibration testing.

PG1–40:

Plumbing on: Segments S1, P1, S0, S3 and P3; Nodes 1, 2; Heat exchanger, cold plate, NH3 Tank, N2 Tank, and pump module ORUs

SSP 41172 REQUIREMENT:

Ultimate pressure qualification tests for fluid equipment. See Table 4.1.

EXCEPTION:

McDonnell Douglas Aerospace manufacturing will use sample weld connections, which will include different tube diameters and wall thickness.

RATIONALE:

McDonnell Douglas Aerospace will qualify the weld process and not the actual flight hardware.

PG1–41:

LVS Radiator Assembly ORU qualification

SSP 41172 REQUIREMENT:

Paragraph 4.2.1.3, Supplementary Requirements ”...A functional test shall be conducted prior to each of the environmental tests to ensure that performance meets the requirements of the particular specification.”

EXCEPTION:

The LVS Radiator Assembly qualification tests include a static yield loads test (in the stowed configuration), a thermal cycle test, an electrical integration test, and three squib firing tests before performing a deploy and retract functional check. Later in the qualification series, a deployed MS, deployed static yield loads, proof and leak pressure, and electrical conductivity tests are completed without functional tests between them.

RATIONALE:

This approach was driven by cost convergence and schedule and is a success oriented test program. A failure in the post test deploy/retract functional test will result in a series of tests (which span up to six months) being repeated so that the cause of the failure can be isolated. Because of possible schedule impacts, approach is considered to be of moderate risk.

PG1–42:

LVS Radiator Assembly ORU acoustic acceptance tests of 3 of the 6 ORUs

SSP 41172 REQUIREMENT:

Paragraph 5.1.5.3, Test Levels And Duration, or paragraph 6.1.1, Assembly/Components Protoflight Tests.

EXCEPTION:

Three of the LVS Radiator Assembly flight units will be included in the S1 STA acoustic qualification test. These flight radiators will be subjected to protoflight levels and durations up to 120 seconds.

RATIONALE:

This approach was adopted during cost convergence in order to eliminate radiator ORU acoustic simulators.

PG1–44:

Control Moment Gyro (CMG) assembly

SSP 41172 REQUIREMENT:

Thermal vacuum test (paragraph 4.2.2.3 – Duration) A minimum of three temperature cycles

EXCEPTION:

Thermal vacuum test for a minimum of two cycles

RATIONALE:

The CMG requires 12 hours to bring up to full RPM and another 12 hours to despin, plus 108 hrs for thermal stabilization. This is consistent with the Space Station Freedom Program (SSFP) requirements which have been imposed on the subcontractor.

PG1–45:

CMG assembly

SSP 41172 REQUIREMENT:

Thermal cycle test (paragraphs 4.2.3.3, 5.1.3.2, and 5.1.3.3 – Description)

The article shall be turned off, allowed to stabilize at the cold temperature, and then started.

EXCEPTION:

The CMG shall be turned off, allowed to stabilize at the cold temperature, and then started when the bearing temperature reaches 30 degrees F.

RATIONALE:

The CMG bearing/lubricant system is designed to operate at temperature in excess of 30 degrees F through the use of heaters.

PG1–46:

CMG mechanical assembly

SSP 41172 REQUIREMENT:

Minimum screening level of 6.1 g RMS. See Figure 5–2.

EXCEPTION:

CMG mechanical screening levels will be 4.3 grms for acceptance and 8.6 grms for qualification (i.e. 6 dB of margin).

RATIONALE:

The CMG bearings life during acceptance and survival of bearings during qualification may be a problem at the minimum screening levels. The Electronic Assembly (EA) will be subjected to the minimum screening level of 6.1 grms for acceptance and 10.9 grms for qualification. This is consistent with the SSFP requirements which have been imposed on the subcontractor.

PG1–47:

Rate Gyro Assembly (RGA)

SSP 41172 REQUIREMENT:

Functional (performance) test prior to and after each environmental test to assure that the performance meets the requirements. (throughout section 4.2 and 5.1).

EXCEPTION:

An RGA abbreviated functional test, which consists of a subset of the functional (performance) test will be performed prior to and after each environmental test. The abbreviated functional test includes the parameters necessary to verify proper operation of the RGA. This includes monitoring for failures and intermittences.

Note: the functional test as defined in 4.2.1 is considered a system performance test which verifies parameter output against specification performance requirements.

RATIONALE:

Functional (performance) tests will be performed prior to any environmental test and at the completion of the environmental test phase. This is consistent with the SSPF requirements which have been imposed on the subcontractor.

PG1–48:

Test Tolerances

SSP 41172 REQUIREMENT:

Section 3.2, Test Condition Tolerances.

”Unless otherwise specified, the following maximum allowable tolerances on test conditions shall apply.”

EXCEPTION:

McDonnell Douglas Aerospace has specified other test tolerances for random vibration, acoustic, and shock testing. This is a point of clarification to be sure the intent of the document was to impose test tolerances only when none were specified.

RATIONALE:

The test tolerances specified by McDonnell Douglas Aerospace are in most cases more stringent and are more compatible with modern test equipment than the ones specified in SSP 41172.

PG1–49:

Wire Harnesses

SSP 41172 REQUIREMENT:

Perform environmental qualification and acceptance tests defined in Tables 4–1 and 5–1 on electrical equipment.

EXCEPTION:

Environmental qualification and acceptance tests are not performed on wire harnesses.

RATIONALE:

Inspection will be used to ensure that the correct materials and manufacturing techniques are specified such that environmental requirements are met.

PG1–50:

Fluid lines

SSP 41172 REQUIREMENT:

Perform environmental qualification and acceptance tests defined in Tables 4–1 and 5–1 on fluid equipment.

EXCEPTION:

Environmental qualification and acceptance tests are not performed on fluid lines.

RATIONALE:

Inspection will be used to ensure that the correct materials and manufacturing techniques are such that environmental requirements are met. Several typical fluid line installations will be subjected to qualification acoustic vibration tests.

PG1–53:

Flight Elements

SSP 41172 REQUIREMENT:

Pressure/Leak Qualification tests shall be conducted on Flight Elements. See Table 4–2.

EXCEPTION:

Pressure/Leak Qualification tests will not be performed on the Flight Elements.

RATIONALE:

Fluid lines used on Flight Elements will be qualified for pressure at the coupon/component level. Acceptance tests at proof pressure will be performed on the flight elements.

PG1–54:

Flight Elements

SSP 41172 REQUIREMENT:

Functional tests shall be conducted prior to and following EMC and pressure/leak environmental tests. Reference Table 5–2, Notes.

EXCEPTION:

Functional tests will not be performed on Flight Elements before or after the EMC or pressure/leak tests.

RATIONALE:

Fight Element EMC tests will be performed concurrently with the functional acceptance tests. The EMC tests to be conducted will not be detrimental to the performance of the flight article.

All proof pressure tests on the fluid lines will be conducted prior to installation on the segments. McDonnell Douglas Aerospace will not perform functional tests before or after the proof pressure tests.

The leak tests will be conducted on the flight articles as a part of the segment acceptance test. McDonnell Douglas Aerospace will not perform functional tests before or after the leak tests.

PG1–55:

Static structural loads test

SSP 41172 REQUIREMENT:

Structural qualification test (paragraph 4.3.1.2, Test Description).

EXCEPTION:

Static structural loads test is performed for maximum flight loads (coupled loads) not for design loads

RATIONALE:

If we tested to design loads, we would have increase our test span to account for 7 additional load cases (800 engineering hours of additional testing). If we were flying multiple times, testing to design loads is a good approach, but since we are only flying once, it is excessive.

PG1–56:

Flight Elements

SSP 41172 REQUIREMENT:

Section 4.4.1, Flight Element Functional Qualification Test.

Mechanical and electrical functional tests shall go beyond specified operational limits.

A segment of the test shall go through a mission profile with all events happening in actual flight sequence.

EXCEPTION:

PG1–01 does not have element qualification units so the only articles available for test are the flight units. They will not go through functional qualification, only acceptance.

Mechanical and electrical functional tests at the element level are performed at nominal performance requirements and will not go beyond specified operational limits.

Not planning a mission profile test with all events happening in actual flight sequence.

RATIONALE:

Functional operational limits will be demonstrated for the electrical and mechanical components at the component level. Full functionality of the element and its interfaces will be verified during the acceptance test. Mission profiles may be performed as practical but it is not planned to formally go through a complete mission profile.

PG1–57:

Modal Survey test

SSP 41172 REQUIREMENT:

Modal Survey (paragraph 4.3.2, Test Description). The data obtained shall be adequate to define orthogonal mode shapes, mode frequencies, and mode damping ratios of all modes which occur within the frequency range of interest.

EXCEPTION:

Only Target modes as defined by an effective mass greater than 2 percent will meet this criteria.

RATIONALE:

The Target modes will contribute more than 90 percent of the total structural loads. The remaining 10 percent will not be significant to the system structural response.

PG1–58:

Pressurized Mating Adapter (PMA)

SSP 41172 REQUIREMENT:

Paragraph 4.4.2, Electromagnetic Compatibility Test, Flight Element Qualification.

EXCEPTION:

EMC test performed at the element level will be Acceptance, not Qualification tests. The test will ensure self and inter–compatibility of the flight electrical/electronic equipment installed in the element that is being tested, to the extent achievable or to the limits imposed by the fidelity of simulated interface elements or ORUs.

RATIONALE:

Compliance to the Specification Requirements will be shown by Analysis. System testing cannot be accomplished because each element requires intrusive testing (signal injection) that may result in stressing flight hardware. The data derived from element acceptance testing will be provided to the Prime for input to the CAT data base for EMC system analysis for system qualification (Reference SS–VE–058).

PG1–59:

SARJ Qualification Hardware – RJMC, DLA, UTA, and Trundle Bearing Assembly

SSP 41172 REQUIREMENT:

Paragraph 4.1. Acceptance testing, including functional and environmental, shall be conducted on all test articles prior to or in conjunction with qualification tests.

EXCEPTION:

Environmental acceptance testing will not be performed prior to or in conjunction with qualification tests.

RATIONALE:

Lockheed Missiles and Space Corporation (LMSC) Qualification test levels will, as a minimum, encompass or exceed all environmental acceptance test levels. LMSC will perform functional testing prior to and following environmental qualification tests. Significant cost and schedule impacts would be incurred if LMSC were forced to change their testing philosophy.

PG1–62:

SARJ – Second Protoflight Assembly

SSP 41172 REQUIREMENT:

Paragraph 6.1. ”Subsequent assemblies to the first assembly subjected to protoflight tests shall be subjected to identical protoflight tests.”

EXCEPTION:

The first SARJ assembly will be tested in a thermal vacuum environment to 10 degrees F beyond the maximum and minimum predicted temperatures. A thermal balance test will be performed to support analysis. The second SARJ assembly will not be subjected to a thermal vacuum test or a thermal balance test.

RATIONALE:

All of the SARJ ORUs (RJMC, UTA, DLA, and Trundle Bearings) will be qualification and acceptance tested in a thermal vacuum environment. Qualification testing of the ORUs will be performed in a thermal vacuum environment to 20 degrees F beyond maximum and minimum predicted temperatures. The only remaining electrical items on the SARJ assembly not tested in a thermal vacuum environment are the wire harnesses (exception PG1–49 exempts wire harnesses). The first flight SARJ unit will be tested in a thermal vacuum environment to protoflight levels (10 degrees F beyond the maximum and minimum predicted temperatures). This position was established as a part of the cost convergence activities.

PG1–63:

Common Hardware – Mechanical Assemblies which include Deployment Mechanisms (UHF, ACS Antenna, and Charged Particle Directional Spectrometer), Segment–to Segment Attach System (SSAS), Module–to–Truss Structure (Adjustable Strut and Attach System), Trailing Umbilical System Cable Guide Assembly, and Common Attach System (Unpressurized Logistics Carrier Attach System and PAS).

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Thermal extreme functional tests at qualification levels will be performed on the indicated Common Mechanical Systems hardware assemblies in lieu of a formal Qualification Thermal Vacuum Test.

RATIONALE:

A thermal test only, not thermal/vacuum, will be performed on the Common Hardware – Mechanical Assemblies to certify the design for operation at temperature extremes. To perform these tests under vacuum conditions would result in a substantial increase in cost and complexity of test, and was deemed unnecessary since associated sub–assemblies are independently certified in a thermal vacuum environment, or these assemblies do not have vacuum sensitive components.

The Segment–to–Segment Attach System Striker Assembly and Segment–to–Segment Attach System Latch EVA Extension will not undergo a thermal vacuum or thermal extreme test at qualification levels. Rationale for this exception is documented in PG1–257.

PG1–64:

Quick Disconnects

SSP 41172 REQUIREMENT:

Acceptance random vibration, thermal cycle and thermal vacuum tests required on Thermal and Fluid Equipment. See Table 5–1.

EXCEPTION:

Acceptance random vibration, thermal cycle, and thermal vacuum tests will not be performed.

RATIONALE:

Each quick disconnect (QD) coupling will undergo acceptance leak testing in both mated and unmated conditions at operational pressure and temperature extremes with helium as the test fluid. Acceptance leak testing at extreme temperatures is the most cost effective test to screen each QD coupling for workmanship defects and satisfies the intent of SSP 41172.

PG1–65:

Heat Exchanger ORU and Cold Plate ORUs

SSP 41172 REQUIREMENT:

Qualification thermal vacuum tests required on Thermal Equipment. See Table 4–1.

EXCEPTION:

Qualification thermal vacuum tests will not be performed on the Heat Exchanger ORU and Cold Plate ORUs.

RATIONALE:

The ORU component hardware (e.g. cold plate, valves, heat exchanger) will undergo Qualification thermal vacuum testing. The Thermal Test Article, which will include the ORUs will be subjected to a vacuum test (not thermal vacuum)

PG1–66:

Utility Rail

SSP 41172 REQUIREMENT:

Equipment level EMC qualification test requirement in accordance with SSP 30238. See SSP 41172, paragraph 4.2.12.2.

EXCEPTION:

No EMC qualification testing to be performed. The data bus (MIL–STD–1553) EMC required measurements will be submitted as an analysis.

RATIONALE:

The Utility Rail contains no active circuits to generate electromagnetic emissions or respond to interference. The unit performs no definitive function that may be tested; therefore, an EMC test is not possible.

PG1–67:

Impedance Matching Unit

SSP 41172 REQUIREMENT:

Equipment level EMC qualification test requirement in accordance with SSP 30238. See SSP 41172, paragraph 4.2.12.2.

EXCEPTION:

No EMC qualification testing to be performed.

RATIONALE:

The Impedance Matching Unit contains no active circuits to generate electromagnetic emissions or respond to interference. The unit performs no definitive function that may be tested; therefore, an EMC test is not possible.

PG1–69:

Signal Conditioning Unit

SSP 41172 REQUIREMENT:

Thermal vacuum acceptance tests required on Electronic and Electrical Equipment. See Table 4–1.

EXCEPTION:

Thermal vacuum tests are not performed on the Signal Conditioning Unit.

RATIONALE:

A thermal vacuum qualification test will be performed to certify the design for operation in a vacuum condition, under thermal stress greater than on–orbit conditions. A thermal cycle acceptance test of the Signal Conditioning Unit is the only thermal test deemed necessary to screen for manufacturing and workmanship defects. Since this ORU is a noncritical and very simplistic component and thermal vacuum is not required for screening of this ORU, performing this test will result in added cost with no value added.

PG1–70:

Impedance Matching Unit

SSP 41172 REQUIREMENT:

Thermal vacuum and burn–in acceptance tests required on Electronic and Electrical Equipment. See Table 4–1.

EXCEPTION:

Thermal vacuum and burn–in tests are not performed on the Impedance Matching Unit

RATIONALE:

A thermal vacuum qualification test will be performed to certify the design for operation in a vacuum condition, under thermal stress greater than on–orbit conditions. A thermal cycle acceptance test of the Impedance Matching Unit is the only thermal test deemed necessary to screen for manufacturing and workmanship defects. Since this ORU contains a single capacitor and resistor as the only electrical components, a thermal vacuum test nor burn–in test are required for screening of this ORU, performing this test will result in added cost with no value added.

PG1–71:

Bolt Bus Controller

SSP 41172 REQUIREMENT:

Thermal vacuum acceptance tests required on Electronic and Electrical Equipment. See Table 4–1.

EXCEPTION:

Thermal vacuum tests are not performed on the Bolt Bus Controller.

RATIONALE:

A thermal vacuum qualification test will be performed to certify the design for operation in a vacuum condition, under thermal stress greater than on–orbit conditions. A thermal cycle acceptance test of the Bolt Bus Controller is the only thermal test necessary to screen for manufacturing and workmanship defects. According to a NASA Environmental Acceptance Testing specification, SP–T–0023, Revision B, dated September 1975, Thermal/Vacuum cycling will root out manufacturing defects in voids in potting and corona etc. These items are not a concern for the Bolt Bus Controller since there is a high degree of confidence in the processes used. The elimination of this test will result in significant cost savings in the acceptance test area with little or nothing lost in the integrity of the product.

PG1–72:

External Luminaires

SSP 41172 REQUIREMENT:

Paragraph 5.1.2.1, Thermal Vacuum Acceptance testing shall be conducted to detect material and workmanship defects prior to installation into a flight element by subjecting the article to a thermal vacuum environment

EXCEPTION:

Thermal vacuum acceptance testing will not be performed prior to installation into a flight element.

RATIONALE:

During qualification testing, the external luminaires will undergo thermal vacuum as well as thermal cycle testing. Luminaire performance and temperature measurements during the qualification test will demonstrate that the luminaire design is compatible with operation under the vacuum environment and that thermal control is adequate via conduction and radiation paths, and not dependent on convection. Acceptance testing is oriented toward material and workmanship screening; thermal cycle testing is considered the best technique of disclosing any problems caused by thermal stresses. A high degree of repeatibility in luminaire construction from unit to unit is expected; performance of the qualification unit in a vacuum should be representative of the flight units. Therefore, thermal vacuum acceptance testing is not necessary.

PG1–73:

ITEM:

TUS IUA Part Number 1F42993

SSP 41172 REQUIREMENT:

Paragraphs 4.2.2.3 and 4.2.2.4, Qualification Thermal Vacuum testing of components. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Thermal Vacuum test on the combined TUS IUA will not be performed.

RATIONALE:

The TUS IUA component hardware (Impedance Matching Unit and TUS Disconnect Actuator) are independently certified in a thermal vacuum environment. The TUS IUA does not contain any additional vacuum sensitive components. Performing additional qualification thermal vacuum testing at the TUS IUA level of assembly would unnecessarily increase test cost and complexity. The TUS IUA will be subjected to qualification thermal extreme testing to certify compliance with the design requirements.

PG1–74:

ITEMS:

Active Thermal Control System (ATCS) ORUs
Heat Exchanger Part Number 1F28940
DC–to–DC Converter Unit (DDCU) Coldplate Part Number 1F29200
MBSU Coldplate Part Number 1F39990
Nitrogen Tank Part Number 1F96000
Ammonia Tank Part Number 1F28801
Pump Module Part Number 1F96100

SSP 41172 REQUIREMENT:

Paragraph 4.2.3.3C, Test Levels and Duration. The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total.

Note: PG–1 exceptions PG1–38 and PG1–65 eliminated requirement for McDonnell Douglas Aerospace to perform qualification thermal vacuum tests on the ATCS ORUs.

EXCEPTION:

Qualification Thermal Cycle tests will not be performed on the ATCS ORUs. Qualification Thermal Vacuum tests of three cycles will be performed on the ATCS ORUs.

RATIONALE:

The primary purpose for Thermal Cycle testing is to stress electronic equipment to expose faulty components and electrical connections (joints). Full compliance with the specified number of thermal cycles is planned for the electronic controller portion of the Pump and Control Valve Package and the Electronic Control Unit for the Iso, Iso–Relief, and Bypass Valves. The thermal/mechanical portions of the ORUs are not susceptible to degradation or failure by thermal cycling. Thermal Vacuum testing is adequate to demonstrate the design and workmanship by operation over the full temperature specification range. The three qualification Thermal Vacuum cycles will fulfill this purpose. As heaters have been added to the ATCS ORUs, their sizing and operation are better verified by Thermal Vacuum testing.

PG1–75:

ITEM:

Structural Dynamic Measurement System (SDMS) Accelerometers Boeing Part Number SCD 1F08080–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.2.3B and 4.2.3.3B. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

Qualification Thermal Vacuum and Thermal Cycle temperature will be –85 degrees F (acceptance less 10 degrees F) for the SDMS accelerometers.

RATIONALE:

The SDMS, which includes accelerometers, strain gage bridges, signal conditioning units, connecting wires, and controlling software on the five PG–1 truss segments, is not critical for the health of crewmembers or the Space Station structure. Loss of SDMS, or any part therein, will not create a hazard for the crew or the Space Station.

PG1–76:

ITEM:

SARJ Trundle Bearing LMSC Part Number 5846485–501

SSP 41172 REQUIREMENT:

Paragraphs 5.1.2.3 and 5.1.2.4, Acceptance Thermal Vacuum testing components. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Thermal Vacuum test on the Flight SARJ Trundle Bearings will not be performed.

RATIONALE:

The Trundle Bearing will be Thermal Vacuum tested to qualification levels to verify design in full compliance to SSP 41172. Each flight component will be subjected to a temperature extreme test which will include a functional at the hot and cold extreme. Performing Acceptance Thermal Vacuum testing to verify workmanship is not necessary since the Trundle Bearing does not have vacuum sensitive components. Functional testing at temperature extremes is the most cost effective test to screen components for workmanship defects.

PG1–77:

ITEMS:

External Luminaires
CETA Luminaire Part Number 1F03046–1
Video Camera Luminaire Part Number 1F01194–1

SSP 41172 REQUIREMENT:

Paragraphs 4.2.2.2 and 5.1.2.2, Thermal Vacuum Test, Test Description. A temperature cycle begins with the chamber and component at ambient temperature. The temperature of the chamber is reduced to bring the component to the specified low qualification level and stabilized. Temperature stability has been achieved when the rate of change is no more than 5.4 degrees F per hour (3 degrees C per hour).

Paragraphs 4.2.2.3C and 5.1.2.3C, Thermal Vacuum Test, Test Levels and Duration. Transition shall be at a rate no less than 1.0 degree F (0.56 degrees C) per minute.

EXCEPTION:

A temperature cycle begins with the chamber and component at ambient temperature. The temperature of the chamber is reduced to bring the power on component near its stabilized temperature that will be above its specified low qualification level. The component is to be powered off for the remainder of the transition down to the specified low qualification level and stabilized. Temperature stability has been achieved when the rate of change is no more than 5.4 degrees F per hour (3 degrees C per hour).

The transition rate during the External Luminaires assembly thermal vacuum qualification and acceptance test will be less than 1.0 degree F (0.56 degrees C) per minute on an average transition.

RATIONALE:

A pre–test thermal prediction shows that the average rate of temperature change may be as low as 0.33 degrees F per minute during the transition from ambient temperature to 44 degrees F and 0.5 degrees F per minute during transitions from 180 degrees F to –44 degrees F. The predictions show that the requirement will be satisfied during transitions from cold to hot. It has been agreed between prime and PG–1 that the 1 degree F per minute minimum ramp rate requirement should apply to thermal cycle testing only but not to thermal vacuum testing. Test cost constraints dictate that ramp rates be maintained as high as practical. As a result, short term ramp rates will be as high and generally much higher during test than those the hardware will experience for comparable temperature changes on orbit. Transitions between extremes take longer because the rate of change approaches zero as the hardware approaches the target temperature. In the case of the luminaire, the short term ramp rate will exceed 4 degrees F per minute, even though the average over the entire transition is 0.5 degrees F per minute or less. By the same token, transitions between the extremes require months on orbit. Thermal cycle testing for acceptance will be conducted to meet the 1.0 degree F per minute (0.56 degrees C per minute) requirement.

PG1–78:

ITEM:

PMA–1 Protoflight Heat Pipe/Radiator Assemblies Tests Part Number 1F93223 and 1F93224

SSP 41172 REQUIREMENT:

Paragraph 4.2.2.3 and 5.1.2.3. The transition shall be at a rate no less than 1.0 degree F (0.56 degrees C) per minute.

EXCEPTION:

The transition rate during the PMA–1 Protoflight Heat Pipe Radiator assembly thermal vacuum protoflight test will be no less than 0.6 degrees F per minute.

RATIONALE:

A thermal analysis performed by McDonnell Douglas Aerospace identified the worst case on–orbit thermal ramp rate as 0.6 degrees F per minute, and the heat pipe radiator assembly will be tested to this rate. The maximum ramp rate applies only to limited temperature ranges between 52 degrees F and 60 degrees F. The thermal vacuum test will cover the worst case orbital cycle extreme, and will meet the predicted flight ramp rate. Since the thermal vacuum test is intended to simulate flight conditions, it is reasonable to test per analytical predictions.

Also, the PMA–1 Heat Pipe/Radiator assemblies will be subjected to thermal cycle protoflight tests and will meet required ramp rate of 1.0 degrees F (0.56 degrees C) per minute. The 1.0 degrees F (0.56 degrees C) per minute ramp rate during the thermal cycle test will be sufficient to detect workmanship defects.

PG1–79:

ITEM:

Utility Transfer Assembly Part Number 8259150–901

SSP 41172 REQUIREMENT:

Paragraph 4.2.2.3 and 5.1.2.3. The transition shall be at a rate no less than 1.0 degree F (0.56 degrees C) per minute.

EXCEPTION:

The transition rate during the UTA thermal vacuum qualification and acceptance test will be less than 1.0 degree F (0.56 degrees C) per minute on an average transition.

RATIONALE:

A thermal analysis performed by McDonnell Douglas Aerospace identified the worst case on–orbit thermal ramp rate as 0.29 degrees F per minute. The maximum ramp rate applies only to a limited temperature ranges between 145 degrees F and 136 degrees F. The thermal vacuum test will cover the worst case orbital cycle extremes. Since the thermal vacuum test is intended to simulate flight conditions, it is reasonable to test per analytical predictions.

Also, the UTA will be subjected to thermal cycle qualification and acceptance tests, and will meet required ramp rate of 1.0 degree F (0.56 degrees C) per minute. The 1.0 degree F (0.56 degrees C) per minute ramp rate during the thermal cycle test will be sufficient to detect workmanship defects.

PG1–80:

ITEM:

Radiator ORU, ATCS Part Number 83–39400–101

SSP 41172 REQUIREMENT:

Thermal cycle test required for qualification and acceptance of fluid and /or thermal components, 24 cycles for qualification, 8 cycles for acceptance (reference Tables 4–1 and 5–1 and Figures 4–1 and 5–1 of SSP 41172).

EXCEPTION:

Thermal cycle test (qualification and acceptance) will not be performed on the Radiator ORU.

RATIONALE:

The primary purpose for thermal cycle testing is to stress the electronic equipment in order to expose faulty components and electrical connections (joints). The Radiator ORU Electrical/Mechanical components (IMCA, Squib Fire Unit/Electrical System, Gear/Brake Assembly, and Pin Puller) will be tested to full SSP 41172 Thermal Cycle compliance. Radiator Panel coupons (bonded and cured with each Radiator panel) will undergo thermal cycle testing; this will provide effective workmanship screening checks for the panels. The mechanical portion of the Radiator ORU is not susceptible to degradation or failure by thermal cycling. The Radiator ORU Qualification Thermal Vacuum deployment test is adequate to demonstrate the design by operation over the full temperature specification range. This test will include three temperature cycles and in addition verify the ORU deployment under temperature gradients.

PG1–81:

ITEM:

Multiplexer/Demultiplexers (MDM) Part Number 8258906–911

SSP 41172 REQUIREMENT:

Paragraph 4.2.2.3, Component Qualification Thermal Vacuum Test Levels and Duration, requires that the component be thermally stabilized (i.e., dwell) for at least 12 hours at the hot and cold extremes of the first cycle.

EXCEPTION:

The dwell time for all cycles will be sufficient to insure that the component has reached internal thermal equilibrium at the specified level, and not less than one hour. This minimum dwell time will be established on the basis of the development testing conducted prior to the qualification tests rather than during the first cycle of qualification testing.

RATIONALE:

The purpose of the extended dwell time during the first cycle of qualification testing is to measure the time required for the component to reach internal thermal equilibrium. This measured time is then intended to be used as the dwell time for subsequent cycles. The MDM dwell time will be measured during development testing prior to qualification testing so there is no added value in the longer dwell time during the first qualification test cycle.

PG1–82:

ITEM:

Control Moment Gyro Part Number 5080097–0001

SSP 41172 REQUIREMENT:

Paragraph 4.1. Acceptance testing, including functional and environmental, shall be conducted on all test articles prior to or in conjunction with qualification tests.

EXCEPTION:

Acceptance thermal vacuum testing of a qualification unit may be accomplished in either one of two ways. The qualification unit may be subjected to acceptance thermal vacuum testing prior to, and separate from, the qualification thermal vacuum tests. Alternatively, the qualification unit may be subjected only to the required number of qualification thermal vacuum test cycles. In this case, the first of those qualification test cycles would also constitute the acceptance thermal vacuum test of the qualification unit “in conjunction with” the qualification test. A separate acceptance thermal vacuum test of the qualification unit is not required.

RATIONALE:

Component acceptance thermal vacuum tests and qualification thermal vacuum tests are the same except that the qualification tests include more cycles (3 versus 1 per SSP 41172) and the qualification temperatures are more extreme. Thus the first qualification thermal vacuum test cycle encompasses the requirements of the acceptance thermal vacuum test.

A separate acceptance test cycle performed prior to the qualification test will not identify any design or workmanship flaws that will not be identified in the qualification test. When the acceptance testing of the qualification unit occurs some time before the qualification tests, the acceptance test may surface such flaws earlier and reduce risk in the qualification program. However, when the acceptance tests are scheduled immediately prior to the qualification tests, as is the case with the CMGs, there is no value added by the separate test. Nevertheless, the separate test will add significant cost and schedule time.

Boeing has already accepted this interpretation of SSP 41172 for thermal cycle tests of qualification units as documented in the PG–1 Master Verification Plan based on analogous rationale. McDonnell Douglas Aerospace believes that the interpretation is equally valid for the thermal vacuum tests.

PG1–83:

ITEM:

Radiator ORU Part Number 83–39400–101

SSP 41172B REQUIREMENT:

Paragraphs 4.2.2.3C and 5.1.2.3C. Transition shall be at a rate of no less than 1.0 degree F per minute.

EXCEPTION:

The thermal transition rate during the Radiator ORU Thermal Vacuum Qualification Test will be approximately 0.5 degrees F per minute, which will be less than the 1.0 degree F per minute requirement.

RATIONALE:

The primary purpose of performing a Thermal Vacuum test is to determine if the component and ORU will perform when exposed to the worst case on–orbit transient thermal environment (plus margin for Qualification). The most severe on–orbit transient thermal environment occurs when the ISS enters and exits the earths shadow which produces a step change in the thermal environment. The Radiator ORU Thermal Vacuum Qualification test ramps the Thermal environments at 1/10th or less of the Radiator ORU skin time constant (this is accomplished with a step change of IR lamp power). Thus, the skin temperatures will change at a rate as severe as they would ever experience on–orbit. The test will transition the chamber environment from –125 to +125 degrees F in 2 to 3 minutes.

The Radiator ORU is designed to be thermally isolated from the environment in which it operates. It is therefore not practical to attempt to transition the ORU components from one temperature extreme to another at an arbitrary rate of change which is difficult or impossible to achieve with a step change in the thermal environment.

The Radiator ORU Electrical and Mechanical components (IMCA, Squib Fire Unit/Electrical System, Gear/Brake Assembly, and Pin Puller) will be Thermal Vacuum/Cycle tested to full compliance to SSP 41172. These components will therefore be exposed to the minimum 1.0 degrees F per minute ramp rate at the component level.

PG1–84:

ITEM:

PMA–1 Heat Pipe Passive Radiator Part Numbers 1F93223 and 1F93224

SSP 41172B REQUIREMENT:

Paragraph 6.1.1. The minimum number of cycles of protoflight thermal vacuum and thermal cycle testing shall be the same as the number for qualification testing (Three cycles of thermal vacuum and 24 cycles of thermal cycles).

EXCEPTION:

The minimum number of cycles for the thermal vacuum test shall be one and the minimum number of cycles for the thermal cycle test shall be eight. Note: Two thermal vacuum cycles are planned.

RATIONALE:

The minimum number of cycles for qualification is three times the number of acceptance cycles in order to ensure that the flight hardware can survive multiple retests if necessary. For protoflight items, these additional cycles are not necessary.

PG1–85:

ITEMS:

Heat Exchanger ORU Part Number 1F28940–1
MBSU Cold Plate ORU Part Number 1F39990–501
DDCU Cold Plate ORU Part Number 1F29200–501
Ammonia Tank Assembly Part Number 1F28801–1
Nitrogen Tank Assembly Part Number 1F96000–1
Pump Module Assembly Part Number 1F96100–1

SSP 41172B REQUIREMENT:

Paragraph 4.2.2.3C, Duration. A minimum of three temperature cycles shall be used. During the first cycle the component shall be thermally stabilized for at least 12 hours at both the high and low temperature extremes with power off, and then turned on. During subsequent cycles, shorter dwell times may be used that are equivalent to those required to reach internal thermal equilibrium during the first cycle but not less than one hour. Transition shall be at a rate no less than 1.0 degree F (0.56 degrees C)

EXCEPTION:

Duration. A minimum of one thermal cycle shall be used. The component shall be thermally stabilized for at least 12 hours at both the high and low temperature extremes with power off and then turned on.

RATIONALE:

Three Qualification Thermal Vacuum test cycles are performed to provide margin over the one required for the Acceptance Thermal Vacuum test to allow for Acceptance retest. This philosophy assures the design is adequate and provides confidence that the hardware will successfully pass Acceptance testing. No ATCS ORU Thermal Vacuum Acceptance testing is required; therefore, multiple thermal vacuum qualification cycles are not required.

The one cycle qualification test approach has been deemed acceptable to verify the passive design of the Heat Exchanger, Cold Plates, Nitrogen Tank Assembly, Ammonia Tank Assembly, and Pump Module ORUs. The Thermal Vacuum Test is not a stress screening test; therefore, one cycle will verify ATCS ORU performance at temperature extremes. ORU components are qualification and acceptance test thermal stress screened to SSP 41172 requirements except for the Quick Disconnects in accordance with SSP 41172 exception PG1–64.

PG1–86:

ITEM:

PMA–1 Heat Pipe Passive Radiator Part Numbers 1F93223–1 and 1F93224–1

SSP 41172B REQUIREMENT:

Paragraph 6.1.1C. For the acoustic vibration qualification test, the test level shall be the maximum predicted flight level but not less than a level derived from an acoustic environment of 141 dB overall (whose spectrum is defined by NSTS 21000–IDD–ISS, paragraph 4.1.1.5).

EXCEPTION:

The PMA–1 MDM Heat Pipe Passive Radiator will not be acoustic noise tested to the specified levels at the 31.5 Hertz, 40 Hertz, and 50 Hertz bands.

RATIONALE:

Finite Element Model analysis shows that the first vibration mode of the MDM heat pipe radiator assembly is 77 Hertz. Lack of acoustic energy in the three frequency bands does not affect the validity of the vibroacoustic test result since there is not a vibration mode below 63 Hertz. Facility limitations will not allow testing at the lower bands.

PG1–87:

ITEM:

Utility Transfer Assembly Part Number 8259150–901

SSP 41172B REQUIREMENT:

Paragraph 4.2.2.2. Temperature stability has been achieved when the rate of change is no more than 5.4 degrees F per hour (3 degrees C per hour).

EXCEPTION:

Allow the worst case temperature stabilization rate of 7.9 degrees F per hour.

RATIONALE:

The stabilization requirement is intended to ensure the UUT soaks at the thermal extremes. The UTA temperature varied between –62.5 degrees F and –70.6 degrees F over a 140 minute cycle. The target temperature was –65 degrees F. The UTA spent approximately 125 minutes per cycle colder than –65 degrees F. The high delta T/delta t rates were encountered when the temperature was around –65 degrees F, with much lower delta T/delta t rates occurring around –70 degrees F. The worst case rate occurred at the transition from –70.6 degrees F to –64 degrees F over a 53 minute period.

PG1–88:

ITEM:

Utility Transfer Assembly Part Number 8259150–901

SSP 41172B REQUIREMENT:

Paragraph 4.2.2.3C. The transitions shall be at a rate no less than 1.0 degree F (0.56 degrees C) per minute.

EXCEPTION:

Allow an average transition rate of greater than or equal to 0.29 degrees F per minute.

RATIONALE:

Analysis has identified the worst case on–orbit thermal transition rate as 0.29 degrees F per minute. This maximum transition rate applies to a limited temperature range between 136 degrees F and 145 degrees F. The Thermal Vacuum transition rate will meet the predicted flight transition rate for the worst case orbital temperature extremes. In addition, the thermal cycle qualification and acceptance tests will meet the required transition rate of 1.0 degree F per minute.

PG1–89:

ITEM:

Utility Transfer Assembly Part Number 8259150–901

SSP 41172B REQUIREMENT:

Paragraph 4.2.3.2. After the component temperature has stabilized at less than 5.4 degrees F per hour (3 degrees C per hour) rate of change, the article shall be turned off, allowed to stabilize at the cold temperature, and then started.

EXCEPTION:

Allow power–on condition before the hot and cold starts on cycles 2 through 11 and cycles 14 through 23.

RATIONALE:

Power on/off dwells do not drive UTA performance. The "power–on" is power applied to the UTA pass through lines and the "hot/cold start" is rotation of the UTA via Special Test Equipment. Application of power has no effect on the breakaway torque of the UTA.

PG1–90:

ITEM:

Utility Transfer Assembly Part Number 8259150–901

SSP 41172B REQUIREMENT:

Paragraph 4.2.3.3C.

(1) The dwell time at the high and low levels shall be long enough to obtain internal thermal equilibrium.

(2) The transitions shall be at a rate no less than 1.0 degree F (0.56 degrees C) per minute.

EXCEPTION:

(1) Allow an internal thermocouple RTD to not reach the cold temperature extreme of –45 degrees F in 14 out of 24 cycles or the hot temperature extreme of 177 degrees F in 6 out of 24 cycles.

(2) Allow three transitions from hot to cold at 0.93, 0.87, and 0.99 degrees F per minute and one transition from cold to hot at 0.81 degrees F per minute.

RATIONALE:

(1) While not all thermocouples reached the extreme temperature conditions (indicating full component internal equilibrium) on all cycles, the control thermocouple did. The control thermocouple also provides measurement of the temperature of the most critical part of the component (the bearings); therefore, the primary target of the thermal stress did reach the required thermal extremes on all cycles. The risk associated with the less critical areas not reaching the required extreme conditions on all cycles is considered minimal.

(2) Twenty–one of the 24 hot to cold transition rates and 23 of the 24 cold to hot transition rates met the 1.0 degree F per minute minimum requirement. The average of all 48 of the thermal transition rates was 1.16 degrees F per minute. No hardware failures occurred due to the qualification thermal cycle or thermal vacuum testing on the qualification UTA or the acceptance thermal cycle and thermal vacuum testing on the first flight UTA.

PG1–91:

ITEM:

TRRJ Number 2, S1 Segment Part Number 5839193–501, S/N 21413–1002

SSP 41172B REQUIREMENT:

(1) Paragraph 6.1.1, Assembly/Components Protoflight Tests

(2) PG1–05, an existing PG–1 exception for the TRRJ Assembly

EXCEPTION:

TRRJ number 2 will not be Acoustic and Pyroshock tested to protoflight levels as required by SSP 41172.

RATIONALE:

The ORUs which are assembled into the TRRJ (Fluid Hose Rotary Coupler, Bearing Assemblies) are all Protoflight Random Vibration tested at their level of assembly. The first TRRJ Assembly is exposed to the acoustic and pyroshock environments either at the TRRJ level for a duration of 30 seconds or as part of the S1/P1 STA Segment for a duration of 2 minutes. The TRRJ number 1 Acoustic test is performed in order to verify the levels to which the TRRJ ORUs were exposed.

PG1–92:

ITEM:

Power Data Grapple Fixture (PDGF) Rigid Umbilical, S0 Part Number 1F75432

SSP 41172B REQUIREMENT:

Paragraph 6.1.1C, Assembly/Components Protoflight Tests. For the acoustic vibration qualification test, the test level shall be the maximum predicted flight level, but not less than a level derived from an acoustic environment of 141 dB overall (whose spectrum is defined by NSTS 21000–IDD–ISS, paragraph 4.1.1.5). The duration of test shall be limited to one minute.

EXCEPTION:

Boeing/Hunting Beach is currently using the (SLP/2104 qualification test levels defined in 5.1.1.3. These test levels are derived from an Overall Sound Pressure Level (OASPL) of 139 dB, not the 141 dB level called out by SSP 41172B. The protoflight test duration shall still be 60 seconds.

RATIONALE:

The qualification random vibration levels for SLP mounted hardware are defined in SLP/2104. These levels envelope both acoustic test and flight data for pallet mounted equipment. Since the PDGF Rigid Umbilical is launched on the SLP, Boeing designed the umbilical to these levels.

The qualification acoustic test levels defined in SLP/2104 were originally based on an OASPL of 145 dB. Those levels have been reduced by six dB based on flight data collected from multiple pallet locations and multiple launches. The SLP requirements were used because they are more specific with respect to the PDGF Rigid Umbilical application. As a result the qualification levels for the PDGF Rigid Umbilical are based on OASPL of 139 dB, not 141 dB as called out in SSP 41172B.

PG1–93:

ITEMS:

Extravehicular Activity (EVA) deployed beams and the corresponding fittings as follows:

P3/S3 Faces 3 and 5 Bay 1 Diagonal Beam Assembly (Part Number 1F38649–1) and Bay 1 Bulkhead Fitting Assembly (Part Number 1F38628–1)

P3/S3 Face 4 Bay 2 Diagonal Beam Assembly (Part Number 1F26508–1) and Face 4 Inboard Bay 2 Bulkhead Fitting Assembly (Part Number 1F26511–1)

P3/S3 Face 4 Bay 1 Diagonal Beam Assembly (Part Number 1F26475–1) and Face 4 Outboard Bay 1 Bulkhead Fitting Assembly (Part Number 1F26480–1)

P3/S3 Faces 3 and 5 Bay 2 Diagonal Beam Assembly (Part Number 1F38650–1) and Bay 2 Bulkhead Fitting Assembly (Part Number 1F38630–1)

P3 Face 3 / S3 Face 5 Brace Beam Assembly (Part Number 1F26604–1) and Face 5 Brace Bulkhead Fitting Assembly (Part Number 1F26609–1)

P3 Face 5 / S3 Face 3 Brace Beam Assembly (Part Number 1F26608–1) and Bay 1 Bulkhead Fitting Assembly (Part Number 1F26606–1)

P3 Face 6 / S3 Face 2 Brace Beam Assembly (Part Number 1F26604–1) and Faces 2 and 6 Brace Bulkhead Fitting Assembly (Part Number 1F26615–1)

P3 Face 2 / S3 Face 6 Brace Beam Assembly (Part Number 1F26608–1) and Faces 2 and 6 Brace Bulkhead Fitting Assembly (Part Number 1F26615–1).

SSP 41172B REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification

(1) Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Qualification Thermal Vacuum Test for the EVA deployed beams and corresponding fittings is replaced by a Thermal Extreme Test. The Thermal Extreme Test consists of one cycle at the qualification thermal extremes at ambient pressure.

RATIONALE:

A thermal extreme test will be performed in place of a thermal vacuum test to certify the design for operation at the thermal extremes. These deployment assemblies do not contain vacuum sensitive components nor are the performance parameters sensitive to a vacuum.

The 1F38649–1 diagonal beam and 1F38628–1 fitting will be the test case. This pair were chosen as they provided the worst case deflection, the highest axial load, and the highest EVA actuation forces. All other pairs (beam/fitting) will be qualified by similarity.

The 1F38649–1 diagonal beam and 1F38628–1 fitting will be represented in the test by Structural Test Equipment, with all mechanism components being of a flight configuration.

The mechanism components in the large beams are:

Active Cone Assembly (1F83036–1)

Passive Cone (1F26491–1)

Large Bushing (1F38632–1)

0.7500–16 CRES Hexagonal Head Shear Bolt (1F26653–1)

Slotted Pin (MS16562–219 or 1F26665–1)

Heavy Externally Threaded Rod End Bearing (1F26650–1 or 1F26651–1) (will not be included in the test)

The mechanism component in the large fitting assemblies is:

Large Guide (1F26497–1)

The mechanism components in the SARJ brace beams are:

Active Cone Assembly (1F83036–1)

Passive Cone (1F26491–1)

Small Bushing (1F26494–1)

0.4375–20 CRES Hexagonal Head Shear Bolt (1F26655–1)

Slotted Pin (1F26665–1)

Medium Externally Threaded Rod End Bearing (1F26652–1)

The mechanism components in the small fitting assemblies is:

Small Guide (1F26482–1)

PG1–94:

ITEMS:

EVA deployed beams and the corresponding fittings as follows:

P3/S3 Faces 3 and 5 Bay 1 Diagonal Beam Assembly (Part Number 1F38649–1) and Bay 1 Bulkhead Fitting Assembly (Part Number 1F38628–1)

P3/S3 Face 4 Bay 2 Diagonal Beam Assembly (Part Number 1F26508–1) and Face 4 Inboard Bay 2 Bulkhead Fitting Assembly (Part Number 1F26511–1)

P3/S3 Face 4 Bay 1 Diagonal Beam Assembly (Part Number 1F26475–1) and Face 4 Outboard Bay 1 Bulkhead Fitting Assembly (Part Number 1F26480–1)

P3/S3 Faces 3 and 5 Bay 2 Diagonal Beam Assembly (Part Number 1F38650–1) and Bay 2 Bulkhead Fitting Assembly (Part Number 1F38630–1)

P3 Face 3 / S3 Face 5 Brace Beam Assembly (Part Number 1F26604–1) and Face 5 Brace Bulkhead Fitting Assembly (Part Number 1F26609–1)

P3 Face 5 / S3 Face 3 Brace Beam Assembly (Part Number 1F26608–1) and Bay 1 Bulkhead Fitting Assembly (Part Number 1F26606–1)

P3 Face 6 / S3 Face 2 Brace Beam Assembly (Part Number 1F26604–1) and Faces 2 and 6 Brace Bulkhead Fitting Assembly (Part Number 1F26615–1)

P3 Face 2 / S3 Face 6 Brace Beam Assembly (Part Number 1F26608–1) and Faces 2 and 6 Brace Bulkhead Fitting Assembly (Part Number 1F26615–1).

<u>SSP 41172B REQUIREMENT</u>:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance

(1) Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS

(2) Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS

<u>EXCEPTION</u>:

Acceptance Thermal Vacuum Tests for the EVA deployed beams and corresponding fittings will not be performed.

<u>RATIONALE</u>:

Table 5–1, Note (6), identifies components with close tolerance requiring precise adjustment or that cannot be inspected effectively require a minimum sweep of 100 degrees F for thermal vacuum tests. These items can be inspected effectively to screen out workmanship defects. Inspection will provide the verification that the mechanisms are assembled as designed. The inspection will be augmented by a tolerance analysis to verify the worst case tolerances will not interfere with the performance of the mechanism.

<u>PG1–95</u>:

<u>ITEMS</u>:

EVA deployed beams and the corresponding fittings as follows:

P3/S3 Faces 3 and 5 Bay 1 Diagonal Beam Assembly (Part Number 1F38649–1) and Bay 1 Bulkhead Fitting Assembly (Part Number 1F38628–1)

P3/S3 Face 4 Bay 2 Diagonal Beam Assembly (Part Number 1F26508–1) and Face 4 Inboard Bay 2 Bulkhead Fitting Assembly (Part Number 1F26511–1)

P3/S3 Face 4 Bay 1 Diagonal Beam Assembly (Part Number 1F26475–1) and Face 4 Outboard Bay 1 Bulkhead Fitting Assembly (Part Number 1F26480–1)

P3/S3 Faces 3 and 5 Bay 2 Diagonal Beam Assembly (Part Number 1F38650–1) and Bay 2 Bulkhead Fitting Assembly (Part Number 1F38630–1)

P3 Face 3 / S3 Face 5 Brace Beam Assembly (Part Number 1F26604–1) and Face 5 Brace Bulkhead Fitting Assembly (Part Number 1F26609–1)

P3 Face 5 / S3 Face 3 Brace Beam Assembly (Part Number 1F26608–1) and Bay 1 Bulkhead Fitting Assembly (Part Number 1F26606–1)

P3 Face 6 / S3 Face 2 Brace Beam Assembly (Part Number 1F26604–1) and Faces 2 and 6 Brace Bulkhead Fitting Assembly (Part Number 1F26615–1)

P3 Face 2 / S3 Face 6 Brace Beam Assembly (Part Number 1F26608–1) and Faces 2 and 6 Brace Bulkhead Fitting Assembly (Part Number 1F26615–1).

SSP 41172B REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance

(1) Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS

(2) Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS

EXCEPTION:

Acceptance Random Vibration Tests for the EVA deployed beams and corresponding fittings will not be performed.

RATIONALE:

Table 5–1, Note (7), identifies components with close tolerance requiring precise adjustment or that cannot be inspected effectively require random vibration tests. Inspection will provide the verification that the mechanisms are assembled as designed. The inspection will be augmented by a tolerance analysis to verify the worst case tolerances will not interfere with the performance of the mechanism. The EVA Deployed Beams will be qualified acoustically via the P3/S3 STA vibro–acoustic test including functional testing before and after the vibro–acoustic test.

PG1–96:

ITEMS:

High Rate Frame Multiplexer (HRFM)
High Rate Modulator (HRM)
Video Baseband Signal Processor (VBSP)

SSP 41172 REQUIREMENT:

Paragraph 6.1.1. Thermal cycle duration shall not be less than 24 cycles total, as stated in paragraph 4.2.3.3.c.

EXCEPTION:

During Protoflight Thermal Cycle Testing, a total of eight thermal cycles shall be conducted on the HRFM, HRM, and VBSP.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–97:

ITEMS:

High Rate Frame Multiplexer (HRFM)
High Rate Modulator (HRM)
Video Baseband Signal Processor (VBSP)

SSP 41172 REQUIREMENT:

Paragraph 6.1. Subsequent assemblies/components shall be subjected to identical protoflight tests.

EXCEPTION:

For the HRFM, HRM, and VBSP hardware assemblies, the Depress/Repress, Shock, and EMI/EMC protoflight tests shall not be conducted on the first set of assemblies, originally designated as flight assemblies. The second set of assemblies, originally designated as qualification assemblies, shall be subjected to all the required protoflight tests.

RATIONALE:

The Depress/Repress, Shock, and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–98:

ITEM:

Pump and Control Valve Package (PCVP) Part Number 1F96451

SSP 41172 REQUIREMENT:

Paragraph 4.2.3. Qualification Thermal Cycle Test,

Paragraph 4.2.3.3C. Test Levels and Duration.

The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total. If Thermal Vacuum cycling temperatures encompass those required for thermal cycling, these cycles may be included in the required 24.

EXCEPTION:

The full PCVP (mechanical and electronic portions) will be tested to three thermal cycles for qualification during the thermal vacuum test.

RATIONALE:

The primary purpose for thermal cycle testing is to stress electronic equipment to expose faulty components and electrical connections (joints). The Pump and Control Valve Package Firmware Controller will be tested as a subassembly such that when combined with the testing at the PCVP assembly level will accomplish full SSP 41172 compliance of 24 qualification thermal cycles. The mechanical portion of the PCVP contains structure, a centrifugal pump, and a mixing valve. These items are not susceptible to degradation or failure by thermal cycling. It is adequate to demonstrate their design and workmanship by operation over the full temperature specification range. The three qualification thermal cycle testing will fulfill this purpose.

PG1–99:

ITEM:

Pump and Control Valve Package (PCVP) Part Number 1F96451

SSP 41172 REQUIREMENT:

Paragraph 4.2.3.4. Qualification Thermal Cycle Test Supplementary Requirements. Functional Tests shall be conducted, as a minimum, at the maximum predicted temperature plus 20 degrees F (11.1 degrees C) and at the minimum predicted temperature minus 20 degrees F (11.1 degrees C) during the first and last thermal cycles and after return of the component to ambient.

EXCEPTION:

Maximum temperature during Qualification Thermal Cycling Functional tests will be 110 +/– 5.4 degrees F.

RATIONALE:

The PCVP contains an electronic firmware controller (FC) which is thermal cycle tested prior to being assembled into the PCVP assembly. During this FC thermal cycle test, there is no active cooling from a coldplate as there is at the PCVP level of assembly. Because of this, full functional testing of the FC cannot be performed at the maximum qualification temperature level (+140 degrees F) as required due to concerns over internal temperatures of the electronics. Test data indicates that pump loads create an increase in temperature of the internal electronics of approximately 20 to 25 degrees F. Therefore, a functional test of all functions except pump loads will be performed at the maximum qualification temperature level. Pump load functions will be demonstrated at 110 degrees F during the qualification thermal cycle test. The temperature increase of the internal electronics while performing this piece of the functional test at reduced temperature will satisfy overall test objectives while not subjecting the internal electronics to unrealistic and potentially damaging temperatures.

PG1–100:

ITEM:

Pump and Control Valve Package Part Number 1F96451

SSP 41172 REQUIREMENT:

Paragraph 5.1.3 Acceptance Thermal Cycle Test

Paragraph 5.1.3.3c. Test Levels and Duration.

The minimum number of temperature cycles shall be eight. This test may be performed in thermal vacuum and combined with the component acceptance thermal vacuum test, provided that the temperature limits, number of cycles, rate of temperature change, and dwell times conform to this test.

EXCEPTION:

The full PCVP (mechanical and electronic portions) will be tested to two thermal cycles for acceptance.

RATIONALE:

The primary purpose for thermal cycle testing is to stress electronic equipment to expose faulty components and electrical connections (joints). The PCVP FC will be tested as a subassembly such that when combined with the testing at the PCVP assembly level will accomplish full SSP 41172 compliance of 8 acceptance thermal cycles. The mechanical portion of the PCVP contains structure, a centrifugal pump, and a mixing valve. These items are not susceptible to degradation or failure by thermal cycling. It is adequate to demonstrate their design and workmanship by operation over the full temperature specification range. The two acceptance thermal cycle testing will fulfill this purpose.

PG1–101:

ITEM:

Pump and Control Valve Package Part Number 1F96451

SSP 41172 REQUIREMENT:

Paragraph 5.1.3. Acceptance Thermal Cycle Test.

Paragraph 5.1.3.4. Supplementary Requirements.

Functional Tests shall be conducted during the first and last thermal cycles at the maximum and minimum predicted temperatures and after return of the component to ambient.

EXCEPTION:

Maximum temperature during Acceptance Thermal Cycling Functional tests will be 90 +/– 5.4 degrees F.

RATIONALE:

The PCVP contains an electronic firmware controller which is thermal cycle tested prior to being assembled into the PCVP assembly. During this FC thermal cycle test, there is no active cooling from a coldplate as there is at the PCVP level of assembly. Because of this, full functional testing of the FC cannot be performed at the maximum acceptance temperature level (+120 degrees F) as required due to concerns over internal temperatures of the electronics (see exception PG1–99). In order to maintain margin between qualification and acceptance test temperatures, a functional test of all functions except pump loads will be performed at the maximum acceptance temperature level (+120 degrees F). Pump load functions will be demontrated at 90 degrees F during the FC acceptance thermal cycle test. The temperature increase of the internal electronics while performing this piece of the functional test at a reduced temperature will satisfy overall test objectives, adequately screen the hardware for workmanship, and maintain qualification temperature margins.

PG1–102:

ITEM:

High Rate Frame Multiplexer

SSP 41172 REQUIREMENT:

Paragraph 6.1 "Subsequent assemblies/components shall be subjected to identical protoflight tests."

EXCEPTION:

For the HRFM, the Depress/Repress, Shock, and EMI/EMC protoflight tests shall not be conducted on the first set of assemblies, originally designated as flight assemblies. The second set of assemblies, originally designated as qualification assemblies, shall be subjected to all the required protoflight tests.

RATIONALE:

The Depress/Repress, Shock, and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–103:

ITEM:

High Rate Frame Multiplexer

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 24 Thermal Cycles for protoflight testing.

EXCEPTION:

HRFM will be subjected to eight thermal cycles.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–104:

ITEM:

Assembly Contingency Baseband Signal Processor

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 24 Thermal Cycles for protoflight testing.

EXCEPTION:

Assembly Contingency Baseband Signal Processor will be subjected to eight thermal cycles.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–105:

ITEM:

Assembly Contingency Baseband Signal Processor

SSP 41172 REQUIREMENT:

Paragraph 6.1 "Subsequent assemblies/components shall be subjected to identical protoflight tests."

EXCEPTION:

For the Assembly Contingency Baseband Signal Processor the Shock and EMI/EMC protoflight tests shall be conducted on one unit only.

RATIONALE:

The shock and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–106:

ITEM:

Assembly Contingency Baseband Signal Processor

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 3 Thermal Vacuum Cycles for protoflight testing.

EXCEPTION:

Assembly Contingency Baseband Signal Processor will be subjected to one thermal vacuum cycle.

RATIONALE:

Three thermal vacuum cycles are required for qualification testing in order to provide margin over the one cycle required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–107:

ITEM:

Assembly Contingency Radio Frequency Group

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 24 Thermal Cycles for protoflight testing.

EXCEPTION:

Assembly Contingency Radio Frequency Group will be subjected to eight thermal cycles.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–108:

ITEM:

Assembly Contingency Radio Frequency Group

SSP 41172 REQUIREMENT:

Paragraph 6.1 ”Subsequent assemblies/components shall be subjected to identical protoflight tests.”

EXCEPTION:

For the Assembly Contingency Radio Frequency Group, the Shock and EMI/EMC protoflight tests shall be conducted on one unit only.

RATIONALE:

The shock and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–109:

ITEM:

Assembly Contingency Radio Frequency Group

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 3 Thermal Vacuum Cycles for protoflight testing.

EXCEPTION:

Assembly Contingency Radio Frequency Group will be subjected to one thermal vacuum cycle.

RATIONALE:

Three thermal vacuum cycles are required for qualification testing in order to provide margin over the one cycle required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–110:

ITEM:

High Rate Modem

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 24 Thermal Cycles for protoflight testing.

EXCEPTION:

High Rate Modem will be subjected to eight thermal cycles.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–111:

ITEM:

High Rate Modem

SSP 41172 REQUIREMENT:

Paragraph 6.1 "Subsequent assemblies/components shall be subjected to identical protoflight tests."

EXCEPTION:

For the High Rate Modem, the Depress/Repress, Shock, and EMI/EMC protoflight tests shall not be conducted on the first set of assemblies, originally designated as flight assemblies. The second set of assemblies, originally designated as qualification assemblies, shall be subjected to all the required protoflight tests.

RATIONALE:

The Depress/Repress, Shock, and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–112:

ITEM:

Pan/Tilt Unit

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 24 Thermal Cycles for protoflight testing.

EXCEPTION:

The Pan/Tilt Unit will be subjected to eight thermal cycles.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–113:

ITEM:

Pan/Tilt Unit

SSP 41172 REQUIREMENT:

Paragraph 6.1 "Subsequent assemblies/components shall be subjected to identical protoflight tests."

EXCEPTION:

For the Pan/Tilt Unit, the Shock and EMI/EMC protoflight tests shall be conducted on one unit only.

RATIONALE:

The Shock and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–114:

ITEM:

Pan/Tilt Unit

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 3 Thermal Vacuum Cycles for protoflight testing.

EXCEPTION:

The Pan/Tilt Unit will be subjected to one thermal vacuum cycle.

RATIONALE:

Three thermal vacuum cycles are required for qualification testing in order to provide margin over the one cycle required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–115:

ITEM:

S–Band Transponder (XPNDR)

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 24 Thermal Cycles for protoflight testing.

EXCEPTION:

The S–Band XPNDR will be subjected to eight thermal cycles.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–116:

ITEM:

S–Band Transponder (XPNDR)

SSP 41172 REQUIREMENT:

Paragraph 6.1 ”Subsequent assemblies/components shall be subjected to identical protoflight tests.”

EXCEPTION:

For the S–Band XPNDR, the Shock and EMI/EMC protoflight tests shall be conducted on one unit only.

RATIONALE:

The Shock, and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–117:

ITEM:

S–Band Transponder (XPNDR)

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 3 Thermal Vacuum Cycles for protoflight testing.

EXCEPTION:

The S–Band XPNDR will be subjected to one thermal vacuum cycle.

RATIONALE:

Three thermal vacuum cycles are required for qualification testing in order to provide margin over the one cycle required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–118:

ITEM:

Space–to–Ground Antenna (SGANT)

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 24 Thermal Cycles for protoflight testing.

EXCEPTION:

The SGANT will be subjected to eight thermal cycles.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–119:

ITEM:

Space–to–Ground Antenna (SGANT)

SSP 41172 REQUIREMENT:

Paragraph 6.1 "Subsequent assemblies/components shall be subjected to identical protoflight tests."

EXCEPTION:

For the SGANT, the Shock and EMI/EMC protoflight tests shall be conducted on one unit only.

RATIONALE:

The Shock and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–120:

ITEM:

Space–to–Ground Antenna (SGANT)

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 3 Thermal Vacuum Cycles for protoflight testing.

EXCEPTION:

The SGANT will be subjected to one thermal vacuum cycle.

RATIONALE:

Three thermal vacuum cycles are required for qualification testing in order to provide margin over the one cycle required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–121:

ITEM:

Space–to–Ground Transmitter/Receiver Controller (SGTRC)

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 24 Thermal Cycles for protoflight testing.

EXCEPTION:

The SGTRC will be subjected to eight thermal cycles.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–122:

ITEM:

Space–to–Ground Transmitter/Receiver Controller (SGTRC)

SSP 41172 REQUIREMENT:

Paragraph 6.1 ”Subsequent assemblies/components shall be subjected to identical protoflight tests.”

EXCEPTION:

For the SGTRC, the Shock and EMI/EMC protoflight tests shall be conducted on one unit only.

RATIONALE:

The Shock and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–123:

ITEM:

Space–to–Ground Transmitter/Receiver Controller (SGTRC)

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 3 Thermal Vacuum Cycles for protoflight testing.

EXCEPTION:

The SGTRC will be subjected to one thermal vacuum cycle.

RATIONALE:

Three thermal vacuum cycles are required for qualification testing in order to provide margin over the one cycle required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–124:

ITEM:

TV Camera Interface Converter (TVCIC)

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 24 Thermal Cycles for protoflight testing.

EXCEPTION:

The TVCIC will be subjected to eight thermal cycles.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–125:

ITEM:

TV Camera Interface Converter (TVCIC)

SSP 41172 REQUIREMENT:

Paragraph 6.1 ”Subsequent assemblies/components shall be subjected to identical protoflight tests.”

EXCEPTION:

For the TVCIC, the Shock and EMI/EMC protoflight tests shall be conducted on one unit only.

RATIONALE:

The Shock and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–126:

ITEM:

TV Camera Interface Converter (TVCIC)

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 3 Thermal Vacuum Cycles for protoflight testing.

EXCEPTION:

The TVCIC will be subjected to one thermal vacuum cycle.

RATIONALE:

Three thermal vacuum cycles are required for qualification testing in order to provide margin over the one cycle required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–127:

ITEM:

Video Baseband Signal Processor (VBSP)

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 24 Thermal Cycles for protoflight testing.

EXCEPTION:

The VBSP will be subjected to eight thermal cycles.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–128:

ITEM:

Video Baseband Signal Processor (VBSP)

SSP 41172 REQUIREMENT:

Paragraph 6.1 ”Subsequent assemblies/components shall be subjected to identical protoflight tests.”

EXCEPTION:

For the VBSP Depress/Repress, the Shock and EMI/EMC protoflight tests shall be conducted on one unit only.

RATIONALE:

The shock and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–129:

ITEM:

External Video Switch (VSW)

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 24 Thermal Cycles for protoflight testing.

EXCEPTION:

The VSW will be subjected to eight thermal cycles.

RATIONALE:

Twenty–four thermal cycles are required for qualification testing in order to provide margin over the eight cycles required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–130:

ITEM:

External Video Switch (VSW)

SSP 41172 REQUIREMENT:

Paragraph 6.1 "Subsequent assemblies/components shall be subjected to identical protoflight tests."

EXCEPTION:

For the VSW, the Shock and EMI/EMC protoflight tests shall be conducted on one unit only.

RATIONALE:

The shock and EMI/EMC tests are conducted for design verification, not workmanship verification. Consequently, they are only required to be performed once.

PG1–131:

ITEM:

External Video Switch (VSW)

SSP 41172 REQUIREMENT:

Paragraph 6.1 requires 3 Thermal Vacuum Cycles for protoflight testing.

EXCEPTION:

The VSW will be subjected to one thermal vacuum cycle.

RATIONALE:

Three thermal vacuum cycles are required for qualification testing in order to provide margin over the one cycle required for acceptance testing. In the protoflight testing approach, qualification and acceptance testing are combined, thus eliminating the margin. Additional cycles would only expose the flight hardware to additional stress.

PG1–132:

ITEM:

Marotta Valve Assemblies:
Bypass Valve Assembly (Type 5D Valve, part number 284243–9001),
Isolation Relief Valve Assembly (Type 1D Valve, part number 284180–90001),
Tank Isolation Valve Assembly (Type 4B Valve, part number 284187–9001), and
Isolation Valve Assembly (Type 2E Valve, part number 284185–9001).

SSP 41172 REQUIREMENT:

Paragraph 5.1.3.3: Each cycle shall have a one hour minimum dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on. The dwell time at the high and low levels shall be long enough to obtain internal thermal equilibrium.

EXCEPTION:

The one hour minimum dwell time shall not apply for cycles 2 through 7 for these valves. The dwell time shall be long enough to obtain component internal thermal equilibrium.

RATIONALE:

The critical elements of the acceptance thermal cycle test for workmanship screening is to achieve a rapid rate of change of temperature, to subject the hardware to differential expansion effects and to achieve component internal thermal equilibrium at the temperature extremes prior to performance of the functional tests or beginning transition to the opposite temperature extreme. It is believed that the stress which precipitates defects into failures during the thermal cycle test is mainly a result of the rate of temperature change and mechanical motion resulting from differential expansion and contraction of materials. Marotta's thermal cycle test meets the SSP 41172 requirements for temperature rate of change. Marotta has experimental data showing that the valve reaches internal temperature equilibrium at the critical moving ball component with the external case temperature monitored during the test within 20 minutes at the temperature extremes. The automated test software controls the dwell time with for the first and last cycles at 60 minutes including power–on functional tests of the valve. For cycles two to seven, a 20 minute dwell after temperature stabilization is achieved provides sufficient time for valve internal components to reach thermal equilibrium.

PG1–133:

ITEMS:

Spur Assembly, Airlock, Part Number 1F76242
Structure Installation, Module to Truss–Stowed, Part Number 1F37747
Structure Assembly, Umbilical, Avionics–Port, Part Number 1F75285
Structure Assembly, Umbilical, Avionics–Starboard, Part Number 1F75210
Structure Assembly, Umbilical, Fluids–Port, Part Number 1F75281
Structure Assembly, Umbilical, Fluids–Starboard, Part Number 1F75208
Installation, Aft Lab, Part Number 1F76960

SSP 41172 REQUIREMENT:

Paragraph 4.2.2.2, Thermal Vacuum Test, Component Qualification

(1) Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Qualification Thermal Vacuum Test for the assemblies with spherical bearings is replaced by a Thermal Extreme Test on the component spherical bearings.

RATIONALE:

A thermal extreme component test will be performed in place of a thermal vacuum structure assembly test to certify design for operation at the thermal extremes. These rotating assemblies do not contain vacuum sensitive components nor are the performance parameters sensitive to a vacuum.

The Spherical Bearings KPW9PD3, KPW8PD3, KPD8P, and KPW16CRPD7 will be the test cases. The bearings were chosen because they are used in the Truss Element Moving Mechanical Assemblies and provided the worst case EVA actuation forces for each assembly. The results of these tests will be used in the analysis that will qualify the Forward Avionics and Fluid Umbilicals, Aft Lab Tray, Airlock CETA Spur Hinge, and Mobile Transporter System Struts.

The Test will consist of a Spherical Bearing set in a lug. Two bushings will be bolted together on each side of the bearing. A calibrated torque wrench will be attached to the bolt. The maximum torque required to rotate the bearing will be measured by the torque wrench.

NASA will perform a Human Thermal Vacuum Test on the Spur Assembly, Airlock (1F76242), Structure Installation, Module to Truss–Stowed (1F37747), Structure Assembly, Umbilical, Avionics–Port (1F75285), Structure Assembly, Umbilical, Avionics–Starboard (1F75210), Structure Assembly, Umbilical, Fluids–Port (1F75281), and Structure Assembly, Umbilical, Fluids–Starboard (1F75208).

The assemblies will be functioned in the Human Thermal Vacuum Test for 1 cycle at temperature extremes (approximately –150 degrees F to 170 degrees F). NASA and Boeing–Huntington Beach will coordinate the test requirements. The results of the Human Thermal Vacuum Test will be used to substantiate the analysis which will qualify the assemblies with spherical bearings.

PG1–134:

ITEMS:

Thermal Radiator Rotary Joint (TRRJ) Drive Lock Assemblies (DLA) Part Number 5846872

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)..." This requires protoflight hardware thermal cycling duration to be 24 thermal cycles as indicated in paragraph 4.2.3.3C. SSCN 1088 has since been authorized to reduce the protoflight hardware thermal cycling duration to eight thermal cycles.

EXCEPTION:

A Protoflight Thermal Cycle test will not be performed on the full TRRJ DLAs.

RATIONALE:

The primary purpose for thermal cycle testing is to screen for workmanship defects. Specifically, the test stresses electronic equipment to expose faulty components and electrical connections (joints). The DLA is primarily a mechanical assembly with some passive electrical and electronic components (i.e. Limit Switches, Drive Motor, Resolver, Stepper Motor, and Resistor Network Box). Each of the DLA components are thermal shock tested at the component level as indicated:

(in accordance with MIL–STD–202, Method 107, Test Conditions A and B)

Component	Number of Cycles	Low	High	Dwell/Weight
Limit Switches 8263601 (HSSO)	5	–65 degrees C	125 degrees C	0.5 hours/ 0.026 lbs
Drive Motor T2981054–1 (Honeywell)	5	–67 degrees F	185 degrees F	2 hours/ 16.5 lbs
Resolver 5847054–G (LMMS)	5	–67 degrees F	185 degrees F	2 hours/ 3.7 lbs
Stepper Motor 5847182–NC (LMMS)	5	–67 degrees F	185 degrees F	2 hours/ 10 lbs

The following component was thermal cycle tested in full compliance to SSP 41172 in accordance with SSCN 1088:

Component	Number of Cycles	Low	High
Resistor Network Box G847281 (LMMS)	8	–45 degrees F	160 degrees F

No solder joints are made in the assembly of these electrical components into a DLA. A DLA Protoflight Thermal Vacuum test will be performed and will include three temperature cycles (–55 degrees F to 170 degrees F) in full compliance to SSP 41172. Further, the SARJ DLA is exposed to a Qualification Thermal Cycle Test (–65 degrees F to +182 degrees F) which is performed in full compliance to SSP 41172. The SARJ DLA is identical with respect to the electronic and electrical components, except for the Drive Motor which is slightly larger. The DLAs experience one additional thermal cycle during the TRRJ level Thermal Vacuum test. Additionally, the TRRJ DLAs are Protoflight Random Vibration tested to 8.9 grms and undergoes Burn–In testing of 300 hours. Thus, the DLA component thermal shock tests along with the Thermal Vacuum, Random Vibration, and Burn–In test will screen the TRRJ DLAs for workmanship.

PG1–135:

ITEMS:

Caution and Warning Panel Part Number 1F51710–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.3.3. Test Levels and Duration

(1) Paragraph 5.1.3.3C, Duration. Each cycle shall have a one hour minimum dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on.

EXCEPTION:

The minimum dwell time for the acceptance testing of the Caution and Warning Panel shall be 25 minutes.

RATIONALE:

Due to the simple electromechanical design using solder post construction, thermal dwell time in excess of time to reach internal thermal equilibrium does not significantly add stress to the screening process.

The assembly drawing for the C&W Panel sub–assembly calls for soldering in accordance with NHB 5300.4 (3A–1). This type of solder joint is qualified by NHB 5300.4 (3A–1) to 200 cycles with a larger temperature range (–55 degrees C to 100 degrees C) than the C&W Panel assembly. The dwell time for qualification of solder joints described in NHB 5300.4 (3A–1) is 15 minutes.

Thermal transitions during the thermal cycle test and vibration testing are sufficient to expose workmanship deficiencies. Although SSP 41172 specifically requires a minimum of 1 hour dwell time, it also states: “The dwell time at the high and low levels shall be long enough to obtain internal thermal equilibrium.” Instrument readings obtain during Qualification indicate that thermal equilibrium was reach at 25 minutes. Thus, the acceptance test procedure does meet the intent of the specification.

PG1–136:

ITEMS:

Assembly Power Converter Unit (APCU) Part Number 1F67740.

SSP 41172 REQUIREMENT:

Paragraph 4.2.5.3, Random Vibration Test, Component Qualification, Test Levels and Duration: Component random vibration test levels and spectrums shall be the envelope of the following:

A. The maximum predicted flight level and spectrum, but not less than a level derived from an acoustic environment of 141 dB (whose spectrum is defined by NSTS 21000–IDD–ISS, Table 4.1.1.5–1)

B. Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The APCU input qualification spectrum was notched 3 dB below ICD–A–21321, Shuttle Orbiter/APCU Cargo Element ICD, maximum predicted flight environments for the first 180 seconds of the total 810 seconds in each axis. Since the APCU must be qualified for 25 flights, subjecting it to 810 seconds at the enveloped maximum predicted flight and acceptance environments would have subjected the APCU to an unrealistic and unreasonable qualification environment and risked unnecessary test failure. Therefore, the APCUs were initially random vibration tested for the standard 3 minutes per axis utilizing the enveloped environments with notching, which qualifies the unit for 4 flights and 4 acceptance tests. This was followed with 630 additional seconds per axis at unnotched maximum predicted flight levels as defined in ICD–A–21321 which qualifies the APCU for the remanding (21) required flights. The qualification input frequencies during the first 180 seconds of vibration were notched (in each axis) below maximum predicted flight environments due to the sidewall mounted, items as follows:

A. Power Transformers; Source Control Drawing 1F97557, Rev E; DWG No. 1F97572–1.

B. Input Power Inductor; Source Control Drawing 1F64573, Rev D, DWG No. 1F97573–1

Approximate input notching frequencies where notch depth was below the maximum predicted flight environment are as follows:
X–axis: 205–300 Hz
Y–axis: 240–300 Hz
Z–axis: 160–350 Hz

RATIONALE:

APCUs have been subjected to the following random vibration environments:

A. APCU CITE units, Shuttle fight STS–91.

B. APCU CITE units, protoflight random vibration testing using unnotched protoflight spectra for a duration of 60 seconds/axis.

C. APCU qualification unit, 180 seconds/axis at notched qualification random vibration spectra.

D. APCU qualification unit, 630 seconds/axis at ICD–A–21321 maximum predicted flight spectra.

From a stress and dynamic clearance standpoint, the APCU CITE units, during protoflight vibration testing for STS–91, were subjected to vibration levels equal to, and often much higher than, the maximum predicted flight environment in the frequency ranges where the qualification notch was below the maximum predicted flight level. In addition, the APCU qualification unit was subjected to 630 seconds per axis exposure to an unnotched maximum predicted flight level. Therefore, there is no risk that the hardware has an undetected stress/dynamic clearance issue.

From a fatigue standpoint, the likelihood of any APCU actually experiencing the maximum predicted flight level in service is remote. The probability that one would see the maximum flight environment for 25 missions is infinitesimal. Therefore, the 180 seconds during which the notching was below the maximum predicted flight level is of minimal risk to the demonstrated fatigue life of the hardware.

PG1–137:

ITEMS:

Electronic Assembly Part Number 5092021–9, Serial Numbers 0101, 0102, 0103, and 0104

SSP 41172 REQUIREMENT:

Table 5.1. A minimum random vibration test shall be performed on electronic or electrical equipment

Paragraph 5.1.4.3. Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 5.1.4.3B. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime Contractor

EXCEPTION:

The EA noted is exempt from meeting the workmanship screening level and spectrum shown in Figure 5–2.

RATIONALE:

The EAs were mounted on the CMG during qualification and acceptance random vibration test. The levels specified in Figure 5–2 were not reached in all three axes over the entire frequency band specified. Post test data analysis indicated that the EAs experienced the required minimum workmanship screening input over the 300 to 500 Hertz frequency range in the x–axis (the axis perpendicular to the plane of the Circuit Card Assemblies (CCA)). Since CCAs typically exhibit resonance in the 100 to 500 Hertz frequency range, and the axis perpendicular to the plane of the CCA is the most critical axis for workmanship screening purposes, the conclusion is that the CMG EAs received reasonable workmanship screening.

The risk of undetected workmanship defects due to failure to achieve minimum required screening levels at other frequencies and in the other two axes (y–axis and z–axis) is considered minimal and, when considered with other risks involved in retesting the EA to required minimum workmanship levels (e.g., risk of handling damage and cost and schedule impacts), does not justify the need for rescreening at this time. Rescreening would potentially subject the flight hardware to unnecessary potential fatigue accumulation, have significant cost and schedule impacts, and provide little additional screening benefit. Time Compliance Technical Instruction will invoke Acceptance Vibration Test commensurate with the level of EA parts rework required when any unit is returned for repair.

PG1–138:

ITEMS:

Electronic Assembly Part Number 5092021–9

SSP 41172 REQUIREMENT:

Table 4.1. A minimum random vibration test shall be performed on electronic or electrical equipment

Paragraph 4.2.5.3. Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 4.2.5.3B. Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The electronic assemblies noted do not meet qualification levels for acceptance random vibration over the following frequency bands:

AXIS	FREQUENCY RANGE	VIB LEVEL DIFFERENCE
X AXIS	170 Hz to 200Hz	2 dB low
	1,300 Hz to 2,000 Hz	0 dB to 8 dB low
Y AXIS	200 Hz to 220 Hz	3 dB low
	800 Hz to 900 Hz	3 dB low
	1,300 Hz to 2,000 Hz	0 dB to 4.6 dB low
Z AXIS	500 Hz to 670 Hz	2.2 dB low

RATIONALE:

The EAs were mounted on the CMGs during qualification random vibration test. The levels required to qualify the EAs for a acceptance random vibration test in a standalone configuration were not reached in all three axes over the entire frequency band specified.

However, the EAs were driven to the required levels over typical resonant frequency bands for electronic circuit card assemblies. The lack of qualification margin over acceptance test levels in the specified frequency bands is considered minimal risk.

PG1–139:

ITEMS:

Multiplexer/Demultiplexer (MDM) Type 8, Part Number 8260525–905

SSP 41172 REQUIREMENT:

Paragraph 4.2.3.3B, Qualification Thermal Cycling Test, Test Levels and Duration.

The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

Qualification thermal cycle tests of the MDM Type 8 shall be performed at a maximum baseplate temperature of 140 degrees F. This is equal to the maximum predicted on orbit temperature.

RATIONALE:

The capability of the MDM to perform at a 140 degrees F temperature on orbit is verified by qualification thermal vacuum testing at 160 degrees F in accordance with SSP 41172. The MDMs Type 8 will exceed a temperature of 120 degrees F no more than 37 days during the first two years of the life of the vehicle. Additionally, the two flight MDMs on the PV Modules are fully redundant, so that the failure of either MDM results in no loss of functionality.

PG1–140:

ITEM:

Multiplexer/Demultiplexer (MDM) Type 8, Part Number 8260525–905

SSP 41172 REQUIREMENT:

Paragraph 5.1.3.3B, Acceptance Thermal Cycling Test, Test Levels and Duration

The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance during the cold portion of the cycle.

EXCEPTION:

Acceptance thermal cycle tests of MDMs Type 8 shall be performed at a maximum baseplate temperature of 120 degrees F. This is 20 degrees below the maximum predicted on orbit temperature.

RATIONALE:

The capability of the MDM to perform at a 140 degrees F temperature on orbit is verified by acceptance thermal vacuum testing at 140 degrees F in accordance with SSP 41172. These MDMs Type 8 will exceed a temperature of 120 degrees F no more than 37 days during the first two years of the life of the vehicle. The two flight MDMs on the PV Modules are fully redundant, so that the failure of either MDM results in no loss of functionality.

Under these circumstances, acceptance thermal cycle acceptance testing for Type 8 MDMs over a temperature range of -45 degrees F to 120 degrees F provides an adequate stress screen to ensure the workmanship of these flight MDMs. Such testing did exceed the minimum temperature sweep requirement of 100 degrees F and exceeds the required minimum thermal cycle test temperature of 32 degrees F. Any additional confidence provided by repeating the acceptance thermal cycle testing of the three flight units over an additional 20 degrees F on the thermal cycle high–end is unnecessary. This exception also applies to logistics spares which can be removed/replaced at the circuit card (Shop Replaceable Unit) level.

PG1–141:

ITEM:

SGANT Gimbal Motors Part Number 10033206–2, Serial Numbers 001 and 002

SSP 41172 REQUIREMENT:

Paragraphs 6.1.1A and 6.1.1B, Assembly/Components Protoflight Tests

A. For the thermal vacuum tests, the temperature extremes shall be 10 degrees F (5.6 degrees C) beyond the minimum and maximum predicted temperatures.

B. For the thermal cycling test, the temperature cycles shall be conducted at 10 degrees F (5.6 degrees C) beyond the maximum and minimum predicted temperatures.

EXCEPTION:

The SGANT Assembly Elevation (EL) and Cross Elevation (XEL) motors will be protoflight thermal tested to the following temperature levels:

Gimbal Components	Predicted flight Temperature	SSP 41172 Require–ment	Temp reached in QM gimbal assembly thermal test	Temp reached in retrofitted SGANT–QM Protoflight ORU thermal vacuum tests	Temps reached in SGANT_FM Protoflight ORU thermal vacuum tests
				Thermal Vacuum	Thermal Vacuum
XEL Motor	151	161	160	158	151
EL Motor	151	161	156	151	144

Thus, the gimbal motors do not meet the minimum required hot temperatures for protoflight thermal cycle and thermal vacuum tests.

RATIONALE:

The assembly of the two motors into the SGANT included the following sequence of tests: (1) tests of the individual motors; (2) tests at the Gimbal Assembly level; and (3) tests at the SGANT level.

The individual motors required burn–in testing at 248 degrees F for 96 hours and operational tests at 199 degrees F. Testing at the Gimbal Assembly level and SGANT level are as shown. Through the course of all testing, the SGANT FM XEL motor and SGANT QM EL motor were exposed to the maximum flight environment. Additionally, analysis shows torque and power consumption requirements will be met with margin at temperatures as high as 170 degrees F.

Thus, all motors were operated above 161 degrees F during motor tests, analysis shows they will perform well within requirements up to 170 degrees F, and, since all motors are identical, the motors can be certified for operation to 151 degrees F.

The SGANT Gimbal Motors Part Number 10033206–2, Serial Number 003 will be thermally tested in compliance to SSP 41172.

PG1–142:

ITEM:

Assembly Contingency Radio Frequency Group Part Number 830699–551, Serial Numbers 001, 002, and 003

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification) with the following exceptions.” This requires protoflight hardware thermal cycling temperature to be 10 degrees F above the maximum predicted temperature for all eight cycles as indicated in paragraph 6.1.1B.

EXCEPTION:

The ACRFG Assembly shall achieve a maximum temperature of 104 degrees F during protoflight thermal cycles 2 through 7 instead of the maximum protoflight temperature of 146 degrees F for the first thermal cycle test only. Any subsequent thermal cycle tests on these units shall require testing in full compliance with SSP 41172.

RATIONALE:

The testing performed on the ACRFG Assembly did provide needed localized thermal stresses to provide a workmanship screen. The ACRFG Assembly experienced the maximum protoflight temperature of 146 degrees F during Protoflight Thermal Cycle 1 and 8 for sufficient time to reach internal thermal equilibrium. The ACRFG Assembly also experienced the maximum protoflight temperature during the Protoflight Thermal Vacuum test. Full functional tests was performed and passed at the maximum temperature.

Two electronic components internal to the ACRFG, the Solid State Power Amplifier (SSPA) and the Low Noise Amplifier (LNA), also underwent component–level thermal testing. The SSPA experienced numerous component–level thermal tests over a temperature range of -41 degrees F to 146 degrees F. This testing was performed prior to installing the component in the higher level assembly to ensure RF performance would be met during the protoflight thermal test run at the ACRFG–level. The LNA also underwent component–level thermal testing over its predicted temperature range of -41 degrees F to 104 degrees F for identical reasons. Both passed component–level performance tests during and after these component–level tests.

The Acceptance Test Procedure for the ACRFG has been updated to correct this deficiency to ensure any future units tested per the procedure are done in accordance with SSP 41172.

PG1–143:

ITEM:

Assembly Contingency Radio Frequency Group Part Number 830699–551, Serial Numbers 001, 002, and 003

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)...” This requires that the protoflight hardware be turned off until the temperature stabilizes and then turned on during each cycle of the thermal cycling test as indicated in paragraph 4.2.3.3C.

EXCEPTION:

The ACRFG Assembly shall not be power cycled during Protoflight Thermal Cycles 2 through 7 for the first thermal cycle test only. Any subsequent thermal cycle tests on these units shall require testing in full compliance with SSP 41172.

RATIONALE:

The ACRFG Assembly 120 V dc operational and heater power were not power cycled during each protoflight thermal cycle. The operational power was turned off/on at minimum operating (–41 degrees F) and maximum operating (146 degrees F) temperature during Protoflight Thermal Cycle 1, and minimum operating temperature during Protoflight Thermal Cycle 8. The heater power was turned off/on at minimum operating temperature during Protoflight Thermal Cycle 1.

As indicated, the ACRFG Assembly was power cycled three times during its protoflight thermal cycling test. The ACRFG Assembly was also power cycled two times during its protoflight Thermal Vacuum test. The purpose of power cycling at temperature extremes is to screen for latent workmanship defects. The lack of power cycling during cycles 2 through 7 does not seriously degrade the effectiveness of the thermal cycle test. Nevertheless, the Acceptance Test Procedure for the ACRFG will be updated to correct this deficiency to ensure any future units tested per the procedure are done in accordance with SSP 41172.

PG1–144:

ITEM:

Assembly Contingency Radio Frequency Group Part Number 830699–551, Serial Number 001

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)...” This requires that the protoflight hardware be functionally tested at the maximum and minimum protoflight temperature during the first and last thermal cycles and after return of the component to ambient as indicated in paragraph 4.2.3.4.

EXCEPTION:

This ACRFG Assembly shall not be functionally tested at the minimum temperature during Protoflight Thermal Cycle 8 for its first thermal cycle test only. Any subsequent thermal cycle tests on this unit shall require testing in full compliance with SSP 41172.

RATIONALE:

This ACRFG Assembly passed functional testing at maximum and minimum protoflight temperatures during Protoflight Thermal Cycle 1 and at maximum temperature during Protoflight Thermal Cycle 8. Subsequently, this ACRFG Assembly did pass functional testing at maximum and minimum protoflight temperatures during the Protoflight Thermal Vacuum Test. Thus, the risk that this unit will not perform in on–orbit conditions is negligible. Additionally, remaining ACRFG Assemblies were functionally tested in compliance with the requirements and performed successfully. This provides confidence in the design of the ACRFG Assemblies.

PG1–145:

ITEM:

Assembly Contingency Baseband Signal Processor (ACBSP) Part Number 10033177–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)...” This requires that the protoflight hardware be turned off until the temperature stabilizes and then turned on during each cycle of the thermal cycling test as indicated in paragraph 4.2.3.3C.

EXCEPTION:

The ACBSP Assembly shall not be power cycled during Protoflight Thermal Cycles 2 through 7 for the first thermal cycle test only. Any subsequent thermal cycle tests on these units shall require testing in full compliance with SSP 41172.

RATIONALE:

The ACBSP Assembly 120 V dc operational power was not power cycled during each protoflight thermal cycle. The operational power was turned off/on at minimum nonoperating (–58 degrees F), minimum operating (–33 degrees F), and twice at maximum operating (138 degrees F) temperature during Protoflight Thermal Cycle 1, and minimum operating and maximum operating temperature during Protoflight Thermal Cycle 8.

As indicated, the ACBSP Assembly was power cycled six times during its protoflight thermal cycling test. The ACBSP Assembly was also power cycled twice during its protoflight thermal vacuum test. The purpose of power cycling at temperature extremes is to screen for latent workmanship defects. The lack of power cycling during cycles 2 through 7 does not seriously degrade the effectiveness of the thermal cycle test. Nevertheless, the Acceptance Test Procedure for the ACBSP will be updated to correct this deficiency to ensure any future units tested per the procedure are done in accordance with SSP 41172.

PG1–146:

ITEM:

Assembly Contingency Baseband Signal Processor (ACBSP) Part Number 10033177–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)..." This requires that the protoflight hardware be fine leak tested at an external test pressure of 0.001 Torr (0.133 Pa) or less, and the duration of the test be four hours as indicated in paragraph 4.2.11.3B.

EXCEPTION:

The ACBSP Assembly shall be fine leak tested using a contra–flow leak detection method with an external pressure less than 0.050 Torr and a minimum test duration of five minutes.

RATIONALE:

The contra–flow leak detection method was performed on the ACBSP Assembly (dimensions 14 inches x 8 inches x 10 inches) with a diffusion pump of capacity 90 liters per second configured with a bell jar (dimension 36 inches high x 24 inches diameter). The fine leak rate on the ACBSP Assembly was monitored using the commercial–off–the–shelf Varian 938–41 Porta–Test Leak Detector with a detection sensitivity of 2 X 10E–10 atm cc per second Helium and a response time of two seconds. In this configuration, a test duration of five minutes is sufficient time for the detector to determine that the fine leak rate is less than the component's allowable leak rate of 8 X 10E–05 atm cc per second Helium. In addition, during the lowering of the external pressure to less than 0.050 Torr, seal features are being exercised; in actuality, a fine leak has a much greater time (an additional 30 minutes minimum) to propagate to the detector to ensure an accurate rate is detected. Thus, the risk in granting an exception to external test pressure and test duration requirements for the configuration indicated is minimal when using the contra–flow leak detection method to accomplish fine leak testing.

PG1–147:

ITEM:

Fluid Line Anchor Patch, Part Numbers 1F98569, 1F98528, and 1F98570

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Random Vibration Test will not be performed on the Fluid Line Anchor Patch.

RATIONALE:

The design of the FLAP does not warrant a Qualification Random Vibration test to verify as the item does not have Random Vibration sensitive parts. All FLAPs will be installed after launch in the on–orbit environment; thus, the part is never exposed to the launch environment while in its operative configuration (attached to a fluid line). Additionally, the FLAPs will be securely stowed during launch to limit exposure.

PG1–148:

ITEM:

Fluid Line Anchor Patch, Part Numbers 1F98569, 1F98528, and 1F98570

SSP 41172 REQUIREMENT:

Paragraph 4.2.10, Pressure Test, Component Qualification.
Paragraph 4.2.10.3C, Test Levels, Ultimate Pressure. "Ultimate Pressure shall be as specified in SSP 30559, section 3." This requires that lines and fittings less than 1.5 inches have an ultimate pressure test performed to four times the maximum design pressure.

EXCEPTION:

The Fluid Line Anchor Patch shall have a qualification ultimate pressure test performed to two times the maximum design pressure.

RATIONALE:

Qualification Ultimate pressure testing was performed to two times its maximum design pressure instead of four times its maximum design pressure since the FLAP serves as a temporary installation onto a Fluid Line until a permanent repair can be made. Thus, the wear predicted on the unit would be less than that expected on a permanent line or structure. Additionally, the FLAP is unlike other lines, fittings, and flex hoses as it consists of a soft seal incorporated into a rigid body. Burst tests to verify design are performed to 1000 psi.

PG1–149:

ITEM:

Fluid Line Anchor Patch, Part Numbers 1F98569, 1F98528, and 1F98570

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.
Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

Paragraph 5.1.6, Pressure Test, Component Acceptance.
Paragraph 5.1.6.3, Test Levels. ALL REQUIREMENTS.

Paragraph 5.1.7, Leak Test, Component Acceptance.
Paragraph 5.1.7.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Thermal Vacuum, Acceptance Thermal Cycle, Acceptance Random Vibration, Acceptance Pressure, and Acceptance Leak Tests will not be performed on the Fluid Line Anchor Patch.

RATIONALE:

A FLAP provides rigid fluid line leak repair on Environmental Control and Life Support System and Thermal Control System fliuid lines. A FLAP serves as a temporary installation onto a Fluid Line until a permanent repair can be made. Inherent to the design of the FLAP is a soft seal. The FLAP seal is expended once attached to a fluid line; thus, the component is a single–use unit. Therefore, a flight–use FLAP cannot be acceptance tested in an operative configuration (attached to a fluid line). All flight–use FLAPs shall be accepted via inspection.

PG1–150:

ITEM:

Tracking and Data Relay Satellite System (TDRSS) Transponder (XPDR) Part Number 10039397–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)...” This requires that the protoflight hardware be turned off until the temperature stabilizes and then turned on during each cycle of the thermal cycling test as indicated in paragraph 4.2.3.3C.

EXCEPTION:

The TDRSS XPDR shall not be power cycled during Protoflight Thermal Cycles 2 through 7 for the first thermal cycle test only. Any subsequent thermal cycle tests on these units shall require testing in full compliance with SSP 41172.

RATIONALE:

The TDRSS XPDR 120 V dc operational power was not power cycled during each protoflight thermal cycle. The operational power was turned off/on at minimum nonoperating (–58 degrees F), minimum operating (–33 degrees F), and twice at maximum operating (138 degrees F) temperature during Protoflight Thermal Cycle 1, and minimum operating and twice at maximum operating temperature during Protoflight Thermal Cycle 8.

As indicated, the TDRSS XPDR was power cycled seven times during its protoflight thermal cycling test. The ACBSP Assembly was also power cycled twice during its protoflight thermal vacuum test. The purpose of power cycling at temperature extremes is to screen for latent workmanship defects. The lack of power cycling during cycles 2 through 7 does not seriously degrade the effectiveness of the thermal cycle test. Nevertheless, the Acceptance Test Procedure for the TDRSS XPDR will be updated to correct this deficiency to ensure any future units tested per the procedure are done in accordance with SSP 41172.

PG1–151:

ITEM:

Tracking and Data Relay Satellite System (TDRSS) Transponder (XPDR) Part Number 10039397–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)..." This requires that the protoflight hardware be fine leak tested at an external test pressure of 0.001 Torr (0.133 Pa) or less, and the duration of the test be four hours as indicated in paragraph 4.2.11.3B.

EXCEPTION:

The TDRSS XPDR shall be fine leak tested using a contra–flow leak detection method with an external pressure less than 0.050 Torr and a minimum test duration of five minutes.

RATIONALE:

The contra–flow leak detection method was performed on the TDRSS XPDR (dimensions 14 inches x 8 inches x 10 inches) with a diffusion pump of capacity 90 liters per second configured with a bell jar (dimension 36 inches high x 24 inches diameter). The fine leak rate on the TDRSS XPDR was monitored using the commercial–off–the–shelf Varian 938–41 Porta–Test Leak Detector with a detection sensitivity of 2 X 10E–10 atm cc per second Helium and a response time of two seconds. In this configuration, a test duration of five minutes is sufficient time for the detector to determine that the fine leak rate is less than the component's allowable leak rate of 8 X 10E–05 atm cc per second Helium. In addition, during the lowering of the external pressure to less than 0.050 Torr, seal features are being exercised; in actuality, a fine leak has a much greater time (an additional 30 minutes minimum) to propagate to the detector to ensure an accurate rate is detected. Thus, the risk in granting an exception to external test pressure and test duration requirements for the configuration indicated is minimal when using the contra–flow leak detection method to accomplish fine leak testing.

PG1–152:

ITEMS:

Capture Latch Assembly (CLA) Part Number 1F03095–1
Umbilical Mechanism Assembly (UMA), Active Half Part Number 1F05101–501
Umbilical Mechanism Assembly (UMA), Passive Half Part Number 1F05104–501
Motorized Bolt Assembly (MBA), Part Number 1F49180–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3C, Test Levels and Duration. A minimum of three temperature cycles shall be used.

EXCEPTION:

A minimum of one thermal cycle shall be used.

RATIONALE:

PG1–63 was granted to eliminate thermal vacuum qualification test for common hardware mechanical assemblies and replace with at least one thermal cycle test at ambient pressure on common hardware mechanism assemblies. The next lower level of moving mechanical assemblies which include the CLA, UMA, and MBA have the same justification. One thermal cycle is sufficient to determine thermal expansion and contraction anomalies. Since heaters are required to maintain internal Integrated Motor Controller Actuator (IMCA) temperatures above –45 degrees F, thermal vacuum testing is required to achieve adequate heat transfer on the component. The SSP 41172 requirement for the first qualification thermal vacuum cycle shall be followed. The CLA, UMA, and MBA internal electronic components (IMCAs and Bolt Motor Actuators) are independently certified in a thermal vacuum environment.

PG1–153:

ITEM:

Space to Ground Antenna (SGANT) Part Number 10033206–2

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests: When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification). This requires that the assembly be power cycled at the minimum and maximum temperature of each cycle of thermal cycling test.

EXCEPTION:

The SGANT assembly during the thermal cycle test was power cycled only during two cycles of the first thermal cycle test: The minimum and maximum extremes during thermal cycle 1 and the maximum extreme during thermal cycle 8. Any subsequent thermal cycle tests on this unit shall require testing in full compliance with SSP 41172.

RATIONALE:

As indicated, the SGANT was power cycled three times during its protoflight thermal cycle test. The SGANT was also power cycled twice during its Protoflight Thermal Vacuum test. The purpose of power cycling at temperature extremes is to screen for latent workmanship defects. The lack of power cycling during cycles 2 through 7 and at the minimum extreme of cycle 8 does not seriously degrade the effectiveness of the thermal cycle test. In total, the quality of the workmanship screen is adequate. Nevertheless, the General Requirements Specification and the Acceptance Test Procedure for the SGANT will be updated to correct this deficiency to ensure any future units tested per the procedure are done in accordance with SSP 41172.

PG1–154:

ITEM:

Space to Ground Antenna (SGANT) Part Number 10033206–2

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests: When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification) with the following exceptions.

Paragraph 6.1.1.B. For the thermal cycling test, the temperature cycles shall be conducted at 10 degrees F (5.6 degrees C) beyond the maximum and minimum predicted temperatures. The minimum number of cycles shall be 8.

EXCEPTION:

SGANT components did not meet the Protoflight Thermal Cycling requirement as indicated:

Main Reflector – Minimum predicted temperature: –116 degrees F;
Protoflight Thermal Cycle as–tested minimum temperature: –34 degrees F

Sub–Reflector – Minimum predicted temperature: –138 degrees F;
Protoflight Thermal Cycle as–tested minimum temperature: –34 degrees F

Tracking Modulator Driver – Maximum predicted temperature: 123 degrees F;
Protoflight Thermal Cycle as–tested maximum temperature: 127 degrees F

RATIONALE:

The Protoflight Thermal Cycle test of the SGANT Assembly was controlled by the on–orbit predicted temperatures of the internal SGANT Motor Drive Amplifier. The minimum predicted on–orbit temperature of the SGANT Motor Drive Amplifier is – 20 degrees F; the maximum predicted on–orbit temperature of the SGANT Motor Drive Amplifier is 115 degrees F. The Protoflight Thermal Cycle test temperature range of the SGANT Assembly was – 32 degrees F to 127 degrees F. During this test, the components indicated did not experience extreme temperatures as required by SSP 41172 Protoflight environmental tests; however, adequate stresses was applied during additional thermal vacuum and thermal cycles tests.

During the cold cycle of the SGANT Protoflight Thermal Vacuum test, the Main Reflector was tested to – 178 degrees F and the Sub–Reflector was tested to –193 degrees F. This exceeded the main reflector minimum predicted environment by 62 degrees F and the sub–reflector minimum predicted environment by 55 degrees F. This provided necessary thermal design margin. In addition, the reflectors do not contain any Electrical, Electronic, and Electromechanical (EEE) or moving parts.

During the SGANT build process, the Tracking Modulator Driver (TMD) is assembled into a RF Feed, and the RF Feed is then assembled into the SGANT. Prior to assembly of the RF Feed into the SGANT, the RF Feed is put through a single–cycle thermal test. During this RF Feed thermal test, the TMD was exposed to 147 degrees F that exceeds its maximum predicted environment by 24 degrees F. Also, during the SGANT Protoflight Thermal Vacuum Test, the TMD was exposed to 143 degrees F that exceeds its maximum predicted environment by 20 degrees F. Additionally, all internal EEE parts in the TMD were exposed to their screening requirements of –67 to 257 degrees F. Combined, the TMD has adequate thermal design margin.

Finally, both reflectors and the TMD experienced at least a 100–degree F temperature sweep in all thermal tests.

PG1–155:

ITEM:

Space to Ground Antenna (SGANT) Part Number 10033206–2

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests: When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification). This requires that the assembly be electically energized and monitored during the Protoflight acoustic test.

Paragraph 6.1.1.C. For the acoustic vibration qualification test, the test level shall be the maximum predicted flight level, but not less than a level derived from an acoustic environment of 141 dB overall, (whose spectrum is defined by NSTS 21000–IDD–ISS, paragraph 4.1.1.5). The duration of the test shall be limited to one minute.

EXCEPTION:

The SGANT assembly is exempt from being electrically energized and monitored during acoustic testing.

RATIONALE:

Individual SGANT electrical components (TRD and MDA) were electrically energized and monitored during component–level vibration testing at the following levels:

Tracking Modulator Driver – X– and Y–axis at 10.48 grms and Z–axis at 12.97 grms

Motor Drive Amplifier – X– and Y–axis at 9.28 grms and Z–axis at 12.07 grms

The Gimbal Assembly Motors and Encoders underwent vibration testing at the component level unenergized but did pass functional tests afterwards. The Motors experienced additional screening tests including DC resistance tests, Dielectric Withstanding Voltage Test, MIL–STD–202 compatible Thermal Cycle Tests, and Burn–In, while the Encoders, a circuit card made up of EEE parts, experienced an eight–cycle thermal test.

The Gimbal Assembly underwent component–level vibration testing above the minimum screening level and afterward passed functional tests.

The Gimbal Asembly was then assembled into the SGANT assembly and underwent acoustic vibration testing. The SGANT assembly did pass functional tests before and after.

Additionally, all thermal testing was performed on the SGANT assembly after the Gimbal Assembly underwent its vibration testing. These thermal tests were performed electrically energized and monitored.

As a whole, the level of testing is adequate as a workmanship screen to uncover intermittences.

PG1–156:

ITEM:

Integrated Motor Controller Actuator Part Number 1F03158

SSP 41172 REQUIREMENT:

Paragraph 5.1.8 Burn–In Test, Component Acceptance.

Paragraph 5.1.8.3C, Duration. For constant temperature burn–in (either at ambient or accelerated via elevated temperature), the total operating time shall be equivalent to 300 hours at ambient temperature.

EXCEPTION:

The constant temperature burn–in will be not less than 67 hours.

RATIONALE:

Sufficient screening has been performed to capture time/temperature related failure mechanisms including diffusion defects, dielectric strength, oxidation, and chemical contamination. This is based on the following:

(1) S–rated EEE piece–part level screening and burn–in
(a) All EEE parts are both functionally tested and tested at elevated temperature
(b) All parts are subjected to thermal shock and component–level burn–in

(2) IMCA Card level testing
(a) Functional testing was performed at –65 degrees F, 75 degrees F, and 160 degrees F
(b) IMCAs experienced an eight–cycle thermal cycle test from –65 degrees F to 160 degrees F; and

(3) Box level testing
(a) An ten–cycle environmental stress screen was performed from –24 degrees F to 128 degrees F with a ramp rate of 1.8 degrees F per minute
(b) The box experienced a three–axes random vibration test for 60 seconds
(c) The box experience an eight–cycle thermal vacuum test from –45 degrees F to 140 degrees F with a ramp rate of 1.8 degrees F per minute.

Finally, additional screening occurs during ORU testing, Segment testing, and MEIT that further reduces the likelihood of failures that could only be identified during additional burn–in testing.

PG1–157:

ITEM:

Integrated Motor Controller Actuator Part Number 1F03158

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Testing, Component Acceptance.
Paragraph 5.1.2.3B, Temperature. The component shall be at the maximum predicted temperature during the hot portion of the cycle and at the minimum predicted temperature during the cold portion of the cycle.

Paragraph 5.1.3, Thermal Cycle Testing, Component Acceptance.
Paragraph 5.1.3.3B, Temperature. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

The IMCA will be powered off at –34.5 degrees F.

RATIONALE:

Thermal survey data, using the internal IMCA RTD, confirms that, while the IMCA coldplate and the chamber environment reached the minimum predicted temperature of –45.0 degrees F, the IMCA was powered off 10.5 degrees F above the minimum predicted temperature at –34.5 degrees F. During the remaining transition to the minimum predicted temperature and internal thermal equilibrium, the unit will not be monitored for failures and intermittent performance. However, failures will be determined from functional testing after the required powered off dwell period during the first and last cycles. During the intermediate cycles (the second cycle through the seventh cycle), failures and intermittent performance will be screened during the cold start after the powered off dwell period and during the transition to the maximum predicted temperatures. Thus, the current testing of the IMCAs will discover any workmanship flaws.

PG1–158:

ITEM:

Integrated Motor Controller Actuator Part Number 1F03158

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Testing, Component Qualification.
Paragraph 4.2.2.3C, Duration. The dwell period shall be long enough for the component to reach internal thermal equilibrium for not less than one hour.

Paragraph 4.2.3, Thermal Cycle Testing, Component Qualification.
Paragraph 4.2.3.3C, Duration. Each cycle shall have a one hour minimum dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on.

EXCEPTION:

The dwell duration at the minimum qualification temperature shall be 40 minutes.

RATIONALE:

Internal RTD data was examined during the thermal survey of the IMCA acceptance test for both a four–hour dwell and a 40–minute dwell. The data shows that after both dwells, the RTD temperature readings are within 0.8 degrees F. This difference in the readings validates the assumption that internal thermal equilibrium has been achieved and the intent of the thermal dwell requirements in SSP 41172 is met.

PG1–159:

ITEM:

Integrated Motor Controller Actuator Part Number 1F03158

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Testing, Component Acceptance.
Paragraph 5.1.2.3C, Duration. The component shall undergo a dwell period of at least one hour or a time sufficient for the component to reach internal thermal equilibrium as established by qualification testing, whichever is greater, at both the high and low temperature extremes with power off and then turned on.

Paragraph 5.1.3, Thermal Cycle Testing, Component Acceptance.
Paragraph 5.1.3.3C, Duration. Each cycle shall have a one hour minimum dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on.

EXCEPTION:

The dwell duration at the minimum predicted temperature shall be 40 minutes.

RATIONALE:

Internal RTD data was examined during the thermal survey of the IMCA acceptance test for both a four–hour dwell and a 40–minute dwell. The data shows that after both dwells, the RTD temperature readings are within 0.8 degrees F. This difference in the readings validates the assumption that internal thermal equilibrium has been achieved and the intent of the thermal dwell requirements in SSP 41172 is met.

PG1–160:

ITEM:

Thermostat Box Assembly Part Numbers 1F80434–1 and 1F80435–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.

“When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)...” This requires Protoflight Thermal Cycling test in accordance with paragraph 4.2.3 as modified by paragraph 6.1.1B.

EXCEPTION:

The full Thermostat Box Assembly will not undergo a Protoflight Thermal Cycle test.

RATIONALE:

The Thermostat Box Assembly is a S1/P1 unique assembly that is used to house Thermostats (Part Number 1F97596) that control the Radiator Beam fluid line heaters. This housing is needed to protect the thermostats from the external environment and maintain a thermally conditioned environment via Heaters (MIL–R–39009C RER75 Style). All Thermostats are thermally screened at the component level in accordance with MIL–PRF–38534, Appendix K that includes 20 thermal cycles from –55 degrees C to 125 degrees C. Also, five samples from each Thermostat lot are subjected to a 100–cycle thermal shock test from –55 degrees C to 125 degrees C with no defects allowed. Thermal Vacuum testing will remain in the Protoflight Testing program.

Additional thermal cycling testing at the Assembly Level is not necessary as the Assembly has only four solder joints. Boeing Quality Assurance and DCMC visually inspect the solder joints. Crimped contacts and splices are not critically affected by Thermal Cycling.

Finally, Protoflight Thermal Vacuum testing at the Thermostat Box Assembly level will remain in the Protoflight Test program.

PG1–161:

ITEM:

Thermostat Box Assembly Part Numbers 1F80434–1 and 1F80435–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.

"When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same..." This requires Protoflight Burn–In testing for Electronics in accordance with paragraph 5.1.8.

EXCEPTION:

The electronics of the full Thermostat Box Assembly will undergo no less than 260 hours of Burn–In testing.

RATIONALE:

The Thermostat Box Assembly is a S1/P1 unique assembly that is used to house Thermostats (Part Number 1F97596) that control the Radiator Beam fluid line heaters. This housing is needed to protect the thermostats from the external environment and maintain a thermally conditioned environment via Heaters (MIL–R–39009C RER75 Style). Each Thermostat undergoes 240 hours of Burn–In testing at the component level in accordance with MIL–STD–883. During Protoflight testing, the full Thermostat Box Assembly will accrue the balance of the 300 hours. Since the Thermostats are "powered cycled" during the Functional Test, some Thermostats may not see 300 hours of operation. At worst case, every Thermostat will undergo an estimated 260 hours of operation. Since most life–cycle failures occur in the early stages of operation, this duration is sufficient to root out any defective components.

PG1–162:

ITEM:

Thermostat Box Assembly Part Numbers 1F80434–1 and 1F80435–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.

"When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)..." This requires Protoflight Pyrotechnic Shock Test in accordance with paragraph 4.2.7 as modified by paragraph 6.1.1D.

EXCEPTION:

The full Thermostat Box Assembly will not undergo a Protoflight Pyrotechnic Shock Test.

RATIONALE:

The Thermostat Box Assembly is a S1/P1 unique assembly that is used to house Thermostats (Part Number 1F97596) that control the Radiator Beam fluid line heaters. This housing is needed to protect the thermostats from the external environment and maintain a thermally conditioned environment via Heaters (MIL–R–39009C RER75 Style). The Thermostats used in the Thermostat Box Assembly are the same as Thermostats used in the LVS Radiator Assembly that successfully underwent Qualification Pyrotechnic Shock Testing. Since the Thermostat Box Assembly is mounted further from a Pyrotechnic event than the LVS Radiator Assembly, the Protoflight Pyrotechnic Shock levels experienced by the Thermostats in a Thermostat Box Assembly would be significantly less. The Thermostat design is fully qualified via the Qualification Pyrotechnic Shock Test of the LVS Radiator Assembly.

PG1–163:

ITEM:

Thermostat Box Assembly Part Numbers 1F80434–1 and 1F80435–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)...” This requires Protoflight Electromagnetic Interference (EMI)/Electromagnetic Compatibility (EMC) testing for Electronics in accordance with paragraph 4.2.12.

EXCEPTION:

The full Thermostat Box Assembly will not undergo a Protoflight Electromagnetic Interference (EMI)/Electromagnetic Compatibility (EMC) Test.

RATIONALE:

The Thermostat Box Assembly is a S1/P1 unique assembly that is used to house Thermostats (Part Number 1F97596) that control the Radiator Beam fluid line heaters. This housing is needed to protect the thermostats from the external environment and maintain a thermally conditioned environment via Heaters (MIL–R–39009C RER75 Style). Thermostats undergo Qualification EMC/EMI testing at its component level. There are not any other active components within the assembled Thermostat Box Assembly that could experience malfunction due to the EMI/EMC properties of the Thermostat. Therefore, the removal of a Protoflight EMI/EMC test is permitted.

PG1–164:

ITEM:

Space–to–Ground Transmitter Receiver Controller (SGTRC) Part Number 1003317–501 Serial Number 002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.

"When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)..." This requires that the protoflight hardware have a minimum one–hour dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on during each cycle of the thermal cycling test as indicated in paragraph 4.2.3.3C.

EXCEPTION:

The SGTRC was not power cycled during Protoflight Thermal Cycles 2 through 7

The SGTRC was not powered until internal thermal stabilization during the maximum temperature extreme of Protoflight Thermal Cycle 1 and minimum temperature extreme of Protoflight Thermal Cycle 8.

NOTE: Any subsequent thermal cycle tests on this unit shall require testing in full compliance with SSP 41172.

RATIONALE:

The SGTRC Serial Number 002 had been subjected to three thermal cycling sequences which accounts for 19 thermal cycles. Six cycles of the thermal cycle tests included power cycling. In addition to the Thermal Cycle tests, the Thermal Vacuum test and retest comprised two more cycles with power cycling. The cycles 1 and 8 of the thermal cycle tests met specific SSP 41172 requirements, where the ORU was soaked and dwelled at the extreme temperatures for needed durations with power cycling, prior to the onset of functional testing. Power cycling was in full compliance with SSP 41172 on the hot phase of Cycle 1 and the cold phase of Cycle 8, at which times the unit remained powered down for at least two hours. These two–hour periods with ORU powered off allowed the unit to discharge all internal circuitry. Also, on the cold phase of Cycle 1 and the hot phase of Cycle 8, the units were turned off for a duration of 30 to 60 seconds. This is adequate for circuitry discharge for the design of this unit.

The purpose of power cycling at temperature extremes is to screen for latent defects. The lack of power cycling during cycles 2 through 7 of the thermal cycle sequences, while preferred, did not seriously degrade the effectiveness of the thermal cycle test.

PG1–165:

ITEM:

Space–to–Ground Transmitter Receiver Controller (SGTRC) Part Number 1003317–501 Serial Number 002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.

"When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)..." This requires that the protoflight hardware have an extended dwell period during the thermal vacuum test of not less than 12 hours at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on as indicated in paragraph 4.2.2.3C.

EXCEPTION:

The SGTRC shall remain powered during the dwell period at the minimum temperature extreme of the Protoflight Thermal Vacuum Test. Any subsequent thermal vacuum tests on this unit shall require testing in full compliance with SSP 41172.

RATIONALE:

The SGTRC Serial Number 002 underwent two protoflight thermal vacuum tests and nineteen cycles of protoflight thermal cycle testing. The quality of the Protoflight Thermal Vacuum test was not compromised as this test was performed in series with the Protoflight Thermal Cycle test. In three Protoflight Thermal Cycling Test sequences, each time Cycle 8 included a minimum of two hours at the minimum temperature extreme during which the ORU is powered off. Thus, any material and workmanship defects could be detected during functional testing after completion of the Protoflight Thermal Cycle and Protoflight Thermal Vacuum Tests.

PG1–166:

ITEM:

Space–to–Ground Transmitter Receiver Controller (SGTRC) Part Number 1003317–501

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)..." This requires that the external test pressure of the protoflight hardware be 0.001 Torr or less during the fine leak test as indicated in paragraph 4.2.11.3B.

EXCEPTION:

The external test pressure shall be no greater than 0.005 Torr during the Protoflight Fine Leak Test.

RATIONALE:

Review of the test method and the unit performance during the fine leak environment test indicates sufficient seal integrity. The SGTRCs were filled with Helium and Nitrogen gases and placed in a Bell Jar. The pressure in the Bell Jar was reduced to the lowest possible level of 0.005 Torr via a vacuum pump for a duration of four hours. At 73 ± 1 degree F, the leak detector measured a fine leak rate of 6.0 x 10E–08 cubic cm per second, less than the specified value of 8 x 10E–05 cubic cm per second. All parameters were within the specified capabilities of the equipment used. Thus, the external test pressure level of 0.005 Torr is adequate for this hardware with the conventional test method used.

PG1–167:

ITEM:

Space–to–Ground Transmitter Receiver Controller (SGTRC) Part Number 1003317–501, Serial Numbers 001 and 002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification) with the following exceptions:

C. For the acoustic vibration qualification test, the test level shall be the maximum predicted flight level, but not less than a level derived from an acoustic environment of 141 dB overall. The duration of the test shall be limited to one minute.

EXCEPTION:

The SGTRC Serial Numbers 001 and 002 shall be tested to modified Z1 acoustic levels in the Y– and Z–axes as indicated:

Y–axis		Z–axis	
Frequency Range (Hz)	Power Spectral Density (g^2/Hz)	Frequency Range (Hz)	Power Spectral Density (g^2/Hz)
80–115	48 dB/Oct	20	0.01
115	0.04	80	0.04
130	0.21	569.9	0.04
150	0.21	700	0.11
170	0.04	840	0.11
170–225	–48 dB/Oct	2000	0.014
Composite = 3.0 grms		Composite = 9.6 grms	
Duration = 30 seconds		Duration = 60 seconds	

RATIONALE:

The SGTRC Serial Number 002 was exposed to three separate Protoflight Random Vibration tests. The duration of the first two tests was 60 seconds per axis for all three axes, and each axis was subjected to the same Power Spectral Density envelope. After rework of the SGTRC Serial Number 002 power supply, and in accordance with the recommendation of the Test and Verification Control Panel, the completion of Protoflight Random Vibration Testing will be performed on the Y– and Z–axes as indicated.

To qualify the SGTRC Serial Number 001 to the Z1 launch environment in case of a manifest change, this unit shall be retested to these identical levels.

SGTRC Serial Number 003 shall be tested to the full Z1 acoustic protoflight levels.

PG1–168:

ITEM:

Control Moment Gyroscope Electrical Assembly Part Number 5092021–9

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.

Paragraph 4.2.3.3, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The Control Moment Gyroscope Electrical Assembly shall be at the minimum acceptance operating temperature minus a margin of 5 degrees F during the cold portion of the qualification thermal cycle test.

The Control Moment Gyroscope Electrical Assembly shall be at the minimum acceptance non–operational temperature minus a margin of 13 degrees F during the cold portion of the qualification thermal cycle test.

RATIONALE:

The 20 degrees F qualification margin was demonstrated in the Control Moment Gyroscope Thermal Vacuum Test. During this test, the wall of the Electrical Assembly was at – 65 degrees F which demonstrated a 20 degree F non–operational margin prior to power on. The functional test was performed with the wall of the Electrical Assembly at – 15 degress F which demonstrated a 20 degree F operational margin at power on. Additionally, the unit was powered on at an Electrical Assembly wall temperature of – 65 F degrees. When powered on, all circuits except for the spin motor drive (which was not turned on) functioned normally. Under this powered on condition, at least 90 percent of the Electrical Assembly's functionality was energized. Thus, the electrical design of the Assembly is shown to be adequately qualified for flight thermal conditions.

PG1–169:

ITEM:

Linear Drive Unit Part Number D60699001–01

Load Transfer Unit Part Numbers D60695001–01, D60695001–02, D60695001–03, and D60695001–04

Roller Suspension Unit Part Number D60698000–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.

Paragraph 5.1.2.4, Supplementary Requirements. Functional tests shall be conducted at the maximum and minimum predicted temperature levels during the first and last operating cycles after the dwell and after return of the component to ambient temperature.

EXCEPTION:

The EVA–actuated mechanisms of the Linear Drive Unit, Load Transfer Unit, and Roller Suspension Unit shall not be exercised during the Acceptance Thermal Vacuum Test.

RATIONALE:

The EVA–actuated mechanisms for the flight LDUs, LTUs, and RSUs will be flight accepted via inspection for workmanship, tolerance analysis, and ambient test results.

Inspection of flight units for workmanship has been verified and documented. Flight Acceptance Tests at ambient pressure and temperature indicate nominal EVA mechanism operation with no binding or galling, increasing confidence that no workmanship–related defects exist. Tolerance analysis assures the design margins at temperature extremes. Additionally, Human Thermal Vacuum Tests on the Qualification units of the indicated EVA–actuated mechanisms were successful. These tests were performed at full Qualification temperature levels of –65 degrees F to 160 degrees F. Qualification units are identical to the flight units except for attachment locking inserts that are Human Thermal Vacuum–qualified using similar hardware.

The Program assumes little additional risk with this exception, as these EVA–actuated mechanisms are either one–time or contingency use.

Finally, operational workarounds are available if an EVA–actuated mechanism does not operate properly on–orbit. To prevent hardware damage, astronaut power tools could be torque–limited to less than the maximum operating torque of the EVA–actuated mechanism. These torques may be documented in the on–orbit procedures and checklists. Also, awaiting less extreme thermal conditions may eliminate any thermally induced bolt concerns.

PG1–170:

ITEM:

Amplifier ORU Assembly Part Number D60696200

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.3.4, Supplementary Requirements. ALL REQUIREMENTS.

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

Paragraph 4.2.12, Electromagnetic Compatibility Test, Component Qualification.
Paragraph 4.2.12.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.12.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Qualification Thermal Vacuum, Qualification Thermal Cycle, Qualification Random Vibration, and Qualification Electromagnetic Compatibility Tests will not be performed on the fully assembled Amplifier ORU Assembly.

RATIONALE:

The Amplifier ORU Assembly consist of two Space Station Buffer Amplifiers (SSBA), Resistance Temperature Detectors (RTD), heaters, and thermostats. The RTDs, heaters, and thermostats experienced a Qualification Test Program in accordance with the Electrical, Electronic, and Electromechanical (EEE) parts requirements of SSP 30312. Internal SSBAs experienced a joint NASA/CSA Qualification Test Program in accordance with the requirements of SPAR–SS–CAL–0519 to allow standalone use on the Space Station Remote Manipulator System. The qualification of the lower level components is adequate to encompass the qualification of the Amplifier ORU Assembly as the Amplifier ORU Assembly consists of these individual components that are bolted or glued to a flat radiator plate via approved specifications and structural adhesives methods.

PG1–171:

ITEM:

Amplifier ORU Assembly Part Number D60696200

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Random Vibration Test will not be performed on the fully assembled Amplifier ORU Assembly.

RATIONALE:

The Amplifier ORU Assembly consist of two Space Station Buffer Amplifiers (SSBA), Resistance Temperature Detectors (RTD), heaters, and thermostats. The RTDs, heaters, and thermostats experienced an Acceptance Random Vibration Test in accordance with the Electrical, Electronic, and Electromechanical (EEE) parts requirements of SSP 30312. Internal SSBAs experienced and passed an Acceptance Random Vibration Test in accordance with SPAR–SS–CAL–0519 which is both equivalent to SSP 41172 and approved by NASA to allow standalone use on the Space Station Remote Manipulator System. Additionally, analysis indicates that the Acceptance Random Vibration Tests on the internal SSBAs are adequate to support the MT launch environment. Finally, the Amplifier ORU Assembly workmanship is further verified by process control and inspections methods.

PG1–172:

ITEM:

HRS Radiator ORU Gear Brake Part Number 83–39512–109

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.

Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The Qualification Thermal Vacuum Test performed does not encompass the worst–case Acceptance Thermal Vacuum test by an average of 3.5 degrees F at minimum temperature and an average of 9.5 degrees F at maximum temperature.

RATIONALE:

The Thermal Vacuum Margins provided relative to the average temperatures during the Qualification Thermal Vacuum Test of the HRS Radiator ORU Gear Brake are as follows:

HRS Radiator ORU Gear Brake Thermal Vacuum Test Temperatures

HRS Radiator ORU Gear Brake Unit	Average Minimum Temperature (degrees F)	Average Maximum Temperature (degrees F)	Margin at Average Minimum Temperature (degrees F)	Margin at Average Maximum Temperature (degrees F)
SN 003	– 53	140	15.5	9
SN 004	– 64	153	4.5	– 4
SN 005	– 70	138	– 1.5	11
SN 006	– 60.5	139.5	8	9.5
SN 007	– 72	158.5	– 3.5	– 9.5
SN 008	– 39.5	134	29	15
Qualification Unit: Average Minimum Temperature: – 68.5 degrees F Average Maximum Temperature: 149 degrees F				

The Gear Brake Assembly performance margin is 33 percent for the Motor Drive and 29 percent for the EVA drive based on Flight # 4 Thermal Vacuum demonstration test at Plumbrook in September, 1999 and Gear Brake component test data at cold temperatures. On–orbit deployment torque requirements will be lower than 1–G testing performed due to friction on the panel roller bearings and test instrumentation. This will result in even high performance margin than demonstrated by ground testing. Considering both the available temperature and performance margins of the Gear Brake, the risk associated with the non–compliant temperature margin during Gear Brake thermal vacuum testing is mitigated.

The Gear Brake minimum on–orbit temperature is the nominal heater–controlled minimum temperature of – 40 degrees F. Based on test data from Flight # 4 demonstration test, the Gear Brake maximum on–orbit temperature is 104.2 degrees F. The Flight # 4 Thermal Vacuum testing successfully operated Gear Brake Unit Serial Number 007 at a peak maximum temperature of 124.2 degrees F and an average maximum temperature of 117.7 degrees F. Thus, in conjunction with the performed Qualification test, the Gear Brake is shown to have a robust design and not be sensitive to temperature extremes. As the Qualification program did test with adequate margin the Gear Brake beyond the on–orbit temperatures, and with the success of the Flight # 4 demonstration testing, the Qualification Thermal Vacuum testing is adequate.

PG1–173:

ITEM:

HRS Radiator ORU Gear Brake Part Number 83–39512–109

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The Qualification Thermal Cycle Test performed does not encompass the worst–case Acceptance Thermal Cycle test with a median delta of 4.5 degrees F at minimum temperature and a median delta of 10.5 degrees F at maximum temperature.

RATIONALE:

The Thermal Cycle Margins provided relative to the median temperatures during the Qualification Thermal Cycle Test of the HRS Radiator ORU Gear Brake are as follows:

HRS Radiator ORU Gear Brake Thermal Cycle Test Temperatures

HRS Radiator ORU Gear Brake Unit	Average Minimum Temperature (degrees F)	Average Maximum Temperature (degrees F)	Margin at Average Minimum Temperature (degrees F)	Margin at Average Maximum Temperature (degrees F)
SN 003	– 73	163.5	– 4.5	– 4.5
SN 004	– 68.5	169.5	0	– 10.5
SN 005	– 44	128	24.5	31
SN 006	– 51	136.5	17.5	22.5
SN 007	– 37.5	121	31	38
SN 008	– 50	133	18.5	26
Qualification Unit: Minimum Median Temperature: – 68.5 degrees F Maximum Median Temperature: 159 degrees F				

The internal Solenoid activated brake is the only electrical component in the Gear Brake Assembly. Thus, Thermal Cycle testing was required and performed during the Qualification and Acceptance Programs of the Gear Brake Assembly. However, a Qualification Thermal Vacuum test performed on the Gear Brake prior to needed design changes verified the performance of this Solenoid component to a minimum operational temperature of –147 degrees F. Therefore, the Solenoid has a verified margin of at least 74 degrees relative to the worst–case acceptance minimum median temperature experienced by Gear Brake Serial Number 003. This test also verified the performance of the Solenoid component to a maximum operational temperature of 200 degrees F. Again, the Solenoid has a verified margin of at least 30.5 degrees F relative to the worst–case acceptance maximum median temperature experienced by Gear Brake Serial Number 004.

The total duration of thermal cycles experienced by the Qualification HRS Radiator ORU Gear Brake Assembly is 37 cycles. These include 8 thermal cycles at acceptance levels, 26 thermal cycles at qualification levels, and an additional 3 cycles under vacuum conditions at qualification levels. Thus, when viewed in addition to a valid qualification test program for the Solenoid that has not changed configuration or undergone redesign, this singular electronic component is proven robust.

The Gear Brake minimum on–orbit temperature is the nominal heater–controlled minimum temperature of – 40 degrees F. Based on test data from Flight # 4 demonstration test, the Gear Brake maximum on–orbit temperature is 104.2 degrees F. The Flight # 4 Thermal Vacuum testing successfully operated Gear Brake Unit Serial Number 007 at a peak maximum temperature of 124.2 degrees F and an average maximum temperature of 117.7 degrees F. Thus, in conjunction with the performed Qualification tests, the Gear Brake is shown to have a robust design and not be sensitive to temperature extremes. As the Qualification program did test with adequate margin the Gear Brake beyond the on–orbit temperatures, and with the success of the Flight # 4 demonstration testing, the Qualification Thermal Cycle testing is adequate.

PG1–174:

ITEM:

HRS Radiator ORU Gear Brake Part Number 83–39512–109

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.

Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Random Vibration Test will not be performed on the HRS Radiator ORU Gear Brake.

RATIONALE:

Table 5–1, Note 7, requires components with close tolerances requiring precise adjustment or that cannot be inspected effectively undergo Acceptance Random Vibration tests. The Note does not apply to the Gear Brake since it is designed with sufficient backlash such that the gear center tolerances on installation in the housing do not require adjustments. The Gear Brake is inspected for minimum backlash to prevent binding at the operating temperatures. Also, Flight Gear Brakes have undergone Acceptance Thermal Vacuum and Thermal Cycle testing in excess of the predicted operating temperatures of –40 to 120 degrees F. The Gear Brakes met all performance requirements at these temperatures, which further demonstrate no close tolerances.

The Qualification Gear Brake is Random Vibration tested to 14.83 grms for 3 minutes. Thus, the design is qualified with an outstanding issue of the screening of the Flight Gear Brakes. The Gear Brake is not sensitive to Random Vibration since it is a mechanism assembled with no interior fasteners that can loosen and all interior piece parts are confined within its housing which allow no significant movement.

In addition, each Radiator ORU Gear Brake undergoes Acceptance Acoustic testing. Three Gear Brakes assembled into HRS Radiator ORUs that will be installed on ITS P1 underwent an acoustic test at 141 dB for 60 seconds during the ITS S1 Structural Test Article Acoustic Test. The remainder are tested at the Radiator ORU level at 138 dB for 60 seconds. The input to the Gear Brake during the 141 dB ITS S1 Structural Test Article Acoustic Test is the maximum predicted flight level for the ORU. Successful performance during post–test functional testing of the Radiator ORUs used as part of the ITS S1 Structural Test Article Acoustic Test will indicate that these are successfully screened to the maximum flight level.

PG1–175:

ITEM:

High Rate Frame Multiplexer Part Number 10033171–1 Serial Numbers Q001 and 001

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…" This requires that the external test pressure be 0.001 Torr (0.133 Pa) or less, and the duration of the test be four hours (for equipment that is operational in orbit for more than one day) for the protoflight fine–leak test as indicated in 4.2.11.3B.

EXCEPTION:

The HRFM protoflight fine leak test will be performed with an external test pressure of less than or equal to 0.050 Torr and a minimum test duration of 5 minutes.

RATIONALE:

The HRFM is a hermetically–sealed unit that is internally pressurized to 8 psia. The ORU will be located inside the USL that has a 14.7 psia ambient pressure. The ORU has underwent pressure testing with its internal pressure varying from 2 to 20 psia. Thus, as the HRFM functionality is not compromised if its internal components are exposed to the ambient pressure of the USL via a leak, the risk associated with the fine–leak test on the HRFM as indicated is minimal.

PG1–176:

ITEM:

High Rate Frame Multiplexer Part Number 10033171–1 Serial Numbers Q001 and 001

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…" This requires that the protoflight hardware be turned off until the temperature stabilizes and then turned on during each cycle of the thermal cycling test as indicated in paragraph 4.2.3.3C.

EXCEPTION:

The HRFM will be exempt from performing a power–cycled test during Protoflight Thermal Cycles 2 through 7.

RATIONALE:

The HRFM 120 Vdc operational power was not power cycled during each cycle of the protoflight thermal cycle test. The operational power was turned off/on at minimum nonoperating (–58 degrees F), minimum operating (18 degrees F), and maximum (124 degrees F) temperature during Protoflight Thermal Cycle 1; and at minimum operating and at maximum temperature during Protoflight Thermal Cycle 8.

The purpose of power cycling at temperature extremes is to screen for latent workmanship defects. The lack of power cycling during cycles 2 through 7 does not seriously degrade the effectiveness of the thermal cycle test nor does it warrant retesting of the hardware. Nevertheless, the Acceptance Test Procedure for the HFRM will be updated to correct this deficiency to ensure any future units tested per the procedure are done in accordance with SSP 41172.

PG1–177:

ITEM:

High Rate Modem Part Number 10033169–1 Serial Numbers Q001, and –001

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…" This requires that the external test pressure be 0.001 Torr (0.133 Pa) or less, and the duration of the test be four hours (for equipment that is operational in orbit for more than one day) for the protoflight fine–leak test as indicated in 4.2.11.3B.

EXCEPTION:

The HRM protoflight fine leak test will be performed with an external test pressure of less than or equal to 0.050 Torr and a minimum test duration of 5 minutes.

RATIONALE:

The HRM is a hermetically–sealed unit that is internally pressurized to 8 psia. The ORU will be located inside the USL that has a 14.7 psia ambient pressure. The ORU has underwent pressure testing with its internal pressure varying from 2 to 20 psia. Thus, as the HRM functionality is not compromised if its internal components are exposed to the ambient pressure of the USL via a leak, the risk associated with the fine–leak test on the HRM as indicated is minimal.

PG1–178:

ITEM:

High Rate Modem Part Number 10033169–1 Serial Numbers Q001, and –001

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…" This requires that the protoflight hardware be turned off until the temperature stabilizes and then turned on during each cycle of the thermal cycling test as indicated in paragraph 4.2.3.3C.

EXCEPTION:

The HRM will be exempt from performing a power–cycled test during Protoflight Thermal Cycles 2 through 7.

RATIONALE:

The HRM 120 Vdc operational power was not power cycled during each cycle of the protoflight thermal cycle test. The operational power was turned off/on at minimum nonoperating (–58 degrees F), minimum operating (18 degrees F), and maximum (124 degrees F) temperature during Protoflight Thermal Cycle 1; and at minimum operating and at maximum temperature during Protoflight Thermal Cycle 8.

The purpose of power cycling at temperature extremes is to screen for latent workmanship defects. The lack of power cycling during cycles 2 through 7 does not seriously degrade the effectiveness of the thermal cycle test nor does it warrant retesting of the hardware. Nevertheless, the Acceptance Test Procedure for the HFRM will be updated to correct this deficiency to ensure any future units tested per the procedure are done in accordance with SSP 41172.

PG1–179:

ITEM:

Video Baseband Signal Processor, Part Number 10033175–501, Serial Number 002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…" This requires that the protoflight hardware be turned off until the temperature stabilizes and then turned on during each cycle of the thermal cycling test as indicated in paragraph 4.2.3.3C.

EXCEPTION:

The Video Baseband Signal Processor (Serial Number 002) is exempt from being power cycled during Protoflight Thermal Cycles 2 through 7.

RATIONALE:

Power Cycling was completed on cycles 1 and 8 during each of 2 VBSP Protoflight Thermal Cycle tests. As the VBSP had been subjected to two thermal cycle tests (16 thermal cycles) due to power supply rework and retest, it has been subjected to 8 power cycles during 4 thermal cycles at the hot and cold extremes.

During the power supply rework and retest, VBSP Serial Number 001 was tested per SSP 41172 thermal cycle requirements with power cycling on each cycle.

The purpose of power cycling at temperature extremes is to screen for latent workmanship defects. The lack of power cycling during cycles 2 through 7 does not seriously degrade the effectiveness of the thermal cycle test nor does it warrant retesting of the hardware. Nevertheless, the Acceptance Test Procedure for the VBSP will be updated to correct this deficiency to ensure any future units tested per the procedure are done in accordance with SSP 41172.

PG1–180:

ITEM:

Control Moment Gyroscope Electronic Assembly Part Number 5080097–9

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3B, Test Levels and Duration. The component shall be at the maximum predicted temperature during the hot portion of the cycle and at the minimum predicted temperature during the cold portion of the cycle.

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance..
Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

The CMG EA maximum acceptance tests temperature is 165 degrees F.

RATIONALE:

The CMG EA qualification unit has experienced thermal testing at a maximum temperature of 188 degrees F. Flight CMG EA units have experienced thermal testing at a maximum temperature of 165 degrees F; however, this is below the maximum predicted temperature of 168 degrees F.

The maximum predicted temperature of 168 degrees F is at extreme high beta angles for XPOP between 5A –12A and for XVV after Assembly Complete and do not occur for Mean Propulsive Attitudes. High beta angles occur for less than 10 days per year. For 168 degrees F (3 degrees F above acceptance test), the EEE parts for CMG EA are below the applicable derating limits. The risk of a workmanship defect in a flight EA which passed undetected during thermal acceptance testing at 165 degrees F yet failing on–orbit due to a temperature of 168 degrees F is considered remote.

PG1–181:

ITEM:

Thermostat Housing Assembly, Part Number 5205253–9

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:
Paragraph 4.2.5.3B. Acceptance Test levels and spectrum plus test tolerances.

EXCEPTION:

The qualification levels indicated do not envelope acceptance levels plus test tolerances below 85 Hertz in all three axes and above 1800 Hertz in the X– and Z–axes for the Thermostat Housing Assembly.

RATIONALE:

The qualification levels indicated do not envelope acceptance levels plus test tolerances below 85 Hertz in all three axes and above 1800 Hertz in the X– and Z–axes for the Thermostat Housing Assembly. At all other frequencies, the qualification levels are 6 dB above acceptance levels. At 85 Hertz there is 1.0 dB margin and at 20 Hz, there is 0 dB margin in all three axes; at 2000 Hertz, the margin between qualification and acceptance is only 1.5 dB in the X– and Z–axes (it should be 3 dB). The Thermostat Housing Assembly was powered and monitored during the test.

According to Allied Signal, the primary Thermostat Housing Assembly component resonances are in the range of 900 to 1200 Hertz. During qualification random vibration testing, the constituent electronic components did experience qualification levels 6 dB above acceptance in this range. Since there are no component resonances below 100 Hz, the vibration loading on the electronics should be low. The acceptance vibration test of the flight unit was like a protoflight test in this frequency range (below 100 Hz). Additionally, all the electronic components inside the Thermostat Housing Assembly were screened to MIL–STD–202 levels (15 g peak–to–peak, 20–minute sinusoidal sweep from 10 to 2000 Hz).

Since the CMG mechanical assembly qualification vibration, upon which the Thermostat Housing Assembly levels are based, is an order of magnitude above the Z1 maximum flight levels, the structural integrity of the Thermostat Housing Assembly is not an issue since the Thermostat Housing Assembly will see much lower levels during launch than when it was vibration–tested.

PG1–182:

ITEM:

Trailing Umbilical System Reel Assembly Part Number 1F45002–501

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.

Paragraph 4.2.2.3C, Test Level and Duration. A minimum of three temperature cycles shall be used.

EXCEPTION:

The TUS Reel Assembly will undergo a minimum of two complete qualification thermal vacuum cycles.

RATIONALE:

The TUS Reel Assembly electronic component hardware (Integrated Motor Controller Actuators, and Video Signal Converter) are independently qualified at the component level. The TUS Reel Assembly does not contain any additional vacuum sensitive components. Performing additional qualification thermal vacuum testing at the TUS Reel Assembly level of assembly would unnecessarily increase test cost. One thermal cycle is sufficient to determine thermal expansion/contraction anomalies. This approach is consistent with PG1–63 and PG1–152 where exceptions were granted to eliminate thermal vacuum qualification test for common hardware mechanical assemblies and replaced with at least one thermal cycle test at ambient pressure. The Qualification TUS Reel Assembly will actually be subjected to one thermal cycle at ambient pressure in the Crew Interfaces and EVA Actuated thermal extreme test and two complete thermal vacuum cycles in two different configurations (“direction reversal” and “reel to reel”) with two different functional tests. Any temperature induced tolerance failures should manifest themselves in the first cycle.

PG1–183:

ITEM:

Linear Drive Unit Part Number D60699401

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C)(minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The Cam–reset function of the Linear Drive Unit was not exercised during the Qualification Thermal Vacuum Test at the maximum temperature.

RATIONALE:

The LDU Cam–reset Function has been partially qualified at hot extreme temperatures via testing of the redundant Changeover Function. The Changeover Function, which was properly qualification–tested at hot extreme, exercises all motions of the Cam–reset Function except for one of three stages (stage I), and the sequence of the stages are different. Stage I moves the primary cam an unknown number of degrees (because of launch vibrations) of which an additional 3.129 degrees is beyond the normal engage position reached during the Changeover Function, to a hard stop. The entire Cam–reset Function has been qualification–tested in the complete and correct sequence at cold and ambient temperatures. Although the Cam–reset Function sequence and Stage I were not qualification tested at the extreme hot temperature, the LDU flight unit has been fully acceptance tested using the complete Cam–reset Function sequence, up to the hot extreme acceptance levels.

The Cam–reset Function is used once post–launch as part of the launch–to–activation process. It would only be used subsequently in the event a cam was left in an unknown state following loss of power or data during a cam operation, or following a prime/redundant drive wheel changeover required due to a drive power or data failure during translation. The routine changeover of LDU drive wheels prior to translation, which is nominally being planned to extend wheel life, does not require use of the cam–reset function.

The likelihood of not completing the cam–reset Stage I is low because the drive wheel positioning motions are complete prior to the last 3.129 degrees of primary cam rotation and the mechanism is not under mechanical load. The Changeover Function demonstrated via test that the mechanism will not bind during hot all the way up to, but not including the last 3.129 degrees of primary cam motion. However, moving the cam beyond the normal engagement demonstrated during test by the Changeover Function to the hard stop is not actually necessary to permit continued MT operations, so a failure to complete this final stage would have no adverse effect on mission success.

PG1–184:

ITEM:

Ammonia Tank ORU Part Number 1F96463
Nitrogen Tank ORU Part Number 1F96464
Pump Module ORU Part Number 1F96462

SSP 41172 REQUIREMENT:

Paragraph 5.1.5, Acoustic Vibration Test, Component Acceptance.
Paragraph 5.1.5.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance acoustic tests will not be performed on the Ammonia Tank ORU, the Nitrogen Tank ORU, and the Pump Module ORU.

RATIONALE:

Acceptance level acoustic testing was performed on the Qualification Ammonia Tank ORU, the Qualification Nitrogen Tank ORU, and the Qualification Pump Module ORU. The excitation levels imparted to the ORUs were closely matched to the test results exhibited during the S1 STA acoustic testing at the –6dB test run. ORU test results indicated the level of excitation of the critical components of the ORUs were well below the 6.1 grms minimum screening level for workmanship. Ammonia tank levels ranged from 0.503 grms to 3.851 grms (specification levels range from 6.302 grms to 17.778 grms). Nitrogen tank levels ranged from 0.052 grms to 5.262 grms (specification levels range from 3.563 grms to 17.766 grms). Pump Module levels included 0.092 grms for the flowwater (spec level is 8.097 grms), 0.249 grms for the accumulator (spec level is 5.818 grms), and 0.255 grms for the Pump and Control Valve Package (spec level is 4.910 grms). Only the current limiter that is mounted directly to the Micrometeroid Orbital Debris Shield experienced a maximum level of 10.621 grms (spec level is 16.012 grms).

Investigations into the methods employed for detecting workmanship defects showed that existing methods such as proof and leak testing for welds and tubes, torque verification using torque striping by Quality Assurance for mechanical and "blind" electrical fasteners, and continuous monitoring during component vibration tests were better suited to identify defects. Critical component electrical and mechanical connections on the ORUs were determined to be easily accessible by technician and inspection personnel. Lastly, the flight configuration does not allow for visual inspection of internal ORU components due to the Micrometeroid Orbital Debris Shields in place. A reliance on functional testing is required post–acoustic test, and these functional tests are currently performed prior to crating and shipping the ORU. In short, workmanship and material defects of the Ammonia Tank ORU, the Nitrogen Tank ORU, and the Pump Module ORU are better detected in other tests and inspections than an acceptance level acoustic test.

PG1–185:

ITEM:

Electronics Control Unit Marotta Part Number 236805–9001

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. Component random vibration test levels and spectrums shall envelope:
A. The maximum predicted flight level and spectrum minus 6dB, but not less than a level derived from a 135 dB overall acoustic environment.

EXCEPTION:

Acceptance Random Vibration testing performed on the ECUs shall not envelope the maximum predicted test levels and spectrum minus 6dB at all frequencies in all three axes. Specifically, the noted amplitude deficiencies for the following frequency ranges for each axis shall be permitted:

Axis	Freq. Range (Hz)	Deficiency from required level(dB) (worst case @peak)
X	50–55	1 dB
X	190–230	10 dB
Y	130–270	15 dB
Z	180–220	6 dB
Z	320–380	6 dB

RATIONALE:

The ECUs have been subjected to numerous Qualification and Acceptance tests at the ECU and higher levels of assembly and have never experienced a workmanship failure. The ECUs have been re–Qualification tested to levels that envelop the maximum flight environment. The ECUs were powered on and monitored during the ECU Acceptance, Qualification, and Delta–Qualification Random Vibration testing. Each ECU goes through acceptance vibration testing at 8.12 grms, sufficient to screen out unacceptable units, and then undergo a thorough electrical functional test. The ECUs are tested above the required minimum screening level of 6.1 grms in all frequencies.

Prior to the Heat Exchanger ORU Qualification Random Vibration test exceedances, the ECUs had been Qualified for the maximum predicted flight environment (equivalent to 144 dB). Even though the ECUs were not powered on during Heat Exchanger ORU Acceptance Random Vibration tests, they are subjected to at least the maximum flight level minus 3 dB (see SSCN 1000) and they have all passed their Heat Exchanger Acceptance Random Vibration post–functional tests. Therefore, not performing ECU component acceptance vibration testing enveloping the maximum predicted flight level minus 6 dB in all axes at all frequencies is considered minimal risk.

PG1–186:

ITEM:

Secondary Power Distribution Assembly Doors Part Number 1F77021

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Thermal Vacuum test shall not be performed for SPDA doors.

RATIONALE:

Qualification Thermal Vacuum testing is primarily performed to assure the hardware design can pass the acceptance thermal cycle testing. The SPDA doors' function is not affected by vacuum and thermal extremes as (1) both the door and their tracks are aluminum with identical coefficients of thermal expansion, and (2) the clearances between the doors and the tracks are relatively large.

The doors, which protect RPCM ORUs, slide in tracks with side rollers to expose the ORUs for removal and replacement. The clearance between the door and the tracks is 0.120 inch nominal (0.100 inch worst case). The clearance between the door and the rollers is 0.250 inch (0.180 inch worst case). There are no lubricants on any surfaces, as the friction is limited to normal contact forces only. Refer to the drawing inspection and tolerance analysis in MDC99H0708 for a more detailed description.

The pip pin used to retain the door during launch is installed in a standard size hole for this type of hardware. This pin is not considered part of the moving mechanical assembly.

PG1–187:

ITEM:

Capture Latch Assembly	Part Numbers	1F67908 Serial Numbers 01–04
		1F95819 Serial Number 02
		1F70143 Serial Numbers 01–02
		1F70147 Serial Number 01
		1F95898 Serial Number 01, and
		1F95896 Serial Number 01

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.4, Supplementary Requirements. Functional tests shall be conducted at the maximum and minimum predicted temperature levels during the first and last operating cycles after the dwell and after return of the component to ambient temperature. NOTE: This requires that Mechanical tests include application of torque, load, and motion as appropriate as indicated in paragraph 5.1.1.2 Test Description, Functional Test, Component Acceptance.

EXCEPTION:

The stated Capture Latch Assembly units were not functionally tested through the manual EVA Drive during acceptance thermal vacuum testing.

RATIONALE:

The CLA is a moving mechanical assembly that captures structural elements on the ISS. During thermal acceptance test the stated CLAs were driven only by the IMCA and not through the manual EVA drive. Visual indication that the EVA Drive is turning during IMCA driven operations at environmental conditions was achieved. The CLA, including the manual EVA Drive mechanism, is adequately screened for workmanship by successful completion of Acceptance Testing. Test data shows that all CLA units have acceptable variation between ambient EVA Drive torque and ambient/thermal IMCA torque for configurations of similar preload classes (2500, 4700, or 7500 lbs). All ambient EVA Drive torques (30 in–lb @ 3400 lbs, 55 in–lb @ 5700 lbs, 108 in–lb @ 8200 lbs) were less than the 143 in–lb maximum and all ambient/thermal IMCA torques were less than the 8 in–lb maximum regardless of preload for latching and unlatching. In addition, it was shown that only the 7500–lb preload configuration (single Module–to–Truss Segment Attach System unit) experiences torque increase as a function of temperature decrease. As a result, the Module–to–Truss Segment Attach System unit was successfully protoflight tested (latching and unlatching via EVA Drive) at the –55 degrees F cold thermal environment.

Additionally, it should be noted that four CLA units (1F95921–1 Serial Number 01 and 1F70147–1 Serial Numbers 02–04) included and successfully passed (less than 130 in–lb) EVA Drive actuation during thermal extreme exposure. The two units yet to finish build (1F95921–501 Serial Numbers 01–02, S5/P5, 2500–lb non–IMCA configurations) will include EVA Drive actuation during thermal testing.

PG1–188:

Integrated Motor Controller Actuator Part Numbers 1F03158–505, 1F03158–513, and 1F03158–515

SSP 41172 REQUIREMENT:

Paragraph 5.1.4.3, Random Vibration Test, Component Acceptance. Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 5.1.4.3A, Test Levels and Duration. The maximum predicted flight level and spectrum minus 6 dB, but not less than a level derived from a 135dB overall acoustic environment (whose spectrum is defined by NSTS 210000–IDD–ISS, paragraph 4.1.1.5)

EXCEPTION:

The acceptance random vibration spectrum for the Integrated Motor Controller Actuator shall be as follows:

Frequency (Hertz)	Minimum Power Spectrum Density q^2/Hz
20	0.01
60	0.12
400	0.12
2000	0.007
Overall	9.3 grms

This acceptance test spectrum does not envelope the maximum predicted flight level minus 6 dB in all axes at all frequencies as required by SSP 41172. The maximum predicted flight level minus 6 dB is determined from protoflight random vibration testing of the Module–to–Truss Structure Attach System (see table below for axes, frequencies, and exceedances where the maximum predicted flight level minus 6 dB criteria exceeded the IMCA acceptance test level).

RATIONALE:

During the Module–to–Truss Structure Attach System Protoflight Random Vibration Test, flight–level exceedances over the IMCA component–level acceptance levels were noted at the following discrete frequencies:

Axis	Frequency (Hertz)	Flight IMCA Acceptance Level (g^2/Hz)	MTSAS Capture Latch IMCA Acceptance Level (Max Flight Level –6 dB) (g^2/Hz)	Exceedances (dB)
X–axis	125	0.12	3.77	14.9
Y–axis	70	0.12	0.35	4.6
Y–axis	95	0.12	0.38	5.0
Y–axis	145	0.12	0.25	3.2
Y–axis	1700	0.006	0.02	6.0
Z–axis	70	0.12	0.21	2.5

Requalification for these exceedances was performed for the IMCA which envelopes this Module–to–Truss Structure Attach System Capture Latch IMCAs maximum predicted flight environment; however, no additional acceptance testing is planned.

The identified IMCAs are to be used as–is for Module–to–Truss Structure Attach System Capture Latch applications. The Module–to–Truss Structure Attach System Capture Latch has redundant IMCAs (primary and secondary) and an EVA drive feature to manually operate the latch. There have been no IMCA failures noted after IMCA acceptance testing or during operation at the next higher level of assembly. Thus, the current IMCA screening level is adequate for Module–to–Truss Structure Attach System Capture Latch applications and no further acceptance testing is required.

PG1–189:

ITEM:

HRS Radiator ORU Gear Brake Part Number 0080–0039–4

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.4, Supplemental Requirements. Functional tests shall be conducted, as a minimum, at the maximum operating temperature plus 20 degrees F (11.1 degrees C) and at the minimum operating temperature minus 20 degrees F (11.1 degrees C) during the first and last operating cycle after the dwell and after return of the component to ambient temperature.

EXCEPTION:

The Gear Brake was not functionally tested at the hot temperature extreme of the third thermal cycle.

RATIONALE:

The EVA and IMCA drive functional testing was performed during the Delta Qualification Thermal Vacuum tests. These functional tests were completed in the first two cycles at 5700 in–lbs in order to complete the four SSP 41172 required functional tests (completed prior to the third temperature cycle). In addition, two functional tests were performed at the cold temperature extreme of the third thermal cycle. These functional tests were of the manual and the IMCA drive of the Gear Brake to a 7210 in–lb output torque (deployment torque required to deploy the HRS Radiator ORU during the second Thermal Vacuum Qualification test at Plumbrook, 1997). These 7210 in–lb functional tests were intentionally done on the last cold dwell since the cold temperature functional is the design driver. The hardware developer's concern at the time was to complete the SSP 41172 required Thermal Vacuum functional tests prior to starting the 7210 in–lb functional tests and life cycle functional tests. The 7210 in–lb functional tests were thought to be excessive and not well within the Gear Brake assembly's design capability. The lack of a functional test on the final thermal cycle hot temperature is not a concern since the cold functional test is the design driver for the Gear Brake where it is least efficient. The Plum Brook Thermal Vacuum test of Flight HRS Radiator #4 has a functional test (both EVA and IMCA drive) performed at both hot and cold temperature extremes.

PG1–190:

ITEM:

HRS Radiator ORU Gear Brake Part Number 0080–0039–4

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. The pressure shall be reduced from atmospheric to below 0.0001 Torr.

EXCEPTION:

The chamber pressure was not maintained below 0.0001 Torr during the temperature transitions.

RATIONALE:

The Gear Brake was transitioned between temperature extremes with 5 psi of gaseous Nitrogen in the chamber. The chamber was evacuated below 1.0E–04 Torr prior to performing the Gear Brake functional tests during the temperature dwells. The Gear Brake could not be ramped to the temperature extremes with a vacuum in the chamber. The three rotating shafts of the test setup (EVA and Motor inputs, output) allowed extensive heat leaks to and from the Gear Brake. Convective heating and cooling was required to overcome these heat leaks since Radiative heating and cooling was not adequate to reasonably drive the Gear Brake to the required temperature extremes. Since the Gear Brake is primarily a mechanism with a simple electrical solenoid, vacuum during temperature transitions is not mandatory. The only potential vacuum sensitive hardware is the Integrated Motor Controller Assembly (IMCA) which received a SSP 41172–compliant thermal vacuum qualification test at the component level of assembly. There are no other heat dissipating electrical components in the Gear Brake which are stressed by a thermal vacuum environment. Since the Gear Brake is functioned at the temperature extremes under Thermal Vacuum conditions, performing transitions without the required vacuum is not an issue.

PG1–191:

ITEM:

HRS Radiator ORU Gear Brake Part Number 0080–0039–4

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. The pressure shall be reduced from atmospheric to below 0.0001 Torr.

EXCEPTION:

The chamber pressure was not maintained below 0.0001 Torr during the temperature transitions.

RATIONALE:

The Gear Brake was transitioned between temperature extremes with 5 psi of gaseous Nitrogen in the chamber. The chamber was evacuated below 1.0E–04 Torr prior to performing the Gear Brake functional tests during the temperature dwells. The Gear Brake could not be ramped to the temperature extremes with a vacuum in the chamber. The three rotating shafts of the test setup (EVA and Motor inputs, output) allowed extensive heat leaks to and from the Gear Brake. Convective heating and cooling was required to overcome these heat leaks since Radiative heating and cooling was not adequate to reasonably drive the Gear Brake to the required temperature extremes. Since the Gear Brake is primarily a mechanism with a simple electrical solenoid, vacuum during temperature transitions is not mandatory. The only potential vacuum sensitive hardware is the Integrated Motor Controller Assembly (IMCA) which received a SSP 41172–compliant thermal vacuum acceptance test at the component level of assembly. There are no other heat dissipating electrical components in the Gear Brake which are stressed by a thermal vacuum environment. Since the Gear Brake is functioned at the temperature extremes under Thermal Vacuum conditions, performing transitions without the required vacuum is not an issue.

PG1–192:

ITEM:

NH3 Tank Part Number 1F40057

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Description and Alternatives. ALL REQUIREMENTS.
Paragraph 4.2.11.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

A SSP 41172, Revision T, Method VI leak test for a duration of 30 minutes was performed in lieu of a Revision T, Method II Fine Leak Test.

RATIONALE:

The specified maximum allowable leak rate requirement documented in the Boeing NH3 Tank Specification 1F40057 is 1E–05 scc per sec Helium. However, this leakage rate is valid under Maximum Operating Pressure conditions of 500 psig. As leakage testing was performed at 14.7 psig, it is necessary to validate the leakage rate against a pressure–corrected requirement.

Assuming molecular flow through available leak paths, a pressure–correction factor of 34 (500 psig/14.7 psig) was applied to the specified maximum allowable leak rate. The result is a pressure–corrected maximum allowable leak rate of 3E–07 scc per sec Helium [(1E–05/34) scc per sec Helium]. The released Qualification Test Procedure 98–70293 under which the Leak Test was performed applied a more–stringent maximum allowable leak rate requirement of 1E–07 scc per sec Helium.

For the Qualification Leakage Test, a Mass Spectrometer Leak Detector was calibrated with an external leak source of 2E–10 scc per sec Helium both prior to and after the leak test. Both the NH3 and N2 sides of the tank were leak and pressure tested, and all welds were dye–penetrant and ultrasonic inspected. The performed test duration of 30 minutes is sufficient to check leakage across the tank welds as there are no permeable seals. The measured leakage rate during the Qualification Leak Testing of the NH3 tanks was 2E–09 scc per sec Helium. As this is below the pressure–corrected maximum allowable leak rate of 3E–07 scc per sec Helium and the more–stringent maximum allowable leak rate of 1E–07 scc per sec Helium in the Qualification Test Procedure, the leakage rate of the NH3 Tank during the Qualification Leak Test program is deemed acceptable.

Additionally, successful Pressure Cycle, Proof Pressure, and Burst Tests were also completed on the Qualification test article.

Finally, the NH3 Tank Assembly was installed in the EAS Protoflight Acoustic Test article prior to acoustic testing. An integrated end item vacuum chamber leak test in accordance with SSP 41172, Method II, was conducted after the acoustic test to ensure the integrity of the fully assembled EAS. The measured leakage rate was 2.5E–04 scc per sec Helium, after stability of the Mass Spectrometer Leak Detector was reached over three consecutive readings in 5–minute intervals. This meets the assembly–level requirement to not exceed 6.8E–04 scc per sec Helium (RJ00342, paragraph 3.2.1.1.3).

Thus, based on accumulated test data, the NH3 Tank is acceptable.

PG1–193:

ITEM:

NH3 Tank Part Number 1F40057 Serial Number 009

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Description and Alternatives. ALL REQUIREMENTS.
Paragraph 5.1.7.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

A SSP 41172, Revision T, Method VI leak test for a duration of 30 minutes was performed in lieu of a Revision T, Method II Fine Leak Test.

RATIONALE:

The specified maximum allowable leak rate requirement documented in the Boeing NH3 Tank Specification 1F40057 is 1E–05 scc per sec Helium. However, this leakage rate is valid under Maximum Operating Pressure conditions of 500 psig. As leakage testing was performed at 14.7 psig, it is necessary to validate the leakage rate against a pressure–corrected requirement.

Assuming molecular flow through available leak paths, a pressure–correction factor of 34 (500 psig/ 14.7 psig) is applied to the specified maximum allowable leak rate. The result is a pressure–corrected maximum allowable leak rate of 3E–07 scc per sec Helium [(1E–05/34) scc per sec Helium]. The released Acceptance Test Procedure AT 2351330 under which the Leak Test was performed did incorporate this pressure–corrected maximum allowable leak rate.

For the Acceptance Leakage Test, a Mass Spectrometer Leak Detector was calibrated with an external leak source of 2E–10 scc per sec Helium both prior to and after the leak test. Both the NH3 and N2 sides of the tank were leak and pressure tested, and all welds were dye–penetrant and ultrasonic inspected. The performed test duration of 30 minutes is sufficient to check leakage across the tank welds as there are no permeable seals. The measured leakage rate during the Acceptance Leak Testing of the NH3 Tank Serial Number 009 was 2.3E–08 scc per sec Helium. As this is below the pressure–corrected maximum allowable leak rate of 3E–07 scc per sec Helium, the leakage rate of the indicated NH3 Tank during the Acceptance Leak Test program is deemed acceptable.

Additionally, a Proof Pressure Test was also completed on this unit.

Finally, the NH3 Tank was installed in the EAS Protoflight Acoustic Test article prior to acoustic testing. An integrated end item vacuum chamber leak test in accordance with SSP 41172, Method II, was conducted after the acoustic test to ensure the integrity of the fully assembled EAS. The measured leakage rate was 2.5E–04 scc per sec Helium, after stability of the Mass Spectrometer Leak Detector was reached over three consecutive readings in 5–minute intervals. This meets the assembly–level requirement to not exceed 6.8E–04 scc per sec Helium (RJ00342, paragraph 3.2.1.1.3).

Thus, based on all accumulated test data, the indicated NH3 Tank is acceptable.

PG1–194:

ITEM:

Solar Array Rotary Joint Drive/Lock Assembly Part Number 5847010

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The EVA Follower Arm Deployment Mechanism of the Solar Array Rotary Joint Drive/Lock Assembly will undergo a Qualification Thermal Extreme Test at ambient pressure in lieu of a Qualification Thermal Vacuum test.

RATIONALE:

This mechanism is an EVA–deployable mechanical assembly. The Follower Arm Deployment has been tested successfully at ambient temperature and pressure. Future tests will be at thermal extremes with qualification margin at ambient pressure as there are no vacuum–sensitive components in this assembly.

The design details of the moving parts involved are:

Lock bolt is MoS2 coated Ti thru Stainless Steel locking insert

Draw bolt is MoS2 coated 15–5 PH with Braycote thru nitrided 15–5 PH follower block.

Rotation bolt is MoS2 coated Stainless Steel thru Stainless Steel bushing.

Mounting bolts are A286 w/MoS2 thru silver plated A286 nutplates.

Gears are 303 Stainless Steel with no coating and no load.

Clearances between rotating parts are 0.003–0.007 inches. Thermal contractions reduce clearances less than 0.001 inch.

All like–material parts (primarily stainless) are lubricated to preclude binding during movement. All lubricants are stable in vacuum. All combinations of lubricants are stable, both at ambient pressure and in vacuum conditions. There is no interaction between the Braycote and either of the dry film lubricants. Only the gears are unlubricated; as they have no load on them, there is no binding. Thus, the risk to perform a Thermal Extreme test in lieu of a Thermal Vacuum test to qualify and verify the SARJ DLA Follower Arm Bearing EVA Deployable Mechanism is minimal.

PG1–195:

ITEM:

Solar Array Rotary Joint Drive/Lock Assembly Part Number 5847010

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The EVA Follower Arm Deployment Mechanism of the Solar Array Rotary Joint Drive/Lock Assembly will undergo an Acceptance Thermal Extreme Test at ambient pressure in lieu of an Acceptance Thermal Vacuum test.

RATIONALE:

This mechanism is an EVA–deployable mechanical assembly. The Follower Arm Deployment has been tested successfully at ambient temperature and pressure. Future tests will be at thermal extremes at ambient pressure as there are no vacuum–sensitive components in this assembly.

The design details of the moving parts involved are:

Lock bolt is MoS2 coated Ti thru Stainless Steel locking insert

Draw bolt is MoS2 coated 15–5 PH with Braycote thru nitrided 15–5 PH follower block.

Rotation bolt is MoS2 coated Stainless Steel thru Stainless Steel bushing.

Mounting bolts are A286 w/MoS2 thru silver plated A286 nutplates.

Gears are 303 Stainless Steel with no coating and no load.

Clearances between rotating parts are 0.003–0.007 inches. Thermal contractions reduce clearances less than 0.001 inch.

All like–material parts (primarily stainless) are lubricated to preclude binding during movement. All lubricants are stable in vacuum. All combinations of lubricants are stable, both at ambient pressure and in vacuum conditions. There is no interaction between the Braycote and either of the dry film lubricants. Only the gears are unlubricated; as they have no load on them, there is no binding. Thus, the risk to perform a Thermal Extreme test in lieu of a Thermal Vacuum test to qualify and verify the SARJ DLA Follower Arm Bearing EVA Deployable Mechanism is minimal.

PG1–196:

ITEM:

Pump and Control Valve Package (PCVP) Firmware Controller Part Number SV819401

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:
Paragraph 4.2.5.3B. Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

PCVP Firmware Controller qualification random vibration test level and spectrum does not envelope the acceptance test level and spectrum plus test tolerances at all frequencies in all axes for Flight Firmware Controller Serial Number 0003 internal to the PCVP.

RATIONALE:

PCVP Firmware Controller Serial Number 0003 received a stand–alone workmanship vibration test in all three axes at the minimum workmanship screening level and spectrum defined by SSP 41172, Figure 5–2. There was no formal qualification random vibration test at the PCVP Firmware Controller level of assembly; the formal qualification random vibration test of the Firmware Controller occurred at the PCVP assembly level. The PCVP assembly level qualification and acceptance random vibration spectrums are notched in all three axes in order to preclude unnecessary damage to Firmware Controller circuit card assemblies (CCA) and PCVP bearings. All notches are compliant with Program notching criteria; however, since formal qualification of the Firmware Controller for random vibration occurred at the PCVP assembly level and the qualification spectrum was notched in all three axes, the formal qualification random vibration test level at the input to the Firmware Controller did not envelope the acceptance level plus test tolerances for the stand–alone Serial Number 0003 acceptance test at all frequencies in all three axes. All other flight PCVP Firmware Controllers receive acceptance random vibration workmanship screening as part of the complete PCVP assembly and do not receive a stand–alone vibration test; therefore, all other flight Firmware Controllers are compliant with SSP 41172.

The critical axis for the Firmware Controller is the Z–axis since it is perpendicular to the plane of all the internal CCAs. For the Z–axis, the frequencies where the qualification test do not envelope Firmware Controller Serial Number 0003 acceptance level plus test tolerances are in the frequency range from 360 to 630 Hertz.

While the input to the Firmware Controller does not envelope Serial Number 0003 stand–alone acceptance levels plus test tolerance in the above frequency range during PCVP assembly level Z–axis testing, cross–axis input to the Firmware Controller (Z–axis) during assembly level Y–axis testing did exceed minimum workmanship levels over most of the notched frequency range; hence, some margin exists in the notched frequency range.

In addition, an acceptance random vibration test was performed on the stand–alone qualification Firmware Controller to minimum workmanship levels for one minute per axis. While this still does not demonstrate the required qualification margin on spectral amplitude or duration for Flight Firmware Controller Serial Number 0003 internal to the PCVP, it does ensure no less than 0 dB margin (i.e., there are no negative margins) for all frequencies in all axes. Also, the formal Qualification Random Vibration test performed on the PCVP assembly was performed for 6 minutes per axis which provides additional test duration margin for the assembly and demonstrates additional fatigue life.

Performing additional qualification testing on the Firmware Controller qualification unit in a stand–alone manner is not deemed warranted given the minimal degree of technical risk. Performing additional PCVP assembly level qualification testing without notches to establish required margins for Firmware Controller Serial Number 0003 would likely result in unnecessary damage to the PCVP and is not justified. The risk of accepting Firmware Controller Serial Number 0003 as–is is considered minimal.

PG1–197:

ITEM:

External Video Switch (VSW), Part Number 10033180–501 Serial Numbers 001–004

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…” This requires that the protoflight hardware be turned off until the temperature stabilizes and then turned on during each cycle of the thermal cycling test as indicated in paragraph 4.2.3.3C.

EXCEPTION:

The VSW Assembly were not power cycled during protoflight Thermal Cycles 2 through 7. Also, the VSW shall not be power cycled during Cycle 8 Hot plateau. Any subsequent thermal cycle tests on these units shall require testing in full compliance with SSP 41172.

RATIONALE:

While power cycling at each temperature plateau is desired, the units received a sufficient number of power–on start–ups to verify circuit integrity. Furthermore, each unit was powered on at temperature colder than the specified lower operating temperature, which provides a more rigorous self–induced stress.

As indicated, each VSW (all Serial Numbers) was power cycled 3 times during its protoflight thermal cycling test:

Thermal Cycle 1:

On at 73 degree F, off at –30 degree F (prior to cold non–operating temperature dwell), on at –55 degree F (cold start at non–operating temperature),

Off at 130 degree F (prior to hot non–operating temperature dwell at 170 degree F), on at 130 degree F (prior to operating temperature stabilization and functional tests), then to ambient (to begin next cycle).

Thermal Cycles 2–7:

VSW was on continuously for 6 temperature cycles (from ambient (73 degree F) to –30 degree F, to 130 degree F and back to ambient).

Thermal Cycle 8:

From ambient to –30 degree F, off at –30 degree F, on at –30 degree F,

Transition to hot operating temperature (stabilize, dwell and functional tests, off at 73 degree F (end)).

VSW Serial Numbers 001–003 were also power cycled 3 times during flight thermal cycle retest (repeated Thermal Cycles 1, 2 and 8 as shown above).

The purpose of power cycling at temperature extremes is to screen for latent workmanship defects. The lack of power cycling during cycles 2 through 7 does not seriously degrade the effectiveness of the thermal cycle test. Nevertheless, the Acceptance Test Procedure for the VSW will be updated to correct this deficiency to ensure any future units tested per the procedure are done in accordance with SSP 41172.

PG1–198:

ITEM:

External Video Switch (VSW), Part Number 10033180–501 Serial Numbers 001–004

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…” This requires that the protoflight hardware be powered cycled during thermal vacuum testing as described in paragraph 4.2.2.2.

EXCEPTION:

The VSW Assembly was not power cycled during the Thermal Vacuum Test. Any subsequent thermal vacuum tests on these units shall require testing in full compliance with SSP 41172.

RATIONALE:

While power cycling at each temperature plateau is desired, the units received a sufficient number of power–on start–ups to verify circuit integrity. Furthermore, each unit was powered on at a temperature colder than the specified lower operating temperature, which provides a more rigorous self–induced stress.

For all VSW Serial Numbers the operational power was turned off/on at non–operating (–55 degree F), and at maximum operational temperature (130 degree F) during protoflight Thermal Cycle 1; and off/on at minimum operating temperature (–30 degree F) during protoflight Thermal Cycle 8. Serial Numbers 001–003 were power cycled an additional three times, during the post power supply rework retest (in the same manner as in the initial Thermal Cycle test).

As indicated, VSW Serial Numbers 001–003 were power cycled six times (during its protoflight thermal cycling test and retest). Serial Number 004 was powered off/on three times as described above (no retest was necessary, as assembly occurred after power supply rework), and was sufficient for workmanship screening.

The purpose of power cycling at temperature extremes is to screen for latent workmanship defects. The lack of power cycling during Thermal Vacuum Test does not seriously degrade the effectiveness of the Thermal Vacuum Test.

Also, for Serial Number 001–003, Thermal Cycle retest occurred after the Thermal Vacuum Test adding to confidence in workmanship.

Nevertheless, the Acceptance Test Procedure for the VSW will be updated to correct this deficiency to ensure any future units tested per the procedure are done in accordance with SSP 41172.

PG1–199:

ITEM:

External Video Switch (VSW), Part Number 10033180–501 Serial Number 002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)...” This requires that the protoflight hardware be fine leak tested at an external test pressure of 0.001 Torr (0.133 Pa) or less, and the duration of the test be four hours as indicated in 4.2.11.3B.

EXCEPTION:

VSW Serial Number 002 shall be fine leak tested using a mass spectrometer leak detector in contra–flow configuration with an external pressure less than 0.01 Torr and a duration of approximately five minutes. Any subsequent Fine Leak tests on these units shall require testing in full compliance with SSP 41172.

RATIONALE:

A mass spectrometer leak detector in contra–flow configuration used for fine leak testing of VSW Serial Number 002 does not need an external pressure of 0.001 Torr to work with the required sensitivity of better than 4E–05 scc per sec Helium.

The higher external pressure (less than 0.01 Torr, exact value not documented) does not affect the validity of the test result.

The VSW allowable maximum leakage rate is 8E–05 scc per sec Helium (based on 90 percent Nitrogen/10 percent Helium mix). The actual leakage rate recorded for VSW Serial Number 002 was 9.2E–06 scc per sec Helium at 10 percent of Helium in the fill gas inside the unit.

The reading obtained by the leak detector needs to be corrected to simulate 100 percent of Helium, i.e., multiplying by a ratio of 100 percent/10 percent. The calculated leakage rate for the unit if it contained 100 percent of Helium would be 9.2E–05 scc per sec Helium.

With an additional “safety factor” of 10 applied due to uncertainties such as the shortened duration of test (5 minutes vs. required 4 hours) and uncertain sensitivity of the leak detector, the final calculated leakage rate would be 9.2E–04 scc per sec Helium. With the leakage rate of 9.2E–04 scc per sec Helium or approximately 1.0E–03 of GN2, on–orbit pressure vs. time was calculated.

The results indicate that VSW Serial Number 002 pressure will be maintained above the corona inception point for at least 15 years, even if the leakage rate measurements are in error by an order of magnitude.

PG1–200:

ITEM:

External Video Switch (VSW), Part Number 10033180–501 Serial Numbers 001, 003, and 004.

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in 4.2 for component qualification)...” This requires that the protoflight hardware be fine leak tested at an external test pressure of 0.001 Torr (0.133 Pa) or less, and the duration of the test be four hours as indicated in paragraph 4.2.11.3B.

EXCEPTION:

VSW Serial Number 001, Serial Number 003, and Serial Number 004 shall be fine leak tested using a mass spectrometer leak detector in contra–flow configuration with an external pressure less than 0.005 Torr and a duration of 4 hours. Any subsequent Fine Leak tests on these units shall require testing in full compliance with SSP 41172.

RATIONALE:

A mass spectrometer leak detector in contra–flow configuration used for fine leak testing of VSW Serial Number 001, Serial Number 003, and Serial Number 004 does not need an external pressure of 0.001 Torr to work with the required sensitivity of better than 4E–05 scc per sec Helium. The higher external pressure (less than 0.005 Torr) does not affect the validity of the test result.

The VSW allowable maximum leakage rate is 8E–05 scc per sec Helium (based on 90 percent Nitrogen/10 percent Helium mix). The actual leakage rate recorded for VSW was: 3.2E–07 scc per sec Helium for Serial Number 001, 2.5E–08 scc per sec Helium for Serial Number 003, and 3.0E–07 scc per sec Helium for Serial Number 004, all at 10 percent of Helium in the fill gas inside the unit.

The reading obtained by the leak detector needs to be corrected to simulate 100 percent of Helium, i.e., multiplying by a ratio of 100 percent/10 percent. The calculated leakage rate for the units if it contained 100 percent of Helium would be 3.2E–06 scc per sec Helium for Serial Number 001, 2.5E–07 scc per sec Helium for Serial Number 003, and 3.0E–06 scc per sec Helium for Serial Number 004.

With a "safety factor" of 2 applied due to uncertain sensitivity of the leak detector, the final calculated leakage rate would be 6.4E–06 scc per sec Helium for Serial Number 001, 5.0E–07 scc per sec Helium for Serial Number 003, and 6.0E–06 scc per sec Helium for Serial Number 004. These values are less than an allowable maximum leakage rate of 8E–05 scc per sec Helium.

PG1–201:

ITEM:

Rate Gyro Assembly Part Number GG9534AC01

SSP 41172 REQUIREMENT:

Paragraph 4.2.11 Leakage Test, Component Qualification.
Paragraph 4.2.11.3B, Test Levels and Duration (Method II – Fine Leak Test). The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

For SSP 41172, Revision T, Method II leak testing of the Rate Gyro Assembly, the external test pressure shall be 0.010 Torr or less and the test duration shall be long enough for the measured leak rate to stabilize over a minimum 5–minute period within +/– 10 percent.

RATIONALE:

The Rate Gyro Assembly was Qualification tested in accordance with Honeywell Specification ENV–1010. Internal pressure of the 90 percent Nitrogen/10 percent Helium gas mixture in the RGA was 16.0 +/– 0.1 psia. External pressure in the vacuum chamber/bell jar was 10E–03 Torr or less. The vacuum chamber/bell jar evacuation time was approximately 20 minutes (based upon the logbook entries), and time of the RGA under external pressure less than 75E–03 Torr was approximately 10 minutes (based upon the logbook entries). The mass spectrometer leak detector sensitivity is specified by the manufacturer to be 6E–11 scc per sec Helium. When the leak rate reading remains stable for 5 minutes to within +/– 10 percent, the reading is recorded.

The higher external pressure does not affect the validity of the test result. When the Veeco MS–170 Mass Spectrometer Leak Detector is utilized, the test chamber pressure is specified in order to maintain the leak rate sensor at or below its operational pressure of 5E–04 Torr ("fine–leak" mode). The Veeco MS–170 is calibrated in the "fine–leak" mode and therefore, data is only collected when the Veeco remains in this mode. As long as the sensor pressure is maintained at or below 5E–04 Torr, the result of the leakage rate test is not affected by a higher test chamber pressure.

The RGA Specification has a leak rate of less than or equal to 1.3E–05 scc per sec Helium. The highest leak rate for the RGA was recorded on the ATP data sheet was Serial Number 002 at 5.5E–06 scc per sec Helium. The RGAs are purged and pressurized with nitrogen for flight. The equivalent leak rate for nitrogen is (5.5E–06 x 1.12) or 6.16E–06 scc per sec Nitrogen. This leak rate was multiplied by 10 to provide additional safety margin and the on orbit pressure versus time was calculated. The results show that the RGA life is greater than 15 years even if the leak rate results are in error by an order of magnitude. The ATP value was used over the qualification value since the ATP had the worse case value.

This exception is only applicable when using a Veeco MS–170 Mass Spectrometer Leak Detector. Leak testing using any other mass spectrometer leak detector shall be in compliance with SSP 41172.

PG1–202:

ITEM:

Rate Gyro Assembly Part Number GG9534AC02

SSP 41172 REQUIREMENT:

Paragraph 5.1.7 Leakage Test, Component Acceptance.
Paragraph 5.1.7.3B, Test Levels and Duration (Method II – Fine Leak Test). The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

For SSP 41172, Revision T, Method II leak testing of the Rate Gyro Assembly, the external test pressure shall be 0.010 Torr or less and the test duration shall be long enough for the measured leak rate to stabilize over a minimum 5–minute period within +/– 10 percent.

RATIONALE:

The Rate Gyro Assembly was Acceptance tested in accordance with Honeywell Specification ENV–1010. Internal pressure of the 90 percent Nitrogen/10 percent Helium gas mixture in the RGA was 16.0 +/– 0.1 psia. External pressure in the vacuum chamber/bell jar was 10E–03 Torr or less. The vacuum chamber/bell jar evacuation time was approximately 20 minutes (based upon the logbook entries), and time of the RGA under external pressure less than 75E–03 Torr was approximately 10 minutes (based upon the logbook entries). The mass spectrometer leak detector sensitivity is specified by the manufacturer to be 6E–11 scc per sec Helium. When the leak rate reading remains stable for 5 minutes to within +/– 10 percent, the reading is recorded.

The higher external pressure does not affect the validity of the test result. When the Veeco MS–170 Mass Spectrometer Leak Detector is utilized, the test chamber pressure is specified in order to maintain the leak rate sensor at or below its operational pressure of 5E–04 Torr (“fine–leak” mode). The Veeco MS–170 is calibrated in the “fine–leak” mode and therefore, data is only collected when the Veeco remains in this mode. As long as the sensor pressure is maintained at or below 5E–04 Torr, the result of the leakage rate test is not affected by a higher test chamber pressure.

The RGA Specification has a leak rate of less than or equal to 1.3E–05 scc per sec Helium. The highest leak rate for the RGA was recorded on the ATP data sheet was Serial Number 002 at 5.5E–06 scc per sec Helium. The RGAs are purged and pressurized with nitrogen for flight. The equivalent leak rate for nitrogen is (5.5E–06 x 1.12) or 6.16E–06 scc per sec Nitrogen. This leak rate was multiplied by 10 to provide additional safety margin and the on orbit pressure versus time was calculated. The results show that the RGA life is greater than 15 years even if the leak rate results are in error by an order of magnitude.

This exception is only applicable when using a Veeco MS–170 Mass Spectrometer Leak Detector. Leak testing using any other mass spectrometer leak detector shall be in compliance with SSP 41172.

PG1–203:

ITEM:

P4 Truss Segment Part Number R083500 including the following qualified in the P3/P4 acoustic test:
Alpha Joint Interface Structure On–Orbit Strut Part Number R081799–1
Rocketdyne Truss Attachment System Soft Dock Assembly Part Number R078857

P3 Truss Segment Part Number 1F83000 including the following qualified in the P3/P4 acoustic test
Solar Alpha Rotary Joint EVA Removable Strut Part Number 1F26604
EVA Diagonals Part Numbers 1F38649 and 1F26475
Tether Shuttle Stop Part Number 1F83123
Payload Attach System/Unpressurized Logistics Carrier Attach System Assembly Hinge Part Number 1F67209
Solar Alpha Rotary Joint Multilayer Insulation Clamp Part Number 1F83212
Segment to Segment Attach System Ready to Latch Indicator Part Number 1F70572
Keel Nut Brake Part Number 1F83308

SSP 41172 REQUIREMENT:

Paragraph 4.4.3, Acoustic Vibration Test, Component Qualification.
Paragraph 4.4.3.3, Test Levels And Duration. Exposure test time shall be at least three times the expected flight exposure time to the maximum flight environment or three times the acceptance test duration if that is greater but not less than three minutes.

EXCEPTION:

The duration of the Qualification Acoustic Test of the P3/P4 Structural Test Article including the lower–level assembly hardware listed above shall be 60 seconds.

RATIONALE:

The P3/P4 Structural Test Article acoustic test article consists of the P3 Structural Test Article, a protoflight P4 Integrated Equipment Assembly structural framework, and simulated (mass, dynamic, or acoustic) ORUs/components.

None of the components are ORUs so, they need to be qualified for two missions only, which is 60 seconds. The ORUs on the Solar Alpha Rotary Joint and the Payload Attach System have been qualified by component–level random vibration tests; therefore, the P3/P4 acoustic test serves as a test to acquire the component interface vibration levels for validating ORU/component qualification random vibration test environments. A one–minute test duration is sufficient to gather data to validate ORU/component qualification random vibration environments. NASA JSC, Boeing Huntington Beach, and Boeing Canoga Park personnel reached consensus in reducing the required test duration to one minute, as it is sufficient to achieve all of the P3/P4 acoustic vibration test objectives and does not result in unnecessary life expenditure of the S6 Integrated Equipment Assembly flight hardware present on the test article.

PG1–204:

ITEM:

Solar Alpha Rotary Joint Trundle Bearing Assemblies Part Number 5846485 Serial Numbers 1027 and 1028

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.3, Test Duration and Levels. ALL REQUIREMENTS.
Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Thermal Cycle Test will not be performed on the Solar Alpha Rotary Joint Trundle Bearing Assemblies indicated.

RATIONALE:

Flight 1 Trundle Bearing Assemblies successfully passed acceptance thermal cycle tests. Subsequently, two TBAs failed thermal breakaway test (Serial Numbers 1017 and 1023), and their resistoboxes were removed and installed on Serial Numbers 1028 and 1027, respectively. Subsequently, these two new trundle bearings passed breakaway testing. The new complement of Flight 1 trundles was installed on SARJ 1 for functional and thermal vacuum tests. The trundle bearings are lubricated with a corrosion inhibited Micronic 601 grease, which is widely used in space applications and is good for 30 years in space with exposure to temperature ranges from –120 degrees F to 400 degrees F.

The resistoboxes were successfully tested from –5 degrees F to 163 degrees F on the first units. Resistors are designed to operate over a range from –85 degrees F to 347 degrees F, and connectors are designed to operate over a range from – 148 degrees F to 392 degrees F.

Therefore, all components are satisfactory for much higher temperature ranges and have been screened except the microswitch for the redundant journal bearings which did not have its connectivity and workmanship verified at the component–level thermal cycle test, as did the other flight units.

TBA Serial Numbers 1027 and 1028 did successfully pass one cycle of thermal breakaway testing at –15 degrees F to 148 degrees F, during which the redundant bearing microswitches were monitored. During SARJ System–level Thermal Vacuum Test, these two units were exposed to 3 cold cycles and one hot cycle. Performance was also successful in these tests at –48 degrees F and 131 degrees F. The difference in the number of cold and hot cycles is due to troubleshooting efforts required during the cold tests. The chamber had to be brought back to ambient conditions and opened up twice before performance testing could be completed at the cold temperatures. Once cold testing was completed, testing resumed by performing functional tests at the proscribed one hot cycle, and then concluding at ambient, with no further interruptions in thermal vacuum conditions. The result is TBA Serial Numbers 1027 and 1028 have been thermally cycled four times cold and twice hot.

The microswitch connectivity was otherwise verified at ambient. Thus the risk associated with use as is approval of the Trundle Bearing Assemblies indicated is minimal.

PG1–205:

ITEM:

Solar Alpha Rotary Joint Trundle Bearing Assemblies Part Number 5846485

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The Qualification Thermal Cycle Test of the Solar Alpha Rotary Joint Trundle Bearing Assemblies will be from – 51 degrees F to 163 degrees F.

RATIONALE:

The Flight Trundle Bearing Assemblies and the Qualification Trundle Bearing Assembly were tested together in the test chamber to qualification temperature limits, which incorporated the 20 degree F margins on predicted on–orbit thermal extremes at the time of test. The trundles have been thermal cycle tested to the following temperatures:

Current Predicted Temperature: –31 degrees F to 118 degrees F

Qualification Trundle Bearing Assembly: –51 degrees F to 163 degrees F

Flight 1 TBAs Serial Numbers 1014, 1015, 1016, 1018, 1019, 1020, 1021, 1022, 1024, and 1025: –5 degrees F to 163 degrees F

Flight 1 TBAs Serial Numbers 1014, 1015, 1016, 1018, 1019, 1020, 1021, 1022, 1024, and 1025: –48 degrees F to 131 degrees F in System Thermal Vacuum Test

Flight 2 TBAs Serial Numbers 1017, 1023, 1026, 1029, 1030, 1031, 1032, 1033, 1034, 1035, 1036, and 1037: –51 degrees F to 163 degrees F

Spare TBAs: –31 degrees F to 143 degrees F

The calculated qualification margins are:

Flight 1 TBAs qualification margin = 3 degrees F cold, 0 degree F hot

Flight 2 TBAs qualification margin = 0 degree F cold, 0 degree F hot

Spare TBA margin = 20 degrees F cold, 20 degrees F hot

However, all the components of the Trundle Bearing Assemblies have been rated for limits that fully envelope the thermal cycle test by virtue of the following:

The qualification Trundle Bearing Assembly were non–operating tested to –65 degrees F to 180 degrees F.

The resistoboxes are EEE parts rated for –85 degrees F to 347 degrees F.

Microswitches have been tested from –85 degrees F to 257 degrees F.

Connectors are rated for –148 degrees F to 392 degrees F.

Bearing Lubrication, Micronic 601 is rated for –120 degrees F to 400 degrees F.

Comparing the bearing test temperatures to the non–operational Qualification Trundle Bearing temperatures experienced during test, the resulting margins are:

Flight 1 margin = 17 degrees cold 17 degrees hot
Flight 2 margin = 14 degrees cold 17 degrees hot
Spares margin = 34 degrees cold 37 degrees hot

Thus, the value in repeating testing to obtain 20 degrees F qualification thermal margin for all TBAs is minimal.

PG1–206:

ITEM:

Solar Alpha Rotary Joint Part Number 5846485 Serial Number 1001

Solar Alpha Rotary Joint Trundle Bearing Assemblies Part Number 5846485 Serial Numbers 1014, 1015, 1016, 1018, 1019, 1020, 1021, 1022, 1024, 1025, 1027 and 1028

Solar Alpha Rotary Joint Drive/Lock Assembly Part Number 5847010–501 Serial Numbers 1004 and 1005

Solar Alpha Rotary Joint Rotary Joint Motor Controller Part Number 5842400 Serial Numbers 1002 and 1007

Solar Alpha Rotary Joint Utility Transfer Assembly Part Number 5839153 Serial Number 9607010

SSP 41172 REQUIREMENT:

Paragraph 6.1.1A, Assembly/Components Protoflight Tests. For the thermal vacuum tests, the temperature extremes shall be 10 degrees F (5.6 degrees C) beyond the minimum and maximum predicted temperatures.

EXCEPTION:

The indicated SARJ ORUs are exempted from meeting the 10 degrees F protoflight thermal margin in the System Protoflight Thermal Vacuum Test.

RATIONALE:

The SARJ mechanism is composed of electromechanical subassemblies or ORUs, structure, and cabling. All SARJ ORUs, the Drive/Lock Assembly, Trundle Bearing Assembly, Utility Transfer Assembly, and Rotary Joint Motor Controller, have been qualified and acceptance tested at the subsystem level. Qualification testing included temperature cycling and thermal vacuum testing to qualification margins of 20 degrees F beyond the maximum and minimum predicted operating temperatures.

Because these ORUs had been previously qualified, a tailored protoflight thermal profile was established for Flight 1 SARJ. The said ORUs would not be further exposed beyond prior acceptance thermal levels.

The margins obtained are summarized. Note that each ORU has been qualified with 20 degrees F margin, or greater, over the operational extremes.

ORU	Predicts	SARJ thermal vacuum temps	Protoflight margin (cold/hot)
RJMC	–25 to 140 degrees F	–25 to 150 degrees F	0/10 degrees F
DLA	–38 to 153 degrees F	–45 to 162 degrees F	7/9 degrees F
UTA	–39 to 133 degrees F	–45 to 143 degrees F	6/10 degrees F
TBA	–31 to 118 degrees F	–48 to 131 degrees F	17/13 degrees F

The system test indicates that there are no detrimental thermal interactions between the ORUs and the SARJ structure. The SARJ was tested to extreme hot and cold cases as well as extreme thermal gradient cases across the mechanism radially, circumferentially, and linearly. These gradient cases are considered to be more severe environments with respect to required DLA torque and overall SARJ system performance than thermal equilibrium at the extremes.

Mechanical interactions are not adversely affected by temperature. The RJMC is a fixed box. Thermal interactions between it and other SARJ structure are not applicable. The UTA transfers power and data and has a fixed mechanical interface not sensitive to system level thermal interactions. The DLA interfaces with the SARJ bull gear, which is a steel structure that has been toleranced to accommodate for thermal distortions. The DLA was also thermal vacuum tested at the component level with a bull gear STE with no performance failures. The TBAs have been qualification tested to –51 degrees F and have performed successfully to –48 degrees F in the SARJ System Thermal Vacuum Test. No bearing misalignments or galling, which would be detected through motor torque, were observed. Finally, the race ring was clean.

SARJ performance was within specification limits under all thermal extreme conditions. Motor torque, as measured by average motoring current, was well below the 1.2 ampere limit, and remained relatively constant throughout all temperatures. Pointing accuracy remained within the +/– 0.58 degree requirement for all temperatures. Analytical extrapolation of the data to 10 degrees F hotter or colder shows that the SARJ still performs to specification.

PG1–207:

ITEM:

Heat Pipe Radiator Part Numbers 1F76566

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2E, Test Description and Alternatives. The components shall be pressurized to their maximum working pressure in each of the functional modes.

EXCEPTION:

The Heat Pipe Radiator was not pressurized to the maximum working pressure during the SSP 41172, Revision T, Method V NH3 Qualification Leak Test.

RATIONALE:

Initially, a 2 minute Helium leak test was performed on the transfer tubes and the Radiator Plate tubes prior to completing the assembly. The tubes are all leak tested to verify no leaks greater than 1E–08 scc per sec by evacuating the tubes and spraying Helium around each weld. A "cupped" hand is placed around each weld to concentrate Helium at the weld during the test. A Mass Spectrometer Leak Detector is calibrated with a certified leak source of 1E–08 scc per sec. This test is a risk mitigation test with a pass/fail of 1E–08 scc per sec; however, the Ammonia (NH3) leak test is the formal leakage test.

During the NH3 leak test, all welds are final leak tested using a "aerosol–colormetric developer" in accordance with ASTM E1066–85 down to 1E–08 scc per sec of NH3. A leak of greater than 1E–08 scc per sec is indicated if the color of the developer changes from yellow to blue. The developer is applied to the weld area for a minimum of 15 minutes and is performed to a pressure of 140 psi. The Heat Pipe Radiator Maximum Operating Pressure is 437 psi based upon a maximum on–orbit temperature of 150 degrees F. Once the Heat Pipe Radiator subassemblies are filled with NH3, two NH3 leak tests and a Thermal Proof test are performed. A Thermal Proof test is performed on each hardware assembly to a minimum of 668 psi for 20 minutes, and an ambient NH3 leak test is performed prior to and after the Thermal Proof test.

The Heat Pipe Radiator subassemblies are all welded; thus, there are no permeable seals. The subassembly design does not allow conventional leakage tests to be performed since they are welded–closed tubes. Radiographic Inspection and Dye Penetrant inspections are performed on welds after the Thermal Proof test. Additionally, this method of leak testing has been performed on other Space Program hardware. i.e. Hubble Space Telescope (NICMDX), FUSE (Goddard Space Flight Center), Eurostar (Hughes A2100 (Lockheed Martin)), and NEC (MUSES). Therefore, the Helium and NH3 leakage tests assure the Heat Pipe Radiator is leak–tight to the specified leakage requirement of 1E–07 scc per sec of NH3.

PG1–208:

ITEM:

Heat Pipe Radiator Part Numbers 1F76566

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Description and Alternatives. "One of the methods given in 4.2.11.2 shall be used." This requires that components be pressurized to their maximum working pressure in each of the functional modes during a Method V Acceptance Leak Test as in 4.2.11.2E.

EXCEPTION:

The Heat Pipe Radiator was not pressurized to the maximum working pressure during the SSP 41172, Revision T, Method V NH3 Acceptance Leak Test.

RATIONALE:

Initially, a 2 minute Helium leak test was performed on the transfer tubes and the Radiator Plate tubes prior to completing the assembly. The tubes are all leak tested to verify no leaks greater than 1E–08 scc per sec by evacuating the tubes and spraying Helium around each weld. A "cupped" hand is placed around each weld to concentrate Helium at the weld during the test. A Mass Spectrometer Leak Detector is calibrated with a certified leak source of 1E–08 scc per sec. This test is a risk mitigation test with a pass/fail of 1E–08 scc per sec; however, the Ammonia (NH3) acceptance leak test is the formal leakage test.

During the NH3 leak test, all welds are final leak tested using a “aerosol–colormetric developer” in accordance with ASTM E1066–85 down to 1E–08 scc per sec of NH3. A leak of greater than 1E–08 scc per sec is indicated if the color of the developer changes from yellow to blue. The developer is applied to the weld area for a minimum of 15 minutes and is performed to a pressure of 140 psi. The Heat Pipe Radiator Maximum Operating Pressure is 437 psi based upon a maximum on–orbit temperature of 150 degrees F. Once the Heat Pipe Radiator subassemblies are filled with NH3, two NH3 leak tests and a Thermal Proof test are performed. A Thermal Proof test is performed on each hardware assembly to a minimum of 668 psi for 20 minutes, and an ambient NH3 leak test is performed prior to and after the Thermal Proof test.

The Heat Pipe Radiator subassemblies are all welded; thus, there are no permeable seals. The subassembly design does not allow conventional leakage tests to be performed since they are welded–closed tubes. Radiographic Inspection and Dye Penetrant inspections are performed on welds after the Thermal Proof test. Additionally, this method of leak testing has been performed on other Space Program hardware. i.e. Hubble Space Telescope (NICMDX), FUSE (Goddard Space Flight Center), Eurostar (Hughes A2100 (Lockheed Martin)), and NEC (MUSES). Therefore, the Helium and NH3 leakage tests assure the Heat Pipe Radiator is leak–tight to the specified leakage requirement of 1E–07 scc per sec of NH3.

The Heat Pipe Radiator Panel undergoes a Protoflight Acoustic Test (141 dB) and the Equipment Plate and Secondary Power Distribution Assembly Heat Pipe Radiator undergo an Acceptance Random Vibration Test. Post–Vibration testing, the assemblies go through a Thermal Performance test and during shipment to Boeing, a Colormetric developer is inserted into the sealed plastic bag to assure no leakage. The developer is exposed to the Heat Pipe Radiator hardware for a minimum of 24 hours during shipment, which verifies no leakage has occurred after the Acceptance test program. Finally, the manufacturer Swales has produced over 1400 Heat Pipe Radiators to date during the same manufacturing and test process. No on–orbit leaks have been reported to date.

PG1–209:

ITEM:

Load Transfer Unit Part Number D60695403–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C)(minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The Roller Link Bearing internal to the Load Transfer Unit shall be at the minimum acceptance temperature minus a margin of 15 degrees F during the cold portion of the qualification thermal vacuum test.

RATIONALE:

The Roller Link Bearing (Part Number D60695067) has a large torque margin to overcome any thermally induced increase in friction. The amount of increased friction is negligible. Based on Roller Suspension Unit wheel bearing test data, the change from –75 degrees F to –80 degrees F will increase the ballscrew load not more than 0.22 lbf. Minimum (torque limited Integrated Motor Controller Assembly) ballscrew capability is 314 lbf.

At –80 degrees F, the bearing worst case analysis calculates that the steel Roller Link Bearing can be an interference fit into the aluminum housing (D60695004), but there is large strength margin with 0.0005 inches interference between link bearing outside diameter and housing diameter at cold temperature extreme. Maximum stress is 5956 PSI with 52000 PSI allowable. Strength margin of safety is 7.73. Stress, fatigue, and life impact are negligible. With increased radial loading, all load margins are positive and life requirements are still met.

The Program assumes little additional risk with this exception, as there is little additional friction and low additional life cycle fatigue and stress.

Finally, operational workarounds are available if the roller link does not operate properly on–orbit. An EVA Contingency Jaw Release exists on the lower part of the Load Transfer Unit jaw mechanism independently allowing release of the Load Transfer Unit jaw in case the Roller Link Bearing seizes. Also, awaiting less extreme thermal conditions may eliminate any thermally induced bolt concerns.

PG1–210:

ITEMS:

NH3 Valves as follows:
Iso Relief Valves (Heat Exchanger) Part Number 1F39986
Iso Valves (Pump) Part Number 1F39985
Iso Valve (Tank/BP leg) Part Number 1F39991
Bypass Valve (Heat Exchanger) Part Number 1F40069

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be 4 hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The NH3 Valves Qualification Leakage Test was conducted with an external test pressure of 0.1 Torr or less and a duration of 30 minutes.

RATIONALE:

External leak testing of NH3 valves verifies the design and workmanship of valve assemblies. The valves are welded assemblies, which contains no seals that can be permeated. The leak testing performed verifies the integrity of the welds in the valves. The leak tests are performed in a Bell Jar Vacuum chamber. A 500 psid pressure between the valves and chamber creates a positive pressure for helium to leak through the welds if a “bad” weld exists (no seals to external environment). The chamber pressure of 0.1 Torr is sufficient to bring the Varian Mass Spectrometer Leak Detector on–line to perform the leak test. Also, a 30 minute test time duration is sufficient since leaks through welds will be immediate (no permeable seals). Thus, a single reading taken after 30 minutes is adequate. Therefore, as seal (weld) integrity will been verified, the intent of the SSP 41172 requirement is met.

PG1–211:

ITEMS:

NH3 Valves as follows:
Iso Relief Valves (Heat Exchanger) Part Number 1F39986
Iso Valves (Pump) Part Number 1F39985
Iso Valve (Tank/BP leg) Part Number 1F39991
Bypass Valve (Heat Exchanger) Part Number 1F40069

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be 4 hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The NH3 Valves Acceptance Leakage Test will be conducted with an external test pressure of 0.1 Torr or less and a duration of 30 minutes.

RATIONALE:

External leak testing of NH3 Valves verifies the workmanship of valve assemblies. The valves are welded assemblies, which contains no seals that can be permeated. The leak testing performed verifies the integrity of the welds in the valves. The leak tests are performed in a Bell Jar Vacuum chamber. A 500 psid pressure between the valves and chamber creates a positive pressure for helium to leak through the welds if a "bad" weld exists (no seals to external environment). The chamber pressure of 0.1 Torr is sufficient to bring the Varian Mass Spectrometer Leak Detector on–line to perform the leak test. Also, a 30 minute test time duration is sufficient since leaks through welds will be immediate (no permeable seals). Thus, a single reading taken after 30 minutes is adequate. Therefore, as seal (weld) integrity will been verified, the intent of the SSP 41172 requirement is met.

PG1–212:

ITEM:

NH3/H20 Heat Exchanger Part Number 1F28940

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate
Paragraph 4.2.11.2E, Method V. Leakage shall be detected using an appropriate method.

EXCEPTION:

The NH3/H20 Heat Exchanger Qualfication Leakage test method was not to a sufficient sensitivity and accuracy to verify the heat exchanger leakage requirements.

RATIONALE:

External leak testing of NH3 / H20 Heat Exchanger using Method V of SSP 41172 verifies the design and workmanship of the ORU assembly. The Heat Exchanger is a welded assembly that contains no seals that can be permeated except the Quick Disconnects, which are tested extensively at the component level. A number of leak tests are performed in the build up of these ORUs. They include pressurizing and sniff leak testing of components prior to assembly into the ORU, ORU level leak tests, as well as system–level tests in Multiple Element Integration Tests and prior to the NH3 fill of the USL end cone. The Leak testing performed on the components (which were Bell Jar tests with a 500 psi differential) verified the component leakage requirements, while the ORU leak tests verified the integrity of the welds of the components in the ORU. The Multiple Element Integration Tests with ammonia and pre–ammonia fill leak tests with Helium further validate the integrity of the ORU from the leakage standpoint. Thus, no additional leakage testing during the qualification program is warranted.

PG1–213:

ITEM:

NH3/H20 Heat Exchanger Part Number 1F28940, Serial Numbers 02, 03, and 04

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.

EXCEPTION:

The NH3/H20 Heat Exchanger Acceptance Leakage test method was not to a sufficient sensitivity and accuracy to verify the heat exchanger leakage requirements. Any subsequent leak tests shall require testing in full compliance with SSP 41172.

RATIONALE:

External leak testing of NH3 / H20 Heat Exchanger using Method V of SSP 41172 verifies the workmanship of the ORU assembly. The Heat Exchanger is a welded assembly that contains no seals that can be permeated except the Quick Disconnects, which are tested extensively at the component level. A number of leak tests are performed in the build up of these ORUs. They include pressurizing and sniff leak testing of components prior to assembly into the ORU, ORU level leak tests, as well as system–level tests in Multiple Element Integration Tests and prior to the NH3 fill of the USL end cone. The Leak testing performed on the components (which were Bell Jar tests with a 500–psi differential) verified the component leakage requirements, while the ORU leak tests verified the integrity of the welds of the components in the ORU. The Multiple Element Integration Tests with ammonia and pre–ammonia fill leak tests with Helium further validate the integrity of the ORU from the leakage standpoint.

This exception applies to Heat Exchangers (2 units) launched on the USL and the spare heat exchanger (1 unit) launched on Flight 5A.1.

PG1–214:

ITEM:

NH3 DDCU Cold Plate ORU Part Number 1F29200

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.
Paragraph 4.2.11.2E, Method V. Leakage shall be detected using an appropriate method.

EXCEPTION:

The NH3 DDCU Cold Plate ORU specification leakage requirement shall not be verified by test.

RATIONALE:

The Cold Plate is a welded assembly that contains no seals that can be permeated except the Quick Disconnects, which are tested extensively at the component level. ORU leak testing is performed using the detector probe (“sniffing”) technique; however, this method does not have sufficient sensitivity and accuracy to verify the ORU leakage requirement (3.15E–04 sccs). The pressurized portion of the ORU consists of the coldplate core and two male non–Nedox quick disconnects. These components receive an acceptable leak test at the component–level prior to being assembled into the ORU. The coldplate core undergoes a Method II type test with a 20 minute duration which is adequate to verify its allowable leakage rate of 1.0E–05 sccs. The quick disconnects receive a Method II type leak test at various temperatures (hot, cold, and ambient) which is adequate to verify their allowable leakage rate of 1.0E–04 sccs. In addition, the qualification coldplate ORU was used during Thermal Test Article testing at NASA JSC. During this test program, a helium pressure decay test was performed on the system indicating that coldplate qualification unit is not leaking significantly. Also, no noticeable ammonia leakage has been observed with the qualification unit during Thermal Test Article testing thus further establishing confidence in hardware design. Based on the component–level leak tests performed and Thermal Test Article testing with the coldplate qualification unit, sufficient confidence in the coldplate design exists such that no additional leak testing during the qualification program is warranted.

PG1–215:

ITEM:

NH3 DDCU Cold Plate ORU Part Number 1F29200 Serial Numbers 5, 6, 7, and 8

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.

EXCEPTION:

The NH3 DDCU Cold Plate ORU specification leakage rate shall not be verified by test. Any subsequent leak tests on these units shall require testing in full compliance with SSP 41172 per ATP 1F29200–P0300 Revision B.

RATIONALE:

The Cold Plate is a welded assembly that contains no seals that can be permeated except the Quick Disconnects, which are tested extensively at the component level. ORU leak testing is performed using the detector probe (“sniffing”) technique; however, this method does not have sufficient sensitivity and accuracy to verify the ORU leakage requirement (3.15E–04 sccs). The pressurized portion of the ORU consists of the coldplate core and two male non–Nedox quick disconnects. These components receive an acceptable leak test at the component–level prior to being assembled into the ORU. The coldplate core undergoes a Method II type test with a 20 minute duration which is adequate to verify its allowable leakage rate of 1.0E–05 sccs. The quick disconnects receive a Method II type leak test at various temperatures (hot, cold, and ambient) which is adequate to verify their allowable leakage rate of 1.0E–04 sccs. These units also undergo system–level testing as part of Multiple Element Integration Tests. As part of this testing, a helium pressure decay test was performed on the system indicating that these coldplates are not leaking significantly. Also, no noticeable ammonia leakage has been observed with these units during Multiple Element Integration Tests thus further establishing confidence in workmanship of these flight units.

Based on the component–level leak tests performed and system–level testing with these coldplate flight units, sufficient confidence in the workmanship of these flight units exists such that no additional leakage testing during the acceptance test program is warranted.

PG1–216:

ITEM:

NH3 MBSU Cold Plate ORU Part Number 1F39990

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate
Paragraph 4.2.11.2E, Method V. Leakage shall be detected using an appropriate method.

EXCEPTION:

The NH3 MBSU Cold Plate ORU specification leakage requirement shall not be verified by test.

RATIONALE:

The Cold Plate is a welded assembly that contains no seals that can be permeated except the Quick Disconnects, which are tested extensively at the component level. ORU leak testing is performed using the detector probe (“sniffing”) technique; however, this method does not have sufficient sensitivity and accuracy to verify the ORU leakage requirement (3.15E–04 sccs). The pressurized portion of the ORU consists of the coldplate core and two male non–Nedox quick disconnects. These components receive an acceptable leak test at the component–level prior to being assembled into the ORU. The coldplate core undergoes a Method II type test with a 20 minute duration which is adequate to verify its allowable leakage rate of 1.0E–05 sccs. The quick disconnects receive a Method II type leak test at various temperatures (hot, cold, and ambient) which is adequate to verify their allowable leakage rate of 1.0E–04 sccs. In addition, the qualification coldplate ORU was used during Thermal Test Article testing at NASA JSC. During this test program, a helium pressure decay test was performed on the system indicating that coldplate qualification unit is not leaking significantly. Also, no noticeable ammonia leakage has been observed with the qualification unit during Thermal Test Article testing thus further establishing confidence in hardware design. Based on the component–level leak tests performed and Thermal Test Article testing with the coldplate qualification unit, sufficient confidence in the coldplate design exists such that no additional leak testing during the qualification program is warranted.

PG1–217:

ITEM:

NH3 MBSU Cold Plate ORU Part Number 1F39990 Serial Numbers 2, 4 , 5, and 6

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.

EXCEPTION:

The NH3 MBSU Cold Plate ORU specification leakage rate shall not be verified by test. Any subsequent leak tests on these units shall require testing in full compliance with SSP 41172 per ATP 1F39990–P0300 Revision A.

RATIONALE:

The Cold Plate is a welded assembly that contains no seals that can be permeated except the Quick Disconnects, which are tested extensively at the component level. ORU leak testing is performed using the detector probe (“sniffing”) technique; however, this method does not have sufficient sensitivity and accuracy to verify the ORU leakage requirement (3.15E–04 sccs). The pressurized portion of the ORU consists of the coldplate core and two male non–Nedox quick disconnects. These components receive an acceptable leak test at the component–level prior to being assembled into the ORU. The coldplate core undergoes a Method II type test with a 20 minute duration which is adequate to verify its allowable leakage rate of 1.0E–05 sccs. The quick disconnects receive a Method II type leak test at various temperatures (hot, cold, and ambient) which is adequate to verify their allowable leakage rate of 1.0E–04 sccs. These units also undergo system–level testing as part of Multiple Element Integration Tests. As part of this testing, a helium pressure decay test was performed on the system indicating that these coldplates are not leaking significantly. Also, no noticeable ammonia leakage has been observed with these units during Multiple Element Integration Tests thus further establishing confidence in workmanship of these flight units.

Based on the component–level leak tests performed and system–level testing with these coldplate flight units, sufficient confidence in the workmanship of these flight units exists such that no additional leakage testing during the acceptance test program is warranted.

PG1–218:

ITEM:

HRS Radiator ORU Heater Control Assembly, LMVS Part Number 83–45547–119

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Qualification Thermal Cycle Testing of the HRS Radiator ORU Heater Control Assembly will not be performed.

RATIONALE:

Qualification Thermal Cycle testing is primarily performed to assure the hardware design can pass the acceptance thermal cycle test. All flight Heater Control Assemblies have passed their Acceptance Thermal Cycle (–45 degrees F to 120 degrees F) and Acceptance Random Vibration tests.

Thermostats are the only active component in the Heater Control Assembly and they have passed qualification testing successfully. Thermostat Screening (in accordance with MIL–STD–883E, Table I), includes Thermal Cycling (10 cycles from –85 degrees F to 300 degrees F) and Constant Acceleration testing (5000 g's for 60 seconds). Qualification Thermostat Testing includes taking "Screened Thermostats" and performing Qualification Thermal Cycle testing (200 cycles from –67 degrees F to 212 degrees F). Constant Acceleration testing is also repeated.

Other than the thermostat the Heater Control Assembly is a simple design consisting of structure, wiring, and a heater. Thus, nonperformance of a stand–alone Heater Control Assembly Qualification Thermal Cycle test does not limit the ability to certify the Heater Control Assembly thermal design.

PG1–219:

ITEM:

HRS Radiator ORU Heater Control Assembly, LMVS Part Number 83–45547–119

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperature during Qualification Thermal Vacuum Testing at Plumbrook of the HRS Radiator ORU Heater Control Assembly shall be – 34.3 degrees F.

RATIONALE:

During the Radiator ORU Qualification Thermal Vacuum test at Plumbrook, the Heater Control Assembly experienced temperatures from –34.3 degrees F to 146 degrees F. These limits do not envelope with 20 degrees margin the minimum Heater Control Assembly Acceptance Thermal Cycle of –45 degrees F. However, thermostats are the only active component in the Heater Control Assembly. Thermostats undergo thermal screening in accordance with MIL–STD–883E, Table I, for 10 cycles from –85 degrees F to 300 degrees F. Thus, the thermal screening of the active electronic component envelopes with margin the minimum Acceptance Thermal Cycle temperature.

PG1–220:

ITEM:

HRS Radiator ORU Heater Control Assembly, LMVS Part Number 83–45547–119

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Thermal Vacuum Testing of the HRS Radiator ORU Heater Control Assembly will not be performed.

RATIONALE:

Acceptance Thermal Vacuum testing is primarily performed to screen out workmanship defects. All flight Heater Control Assemblies have passed their Acceptance Thermal Cycle (–45 degrees F to 120 degrees F) and Acceptance Random Vibration Tests.

Thermostats are the only active component in the Heater Control Assembly. Thermostats undergo thermal screening in accordance with MIL–STD–883E, Table I, for 10 cycles from –85 degrees F to 300 degrees F.

The Heater Control Assembly operating temperature limits from –45 degrees F to 120 degrees F are within the thermostats capability of –85 degrees F to 300 degrees F. There are no heat dissipation issues that would be discovered via an acceptance thermal vacuum test as a thermostat dissipates only 1 watt through the baseplate. Analysis indicates the maximum temperature increase under vacuum conditions as 5 degrees F. Thus, the HRS Radiator ORU Heater Control Assembly Thermostats would remain well within the tested capability of 300 degrees F.

PG1–221:

ITEM:

HRS Radiator ORU Heater Control Assembly, LMVS Part Number 83–45547–119

SSP 41172 REQUIREMENT:

Paragraph 4.2, Component Qualification. The word "required" means that, as a minimum, the component is required to be tested if the subject environment is experienced during the component's life cycle.

EXCEPTION:

Qualification Thermal Vacuum and Random Vibration testing shall be performed on 2 separate Heater Control Assembly test articles.

RATIONALE:

A Qualification Heater Control Assembly test unit has experienced Qualification Random Vibration levels. This unit also experienced Acceptance Thermal Cycle and Random Vibration testing prior to the Qualification Random Vibration test.

A different Qualification Heater Control Assembly unit and one Flight Heater Control Assembly underwent Thermal Vacuum testing during ORU Qualification and Flight #4 ORU thermal vacuum testing at Plumbrook (to 3 and 1 thermal cycles, respectively, at qualification temperature extremes). During the Radiator ORU Qualification Thermal Vacuum test at Plumbrook, the Heater Control Assembly experienced temperatures from –34.3 degrees F to 146 degrees F.

Thermostats are the only active component in the Heater Control Assembly. Thermostats undergo thermal screening in accordance with MIL–STD–883E, Table I, for 10 cycles from –85 degrees F to 300 degrees F. Other than the thermostat, the Heater Control Assembly is a simple design consisting of structure, wiring, and a heater. Thus, the Heater Control Assembly design is adequately verified via the Qualification Heater Control Assembly that undergoes Qualification Random Vibration testing with its internal thermostats experiencing MIL–STD–883E, Table I, compliant thermal screening. The anticipated effects on all environmental testing on one test article have been effectively shown via the testing performed.

PG1–222: This exception replaces PG1–20

ITEMS:

Keel Nut Brake Part Numbers 1F83308 and 1F83309
MT Stop– EVA Part Number 1F80512
Energy Absorber Launch Restraint Clamps Part Numbers 1F75356 and 1F75357
Portable Work Platform Launch Restraint Clamps Part Numbers 1F75718, 1F75721, 1F75723, 1F75724, and 1F75796
Bolt Lock Assembly Part Number 1F82450
Tether Shuttle Stop Part Number 1F83123
MT Stop, Robotic Part Number 1F83040
Solar Alpha Rotary Joint Multilayer Installation EVA Clamp Part Number 1F83280

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Thermal Vacuum Test will not be performed on the indicated moving mechanical assemblies.

RATIONALE:

A Qualification Thermal Vacuum Test with a minimum of three temperature cycles will not be performed on the indicated moving mechanical assemblies. Instead, single–cycle thermal tests at qualification temperatures will be performed in the Human Thermal Vacuum chamber.

The performance of this class of simple EVA–actuated moving mechanical assemblies at temperature extremes is not degraded by multiple cycles of temperature extremes. Rather, the performance is dictated by the clearance necessary to allow a particular motion to occur, and as such, is fully repeatable. Assembly tolerance analyses that verify clearance at temperature, along with single–cycle Human Thermal Vacuum testing at qualification temperatures, will be sufficient to prove each design is capable of operating in extreme thermal environments.

Tolerance analysis reports are as follows:
MDC 00H1963 Segment S0 Moving Mechanical Assemblies Thermal Analyses
MDC 00H1964 Segment S1/P1 Moving Mechanical Assemblies Thermal Analyses
MDC 00H1965 Segment P3/S3 Moving Mechanical Assemblies Thermal Analyses

As–run temperatures in Human Thermal Vacuum tests can deviate from test objectives (maximum on–orbit predictions plus 20 degrees F), particularly with smaller components, due to chamber heat loss. For cases where as–run temperatures deviated by more than 5 degrees F, the review of tolerance analyses will constitute acceptance. The following components failed to hold test temperatures during their Human Thermal Vacuum runs:

MT Stop – EVA Part Number 1F80512
Qualification Temperature: –140 degrees F, 156 degrees F
As–run temperatures: –125 degrees F, 160 degrees F

Solar Alpha Rotary Joint Multi–layer Installation EVA Clamp Part Number 1F83280
Qualification Temperature: –130 degrees F, 160 degrees F
As–run temperatures: –82 degrees F, 124 degrees F

PG1–223:

ITEM:

NODE 3 Tray Strut Part Number 1F76284

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration, ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements, ALL REQUIREMENTS.

EXCEPTION:

A Qualification Thermal Vacuum test shall not be performed for the Node 3 Tray Strut.

RATIONALE:

The Human Thermal Vacuum test performed at acceptance temperatures on the Node 3 Tray Strut, along with tolerance analysis showing adequate clearances with a 20 degrees F margin applied at the minimum temperature, will be deemed sufficient to certify the Node 3 Tray Strut design for thermal vacuum conditions.

The simple hinge joint allowing deployment of the Node 3 Tray Strut can function provided clearance exists between the hinge pin (bolt) and either the strut or the clevis to which it is attached. The worst case drawing tolerances between the pin and strut at room temperature provide a .0025 inch clearance. Calculations (MDC 00H1963, Segment S0 Moving Mechanical Assemblies Thermal Analyses) show that the clearance both at –120 degrees F, where the assembly was tested, and –140 degrees F is .0022 inch, indicate the difference is less than one–thousandth of an inch.

The following calculations demonstrate sufficient clearance at –140 degrees F (delta from room temp = 210 degrees F)

Bolt, NAS1160–5, CRES, Maximum Diameter=0.3745 inch
Clevis, 1F76308, Al, Minimum Diameter=0.375 inch
Strut, 1F76306, Al, Minimum Diameter=0.377 inch

Use a_1=8.0E–06 for CRES thermal coefficient at –120 degrees F to –140 degrees F.
a_2=12.4E–06 for Al thermal coefficient at –120 degrees F to –140 degrees F

At room temperature
Minimum clearance between Bolt and Strut
=0.377–0.3745
=0.0025 inch

@–120 degrees F
D Dia (Bolt)=0.3745*8.0E–06*210
=0.000569
D Dia (Strut)= 0.377*12.4E–06*210
=0.000859

Minimum clearance between Bolt and Strut
=0.0025–(0.000859–0.000569)
=0.0025–0.00029
=0.0022 inch

@–140 degrees F
D Dia (Bolt)=0.3745*8.0E–06*210
=0.000629
D Dia (Strut)= 0.377*12.4E–06*210
=0.000950

Minimum clearance between Bolt and Strut
=0.0025–(0.000950–0.000629)
=0.0025–0.000321
=0.0022 inch

PG1–224:

ITEM:

NH3 Accumulator Part Number 1F96456 Serial Numbers D0001 and D0002

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Description and Alternatives. ALL REQUIREMENTS.
Paragraph 5.1.7.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

A SSP 41172, Revision T, Method VI leak test for a duration of 30 minutes was performed in lieu of a SSP 41172, Revision T, Method II Fine Leak Test.

RATIONALE:

The specified maximum allowable leakage rate requirement documented in the Boeing NH3 Accumulator Specification 1F96456 is 1E–05 standard cubic centimeters per second Helium. However, this leakage rate is valid only under Maximum Operating Pressure conditions of 500 psig. As leakage testing was performed at 14.7 psig, it is necessary to validate the leakage rate against a pressure–corrected requirement.

Assuming molecular flow through available leak paths, a pressure–correction factor of 34 (500 psig/ 14.7 psig) is applied to the specified maximum allowable leakage rate. The result is a pressure–corrected maximum allowable leakage rate of 3E–07 standard cubic centimeters per second Helium [(1E–05/34) standard cubic centimeters per second Helium]. The released Acceptance Test Procedure AT 2351650 under which the Leak Test was performed for these units did incorporate this pressure–corrected maximum allowable leakage rate.

For the Acceptance Leak Test, a Mass Spectrometer Leak Detector was calibrated with an internal leak source of 2E–10 standard cubic centimeters per second Helium both prior to and after the leak test. Both the NH3 and N2 sides of the Accumulator were leak and pressure tested, and all welds were dye–penetrant and ultrasonic inspected. The performed test duration of 30 minutes is sufficient to check leakage across the Accumulator welds, as there are no permeable seals. The measured leakage rate during the Acceptance Leak Testing of the NH3 Accumulator Serial Number D0001 was 6.8E–09 standard cubic centimeters per second Helium and for the NH3 Accumulator Serial Number D0002 was 5E–08 standard cubic centimeters per second Helium. As these are at least one–sixth the level of the pressure–corrected maximum allowable leakage rate of 3E–07 standard cubic centimeters per second Helium, the leakage rates of the indicated NH3 Accumulators during the Acceptance Leak Test program are deemed acceptable.

Finally, prior to the Acceptance Leak Tests, successful Proof Pressure Tests were completed on these flight units. Thus, based on all accumulated test data, the indicated NH3 Accumulators are acceptable.

PG1–225:

ITEM:

NH3 Accumulator Part Number 1F96456 Serial Numbers D0003, D0004, D0005, and D0006

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Description and Alternatives. ALL REQUIREMENTS.
Paragraph 5.1.7.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

A SSP 41172, Revision T, Method VI leak test for a duration of at least 60 minutes shall be performed in lieu of a SSP 41172, Revision T, –compliant Method II Fine Leak Test.

RATIONALE:

The specified maximum allowable leakage rate requirement documented in the Boeing NH3 Accumulator Specification 1F96456 is 1E–05 standard cubic centimeters per second Helium. However, this leakage rate is valid under Maximum Operating Pressure conditions of 500 psig. As leakage testing shall be performed at 14.7 psig, it is necessary to validate the leakage rate against a pressure–corrected requirement.

Assuming molecular flow through available leak paths, a pressure–correction factor of 34 (500 psig/ 14.7 psig) is applied to the specified maximum allowable leakage rate. The result is a pressure–corrected maximum allowable leakage rate of 3E–07 standard cubic centimeters per second Helium [(1E–05/34) standard cubic centimeters per second Helium]. Via contract direction to the vendor, the Acceptance Test Procedure AT 2351650 under which the Leak Test shall be performed incorporates a more–stringent allowable leakage rate of 1E–07 standard cubic centimeters per second Helium. The Acceptance Test Procedure includes test duration of at least 60 minutes and leakage rate stabilization demonstrated via three measured leakage rates recorded every five minutes with variation not greater than ten percent.

For the Acceptance Leak Test, a Mass Spectrometer Leak Detector shall be calibrated with an internal leak source of 2E–10 standard cubic centimeters per second Helium both prior to and after the leak test. Both the NH3 and N2 sides of the Accumulator shall be leak and pressure tested, and all welds shall be dye–penetrant and ultrasonic inspected. The test duration of 60 minutes will be sufficient to check leakage across the Accumulator welds as there are no permeable seals. In conjunction with three consecutive readings in 5–minute intervals with variation not greater than 10 percent to verify leakage stability, a measured leakage rate below 1E–07 standard cubic centimeters per second Helium will be sufficient to validate the workmanship of the Accumulator relative to any leakage.

PG1–226:

ITEM:

NH3 Tank Part Number 1F40057–2
NH3 Tank Part Number 1F78210–1 Serial Number 008

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Description and Alternatives. ALL REQUIREMENTS.
Paragraph 5.1.7.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

A SSP 41172, Revision T, Method VI leak test for a duration of at least 60 minutes shall be performed in lieu of a SSP 41172, Revision T, –compliant Method II Fine Leak Test.

RATIONALE:

The specified maximum allowable leakage rate requirement documented in the Boeing NH3 Tank Specification 1F40057 is 1E–05 standard cubic centimeters per second Helium. However, this leakage rate is valid under Maximum Operating Pressure conditions of 500 psig. As leakage testing shall be performed at 14.7 psig, it is necessary to validate the leakage rate against a pressure–corrected requirement.

Assuming molecular flow through available leak paths, a pressure–correction factor of 34 (500 psig/ 14.7 psig) shall be applied to the specified maximum allowable leakage rate. The result is a pressure–corrected maximum allowable leakage rate of 3E–07 standard cubic centimeters per second Helium [(1E–05/34) standard cubic centimeters per second Helium]. Via SSCN 005213, the Acceptance Test Procedure AT 2351330 under which the Leak Test shall be performed incorporates a more–stringent allowable leakage rate of 1E–07 standard cubic centimeters per second Helium. The Acceptance Test Procedure includes test duration of at least 60 minutes and leakage rate stabilization demonstrated via three measured leakage rates recorded every five minutes with variation not greater than ten percent.

For the Acceptance Leak Test, a Mass Spectrometer Leak Detector shall be calibrated with an internal leak source of 2E–10 standard cubic centimeters per second Helium both prior to and after the leak test. Both the NH3 and N2 sides of the tank shall be leak and pressure tested, and all welds shall be dye–penetrant and ultrasonic inspected. The test duration of 60 minutes will be sufficient to check leakage across the tank welds as there are no permeable seals. In conjunction with three consecutive readings in 5–minute intervals with variation not greater than 10 percent to verify leakage stability, a measured leakage rate below 1E–07 standard cubic centimeters per second Helium will be sufficient to validate the workmanship of the NH3 tanks relative to any leakage.

PG1–227:

ITEM:

Radiator ORU Squib Fire Unit Part Number 83–39387

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.4, Supplementary Requirements. During the remainder of the test, electrical components, including all redundant circuits, shall be cycled through various operational modes and parameters monitored for failures and intermittences.

EXCEPTION:

The Squib Fire Unit was not energized and monitored during temperature transitions during the Qualification Thermal Cycle Test.

RATIONALE:

The likelihood of intermittence causing a failure in the 3–second life of the SFU is remote. If a failure did occur, the fire sequence can be repeated as many times as necessary. Functional tests at ambient conditions have been completed at the Squib Fire Unit assembly level between random vibration and thermal cycle tests, and at both hot and cold temperature extremes during the first and last cycles of the thermal cycle qualification test. Functional tests at ambient conditions were also performed on the SFU qualification unit at the Radiator ORU level of assembly before and after Acoustic vibration qualification testing. In the event that the SFU fails to provide the current pulse required, an EVA backup cinch release is available. Power–on and monitoring of the SFU was performed as part of the Radiator ORU Acoustic tests on three of the flight Radiator ORU units (Serial Numbers 007, 008, and 009). No current drift or intermittences were detected.

PG1–228:

ITEM:

Radiator ORU Squib Fire Unit Part Number 83–39387

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.4 Supplementary Requirements. During the remainder of the test within the components operating temperature range, electrical components shall be cycled through various operational modes and parameters monitored for failures and intermittences.

EXCEPTION:

The Squib Fire Units are not energized and monitored during temperature transitions during the Acceptance Thermal Cycle Test.

RATIONALE:

The likelihood of intermittence causing a failure in the 3–second life of the SFU is remote. If a failure did occur, the fire sequence can be repeated as many times as necessary. Functional tests at ambient conditions have been completed at the Squib Fire Unit assembly level between random vibration and thermal cycle tests, and at both hot and cold temperature extremes during the first and last cycles of the thermal cycle acceptance test. Functional tests at ambient conditions were also performed on the SFU flight units at the Radiator ORU level of assembly before and after Acoustic vibration acceptance testing. In the event that the SFU fails to provide the current pulse required, an EVA backup cinch release is available. Power–on and monitoring of the SFU was performed as part of the Radiator ORU Acoustic tests on three of the flight Radiator ORU units (Serial Numbers 007, 008, and 009). No current drift or intermittences were detected.

PG1–229:

ITEM:

Radiator ORU Squib Fire Unit Part Number 83–39387

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.4 Supplementary Requirements. Electrical and electronic components shall be energized and monitored during the test.

EXCEPTION:

The Squib Fire Unit was not energized and monitored during the Qualification Random Vibration Test.

RATIONALE:

The likelihood of intermittence causing a failure in the 3–second life of the SFU is remote. If a failure did occur, the fire sequence can be repeated as many times as necessary. Functional tests at ambient conditions have been completed at the Squib Fire Unit assembly level between random vibration and thermal cycle tests, and at both hot and cold temperature extremes during the first and last cycles of the thermal cycle qualification test. Functional tests at ambient conditions were also performed on the SFU qualification unit at the Radiator ORU level of assembly before and after Acoustic vibration qualification testing. In the event that the SFU fails to provide the current pulse required, an EVA backup cinch release is available. Power–on and monitoring of the SFU was performed as part of the Radiator ORU Acoustic tests on three of the flight Radiator ORU units (Serial Numbers 007, 008, and 009). No current drift or intermittences were detected.

PG1–230:

ITEM:

Radiator ORU Squib Fire Unit Part Number 83–39387

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.4 Supplementary Requirements. Electrical and electronic components shall be energized and monitored during the test.

EXCEPTION:

The flight Squib Fire Units are not energized and monitored during the Acceptance Random Vibration Test.

RATIONALE:

The likelihood of intermittence causing a failure in the 3–second life of the SFU is remote. If a failure did occur, the fire sequence can be repeated as many times as necessary. Functional tests at ambient conditions have been completed at the Squib Fire Unit assembly level between random vibration and thermal cycle tests, and at both hot and cold temperature extremes during the first and last cycles of the thermal cycle acceptance test. Functional tests at ambient conditions were also performed on the SFU flight units at the Radiator ORU level of assembly before and after Acoustic vibration acceptance testing. In the event that the SFU fails to provide the current pulse required, an EVA backup cinch release is available. Power–on and monitoring of the SFU was performed as part of the Radiator ORU Acoustic tests on three of the flight Radiator ORU units (Serial Numbers 007, 008, and 009). No current drift or intermittences were detected.

PG1–231:

ITEM:

Radiator ORU Squib Fire Unit Part Number 83–39387

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The Qualification Thermal Cycle test for the Squib Fire Unit was performed from – 65 degrees F to 140 degrees F.

RATIONALE:

The Qualification Thermal Cycle test for the Squib Fire Unit was performed from – 65 degrees F to 140 degrees F. However, six flight units were exposed to temperatures from – 65 degrees F to 140 degrees F during Acceptance tests. Thus, no qualification thermal margin exists.

All Squib Fire Unit electronic components (EEE parts) are qualified from – 67 degrees F to 257 degrees F . During the Radiator ORU Qualification Thermal Vacuum test (at Plumbrook), the Squib Fire Unit reached –110 degrees F and passed a post–test ambient functional test (Pyro Shock test, 3 firings). Additionally, a delta qualification thermal vacuum test was successfully performed to – 79 degrees F which provides 14 degrees F thermal margin. Thus, the risk of incurring undetected damage to the flight Squib Fire Units by exposure to –65 degrees F acceptance tests is minimal and the risk associated with no qualification thermal margin is mitigated.

The delta thermal vacuum test also included exposure and functional performance of the Squib Fire Unit at the maximum temperature of 160 degrees F. This also meets the intent of the qualification thermal margin requirement and mitigates the risk associated with no qualification thermal margin.

Relative to the flight units, acceptance testing the Squib Fire Units to 140 degrees F is well below the 257 degrees F electronic component testing. Functional tests at ambient conditions have been completed at the Squib Fire Unit assembly level between random vibration and thermal cycle tests, and at both hot and cold temperature extremes during the first and last cycles of the thermal cycle acceptance test. Functional tests at ambient conditions were also performed on the SFU flight unit at the Radiator ORU level of assembly before and after Acoustic Vibration acceptance testing. Finally, an EVA backup cinch release is available in addition to repetitive attempts to fire squibs as many times as necessary (including an operational workaround to fire in a more benign temperature if needed) should any difficulties arise.

PG1–232:

ITEM:

Radiator ORU Squib Fire Unit Part Number 83–39387

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Qualification.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Thermal Vacuum test is not performed on the Flight Squib Fire Units.

RATIONALE:

Thermal Vacuum testing at the high temperature extreme assures the hardware can dissipate heat, via radiation, and still operate. However, no heat dissipation concerns exist under nominal operational conditions due to the limited powered duration that is as little as 3 seconds.

PG1–233:

ITEM:

Radiator ORU Squib Fire Unit Part Number 83–39387

SSP 41172 REQUIREMENT:

Paragraph 5.1.8, Burn–In Test, Component Qualification.
Paragraph 5.1.8.3C, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The fully assembled Squib Fire Unit shall not undergo an Acceptance Burn–In Test.

RATIONALE:

The Squib Fire Unit electronics consist of capacitors and two circuit boards which contain solid state relays. The solid state relays, which close to allow the current to fire the squib, are burned in for 160 hours in accordance with MIL–SPEC–R–28750. Each flight Squib Fire Unit is subjected to a minimum of 9 functional tests. These include pre– and post– environmental functional tests at the Squib Fire Unit and Radiator ORU level, as well as functional tests during the Squib Fire Unit Acceptance Thermal Cycle test extremes.

PG1–234:

ITEM:

Radiator ORU Squib Fire Unit Part Number 83–39387

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:
(a) The maximum predicted flight level and spectrum, but not less than a level derived from an acoustic environment of 141 dB; and
(b) Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The Qualification Random Vibration test performed on the Squib Fire Unit assembly did not envelop the maximum predicted flight level and spectrum in the 650–Hertz range.

The Qualification Random Vibration test performed on the Squib Fire Unit assembly did not envelop acceptance test levels and spectrum plus test tolerances.

RATIONALE:

The Qualification Squib Fire Unit and all 6 flight Squib Fire Units underwent Acceptance Random Vibration tests to 7.8 grms for 60 seconds in all 3 axes. The Qualification Squib Fire Unit was subsequently installed in the HRS Radiator ORU Qualification Unit and underwent Acoustic Vibration tests to 138.5 dB for 62 seconds, 144.6 dB for 190 seconds, and 141.8 dB for 24 minutes, 53 seconds. The 141–dB test was performed because the 144–dB test was not performed within test tolerances at frequencies below 90 Hertz. The 141–dB level test duration was derived from MIL–STD–810 for equivalent fatigue life as a 144–dB test for 188 seconds. Subsequently, the Qualification Squib Fire Unit underwent a Qualification Random Vibration test to 6 dB over acceptance levels over the 20–90 Hertz frequency range (5.28 grms) for 188 seconds in all 3 axes. This test was added to account for the launch vibration below 80 Hertz and does account for any peak stress and deflection concerns.

A Development test was performed for 3 minutes per axis in accordance with 1F01920B in 1993 on a SFU development unit to levels corresponding to 3 dB over the acceptance random vibration testing performed on the flight Squib Fire Units. These test results are documented in LM Test Report 3–47300H/3DIR–029, dated 06/15/93. Minor Design differences exist between the Development unit and the Qualification/Flight units.

Significant fatigue life has been demonstrated based on the cumulative vibration testing performed on the Qualification Squib Fire Unit. Based upon the vibration testing performed and the ability to uncinch manually via EVA, it is unnecessary to repeat the Qualification Random Vibration test.

PG1–235:

ITEM:

Radiator ORU Squib Fire Unit Part Number 83–39387

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3C, Test Levels and Duration. A minimum of three temperature cycles shall be used.

EXCEPTION:

The SFU experienced one temperature cycle at the non–operational and operational temperature extremes during qualification thermal vacuum testing.

RATIONALE:

Three Qualification Thermal Vacuum Cycles are performed to provide margin for Acceptance Thermal Vacuum testing to allow for acceptance retest scenarios. This philosophy assures the design is adequate and provides confidence that the hardware will successfully pass acceptance testing and, if required, re–acceptance testing. Since an acceptance thermal vacuum test is not performed on the Squib Fire Unit (SFU) flight units, multiple qualification thermal vacuum cycles are not required. One Thermal Vacuum cycle to both non–operational temperature extremes and operational temperature extremes is sufficient to subject the SFU. This test will verify the design is capable to survive and operate in the predicted on–orbit thermal environment.

PG1–236:

ITEM:

SARJ Trundle Resistor Box (Resistobox) Part Number 5847281–501

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. The test duration in each of the three orthogonal axes shall be three times the expected flight exposure time to the maximum predicted level and spectrum or three times the component random vibration acceptance test time if that is greater, but not less than three minutes per axis. Component random vibration test levels and spectrums shall be the envelope of the following:

A. The maximum predicted flight level and spectrum, but not less than a level derived from an acoustic environment of 141 dB (whose spectrum is defined by NSTS–21000–IDD–ISS, Table 4.1.1.5–1).

B. Acceptance test levels and spectrums plus test tolerances.

EXCEPTION:

An additional Random Vibration Qualification Test of the SARJ Trundle Resistobox in the Z–axis shall be performed at 3 dB over the “incorrect” (X/Y) acceptance level for 60 seconds at levels indicated. This is an exception to the 6 dB difference between acceptance and qualification levels required by test tolerances and also an exception to the duration of three minutes per axis.

Frequency (Hertz)	X/Y–Axes Acceptance Test Levels (Incorrect Z–Axis Level)	Z–Axis Acceptance Levels (Correct)	Z–Axis Qualification (Z–Axis Acceptance + 6 dB)	Z–Axis Delta Qualification (X/Y–Axes Acceptance + 3 dB)
20	0.01	0.01	0.04	0.02
50	0.075	0.075	0.3	0.15
100	0.075	0.075	0.3	0.15
200	0.075	0.25	1.0	0.15
350	0.075	0.25	1.0	0.15
500	0.25	0.25	1.0	0.5
600	0.25	0.25	1.0	0.5
800	0.25	0.1	0.4	0.5
1000	0.75	0.1	0.4	1.5
1700	0.75	0.1	0.4	1.5
2000	0.025	0.0125	0.05	0.05
Grms	28.3	16.0	31.9	40.0
Duration	60 seconds	60 seconds	180 seconds	60 seconds

RATIONALE:

Two SARJ Flight Unit #1 Trundle Resistoboxes, Serial Numbers 1014 and 1015, had X/Y–axes acceptance test levels inadvertently applied in the Z–axis. The “incorrect” Z–axis applied level with the appropriate acceptance and qualification levels is as indicated. Acceptance testing continued on the Resistobox flight units, Serial Numbers 1014 and 1015, with the appropriate levels being applied in all three axes. Additional qualification testing on SARJ Trundle Resistobox Serial Number 1013 was performed to verify and compensate for the inappropriate Z–axis test on SARJ Flight Unit #1 Trundle Resistoboxes, Serial Numbers 1014 and 1015.

The qualification hardware would have been put at higher risk if a test level and duration of 6 dB above acceptance test levels for 180 seconds were used for the additional testing. Expert judgment was that 3 dB above acceptance test levels for 60 seconds was sufficient, and this was confirmed with high–cycle fatigue calculations, which determines that the expendable life predicted for the lifetime of the flight parts was adequate. Since total demonstrated fatigue life expended should be less than 100 percent, this provides a pad against material property uncertainty implicit in the calculations. Finally, the parts are not safety catastrophic or critical.

PG1–237:

ITEM:

SARJ Drive Lock Assembly Resistor Box (Resistobox) Part Number 5847281–505

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. The test duration in each of the three orthogonal axes shall not be less than one minute per axis. The acceptance test spectrum input may be adjusted in the components resonant frequency zones(s) to reduce the component resultant level to within the test level spectrum. Component random vibration test levels and spectrums shall be the envelope of the following:

A. The maximum predicted flight level and spectrum minus 6 dB, but not less than a level derived from a 135 dB overall acoustic environment (whose spectrum is defined by NSTS–21000–IDD–ISS, paragraph 4.1.1.5).

B. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime contractor.

EXCEPTION:

Acceptance Random Vibration tests on the SARJ Drive Lock Assemblies Resistobox shall be performed below the workmanship screening level.

RATIONALE:

On April 29, 1998 the T&VCP approved acceptance random vibration testing of the SARJ Drive Lock Assemblies at 4.3 grms. The T&VCP requested that vibration responses be monitored at the three resistoboxes, Part Number 5847281–505, which are mounted to the SARJ Drive Lock Assemblies. During the SARJ Drive Lock Assemblies acceptance random vibration testing, less energy than expected was transmitted into the resistoboxes. As a result, the SARJ Drive Lock Assembly Resistoboxes acceptance random vibration test level was less than the minimum vibration screening level of 6.1 grms but greater than the predicted maximum flight level of 1.9 grms (Note: the SARJ Drive Lock Assembly resistoboxes do not receive a “stand–alone” acceptance random vibration test; the resistobox acceptance random vibration test occurs at the SARJ DLA level–of–assembly).

The resistobox is a simple assembly consisting of a machined aluminum structure, a circuit board with eight resistors, and an electrical connection. As such, the workmanship of the resistobox can be reasonably verified via visual inspection and thermal cycle acceptance testing, therefore minimizing the risk associated with lower than required acceptance random vibration test levels. The SARJ Drive Lock Assembly Resistoboxes successfully completed thermal cycling and visual inspection to detect material and workmanship defects.

Each SARJ Drive Lock Assembly contains three resistoboxes. Two resistoboxes measure follower arm position, while a third resistobox measures the Engage/Disengage Mechanism stepper motor position. Two microswitches are associated with each position measurement, thus providing a single failure tolerant monitoring capability. Microswitch or resistobox circuit board failure will give a false reading. The software/operator will be able to detect all failure scenarios. The worst case effect would be a shutdown of the SARJ until the failure could be identified.

PG1–238:

ITEM:

Heat Pipe Radiator Part Numbers 1F93263 and 1F93264

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When there is no dedicated qualification test article and all production articles are intended for flight usage, the test shall be the same (as defined in paragraph 4.2 for component qualification)...” This requires that the components be pressurized to their maximum working pressure in each of the functional modes during protoflight leak testing in accordance with 4.2.11.2E.

EXCEPTION:

The Heat Pipe Radiator assemblies were not pressurized to the maximum working pressure during the Ammonia (NH3) Leak Test in accordance with SSP 41172, paragraph 4.2.11.2E, Method V.

RATIONALE:

Initially, a 2–minute Helium leak test was performed on the Radiator assembly heat pipe tubes prior to completing the assembly. The tubes are all leak tested to verify that there are no leaks greater than 1E–08 scc per sec Helium by evacuating the tubes and spraying Helium around each weld. A “cupped” hand is placed around each weld to concentrate Helium at the weld during the test. A Mass Spectrometer Leak Detector is calibrated with a certified leak source of 1E–08 scc per sec Helium. This test is a risk mitigation test with a pass/fail of 1E–08 scc per sec Helium; however, the Ammonia leak test is the formal leak test.

During the NH3 leak test, all welds are final–leak tested using an aerosol color–change developer in accordance with ASTM E1066–95 down to 1E–07 scc per sec of NH3 or less minimum detectable leakage rate depending on test time and Ammonia concentration. A leak of greater than minimum detectable leakage rate is indicated if the color of the developer changes from yellow to blue. The developer is applied to the weld area for a minimum of 15 minutes and the test is performed at a pressure of 140 psia, which corresponds to a temperature of 75 degrees F. Therefore, an absence of leaks with a leakage rate greater than 6E–08 scc per sec of NH3 is verified. The Heat Pipe Radiator Maximum Design Pressure is 433 psia corresponding to an on–orbit temperature of 150 degrees F. The maximum operating pressure is 287 psia corresponding to a temperature of 120 degrees F without causing damage to the MDM. Once the Heat Pipe Radiator subassemblies are filled with NH3, two NH3 leak tests and a Thermal Proof test are performed. A Thermal Proof test is performed on each hardware assembly to a minimum of 989 psia (at 221 degrees F) for 20 minutes, and an ambient NH3 leak test is performed prior to and after the Thermal Proof test.

The Heat Pipe Radiator assemblies are all welded; thus, there are no permeable seals. The subassembly design does not allow conventional leak tests to be performed since they are welded–closed tubes. Radiographic Inspection and Dye Penetrant inspections are performed on welds after the Thermal Proof test. Additionally, this method of leak testing has been performed on other Space Program hardware, i.e. Hubble Space Telescope (NICMDX), FUSE (Goddard Space Flight Center), Eurostar (Hughes A2100 (Lockheed Martin)) and NEC (MUSES). Therefore, the Helium and NH3 leak tests assure the Heat Pipe Radiator Assemblies are leak–tight to the specified leakage requirement of 1E–07 scc per sec of NH3.

Each Heat Pipe Radiator assembly undergoes a protoflight random vibration test. For post random vibration testing, the assembly goes through a thermal cycling test and a final thermal performance test and during shipment to Boeing, a color–change developer is inserted into the sealed plastic bag to assure no leakage. The developer is exposed to the Heat Pipe Radiator hardware for a minimum of 24 hours during shipment, which verifies that no leakage greater than 1E–08 scc per sec of NH3 has occurred after the protoflight test program. Finally, the manufacturer (Swales) has produced over 1400 Heat Pipe Radiators to date using the same manufacturing and test process. No on–orbit leaks have been reported to date.

PG1–239:

ITEM:

TRRJ Drive/Lock Assembly Part Number 5846872, Serial Numbers 001, 002, 003, and 004

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.
Paragraph 6.1.1A. For the thermal vacuum tests, the temperature extremes shall be 10 degrees F (5.6 degrees C) beyond the minimum and maximum predicted temperatures.

EXCEPTION:

The TRRJ Drive/Lock Assembly experienced a maximum temperature of 150 degrees F during Protoflight Thermal Vacuum Testing, resulting in zero degrees of margin. Future testing of TRRJ Drive/Lock Assemblies shall be in compliance with SSP 41172 requirements.

RATIONALE:

The TRRJ Drive/Lock Assembly experienced a maximum temperature of 150 degrees F during Protoflight Thermal Vacuum Testing. However, changes in the predicted On–Orbit environment resulted in an increase in the maximum predicted TRRJ Drive/Lock Assembly temperature from 140 degrees F to 150 degrees F. Therefore, 0 degrees F thermal margin was exhibited during the Protoflight Thermal Vacuum Test.

The test–validated TRRJ thermal model results in the predictions for TRRJ Drive/Lock Assembly components as follows:

Motor Windings	–31.5 degrees F to 150 degrees F
Housing, motor resolver	–31.6 degrees F to 139 degrees F
Housing, inter. Upper	–30.2 degrees F to 120 degrees F
Housing, inter. Lower	–30 degrees F to 105 degrees F
Frame AFT	–23 degrees F to 95 degrees F
Frame FWD	–21 degrees F to 84 degrees F
Stepper motor	–26 degrees F to 99 degrees F
Pinion housing	–34 degrees F to 90 degrees F
Pinion gear	–39 degrees F to 81 degrees F
Overall predict range	–39 degrees F to 150 degrees F

The maximum hot temperature in the TRRJ Drive/Lock Assembly is in the motor windings. Thus, this point was used for monitoring and recording the maximum temperature during thermal testing.

The motor windings experienced 150 degrees F during protoflight thermal vacuum testing. The next hottest predicted temperature in the TRRJ Drive/Lock Assembly is in the motor resolver housing (139 degrees F). The temperature recorded here was 154 degrees F during the protoflight thermal vacuum testing. This, and all remaining components, experienced temperatures at least 10 degrees F higher than the predicted on–orbit temperatures during the protoflight thermal vacuum testing. Although the motor windings did not see a 10 degrees F margin during the thermal vacuum testing of the TRRJ Drive/Lock Assembly, it experienced 257 degrees F during burn–in testing at the vendor, and the source control document requires performance to 176 degrees F. Thus, the vendor testing exposes the motor windings to temperatures far exceeding the expected conditions the TRRJ Drive/Lock Assembly will experience on–orbit. Additionally, the predicted temperatures are based on a maximum power dissipation of 50 Watts from the motor, but the average drive motor torque is 8 Watts under system load.

After the baseline thermal vacuum testing was complete, the TRRJ Drive/Lock Assemblies were reworked and retested at least 2 times. The second retest included a one–cycle thermal extreme test performed to the maximum temperature of 160 degrees F which provides a 10 degrees F thermal margin over the maximum predicted temperature. Thus, all TRRJ Drive/Lock Assembly components have been exposed to thermal temperatures with at least 10 degrees F margin beyond the predicted on–orbit environment.

PG1–240:

ITEM:

SARJ Drive/Lock Assembly Part Number 5847010
Utility Transfer Assembly Part Number 8259150

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3. Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:

A. The maximum predicted flight level and spectrum minus 6 dB, but not less than a level derived from a 135 dB overall acoustic environment (whose spectrum is defined by NSTS–21000–IDD–ISS, paragraph 4.1.1.5).

B. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime contractor.

EXCEPTION:

Acceptance Random Vibration tests on the SARJ Drive/Lock Assemblies and the Utility Transfer Assembly shall be performed to a minimum workmanship screening level of 4.3 grms.

RATIONALE:

The Utility Transfer Assembly and SARJ Drive/Lock Assembly experienced a workmanship screening level of 4.3 grms during Acceptance Random Vibration testing. This is less than the minimum workmanship screening level of 6.1 grms required by SSP 41172 and documented in MDC 95H0215.

While the Utility Transfer Assembly Flight Models 1 and 2 Acceptance Random Vibration tests did not achieve the required input levels, the tests were performed at a level at least 3 dB greater than the maximum predicted flight levels from NASTRAN and SEA models. This does provide confidence in the ability of these units to withstand flight conditions.

The only EEE parts used in the Utility Transfer Assembly are the RTDs, the joint angle resolvers, and the EVA connectors. There are no circuit boards contained in the Utility Transfer Assembly. Therefore, Acceptance Random Vibration Tests are not necessarily the best method to determine material and workmanship defects. Rather, visual inspections during assembly, thermal vacuum testing, and functional testing fulfill these screenings.

During the Test and Verification Control Panel meeting of April 29, 1998, the internal SARJ Drive/Lock Assembly components were reviewed to determine acceptability of the performed random vibration screen. The electrical components in the Drive/Lock Assembly are the drive motor, the commutation resolver, the EDM stepper motor, SSQ electrical connectors, RTDs, the SSQ EVA connectors, and the SARJ Drive/Lock Assembly resistoboxes. Of these, the Panel concurred that the critical component for random vibration workmanship screening is the resistobox. All others were not a concern as they have either been qualified at the vendor or the mass was small enough to not be influenced by vibration environment. Workmanship screening for these components can be accomplished via thermal vacuum, functional tests, and visual inspection.

The resistoboxes did experience a vibration level below the 6.1 grms minimum workmanship screening level during the SARJ Drive/Lock Assembly level testing. SSCN 2341 was previously implemented to certify the acceptance random vibration testing performed for workmanship acceptability.

PG1–241:

ITEM:

Radiator Beam Valve Module Part Number 1F28980–1 and 1F28980–501

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Thermal Vacuum Test shall not be performed on the Radiator Beam Valve Module (RBVM).

RATIONALE:

The purpose of the thermal vacuum test as part of the ATP is to detect material and workmanship flaws which would be induced or exacerbated by exposure to a vacuum environment in combination with temperature extremes.

The major electronic component of the RBVM, the IMCA, undergoes thermal vacuum cycling during its own acceptance testing. During its ATP, the IMCA sees 8 thermal vacuum cycles from –45 degrees F to 140 degrees F, including functional testing. During this testing, the IMCA is monitored for faults. In addition, the IMCA receives 300 hours of burn–in, including electrical and mechanical operation. Other electrical components of the RBVM include the harnesses, temperature sensors, absolute pressure transducer, limit switches, and thermostat controllers. The thermostats, Resistance Temperature Devices, and the IMCA actuator are hermetically sealed. Also, to screen out electrical failures, the ATP performed on the RBVM includes a bonding resistance check and a 1–megohm electrical isolation test. The absolute pressure sensor undergoes thermal vacuum testing at the component level, and the heaters are tested at vacuum while heat–soaked to verify that there are no bubbles or debonding. Finally, the lubricant used in the gearbox and on the geardrive is Braycote, a grease that is low–outgassing under vacuum.

During assembly of the RBVM, the 2–port and 3–port ball valves experience 100 mechanical burn–in cycles of full open–close or open–vent stroke, respectively. This cycling is meant to draw out a mechanical binding failure or any noticeable change in lubricant characteristics that may also have been induced by thermal vacuum testing.

The RBVM assembly does undergo random vibration testing with fault monitoring of the IMCA, and thermal cycling as portions of its performed ATP. In addition, because the RBVM operates in an unpressurized environment, external leakage tests are conducted during acceptance testing, which verifies that the sealing capability of the unit has not been degraded during any of the environments encountered during testing. These several screens of the performed ATP are meant to draw out any failures which thermal vacuum cycling may also have induced, such as electrical intermittencies, latent defective parts, leaking seals or joints, or material outgassing or contamination.

PG1–242:

ITEM:

Heat Exchanger ORU Part Number 1F28940–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.

Paragraph 4.2.5.2. The component shall be mounted to a rigid fixture through the normal mounting point of the component.

Paragraph 4.2.5.3B. Component random vibration test levels and spectrums shall be the envelope of the following:

B. Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The Qualification Random Vibration test level for the Heat Exchanger ORU mounted on the flight support structure shall be the maximum predicted flight and spectrum derived from an acoustic environment of 141 dB (below). These levels do not envelope the acceptance test level plus test tolerances (6.1 grms minimum screening plus 6 dB margin).

Qualification Random Vibration Environment	
Frequency (Hz)	Qualification Level (1)
20	0.004 g^2/Hz
20 – 40	+11.5 dB/Octave
40 – 80	0.056 g^2/Hz
80 – 100	+4.8 dB/Octave
100 – 200	0.08 g^2/Hz
200 – 2000	–5.7 dB/Octave
2000	0.001 g^2/Hz
Composite Level	5.2 grms
Duration	180 seconds/axis
Orientation	Three mutually perpendicular axes
Note: (1) Unit Under Test shall be mounted to the vibration test support strut, TD–1F98740–1ATP1. The qualification vibration levels shall be imposed at the support structure test fixture clevis fittings.	

RATIONALE:

The Heat Exchanger ORUs shall be mounted on the Qualification support structure to demonstrate the Heat Exchanger ORUs ability to withstand the maximum flight random vibration environment. The flight random vibration levels are defined at the Heat Exchanger ORU support structure and USL endcone interface and qualification for flight is performed in the flight configuration. The qualification test at maximum flight levels for 3 minutes per axis will not envelope the minimum acceptance screening test level of 6.1 grms plus 6 dB because the support structure resonant frequencies drive ORU components to vibration levels which exceed their component Qualification test levels. The maximum flight levels were selected in order not to overstress the heat exchanger core and ORU plumbing interfaces as well as the ECUs and valves. In addition, hardmounting the Heat Exchanger ORU on a rigid test fixture was not done to avoid damage to the ORU components. Notching of the input spectra was assessed but not implemented since it would result in testing below the maximum flight random vibration environment. This is determined to be sufficient technically for both qualifying the design for the maximum predicted flight level, as well as for the acceptance random vibration test, while not exposing the unit to unnecessary damage or failure.

PG1–243:

ITEM:

Heat Exchanger ORU Part Number 1F28940–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.

Paragraph 5.1.4.2. The component shall be mounted to a rigid fixture through the normal mounting point of the component.

Paragraph 5.1.4.3B. Component random vibration test levels and spectrums shall be the envelope of the following:

B. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime contractor.

EXCEPTION:

The Acceptance Random Vibration test level for the Flight Heat Exchanger ORUs mounted on the Qualification support structure will be 3 dB below the Qualification test level as described in SSP 41172 exception PG1–242. These levels do not envelope the acceptance test level of 6.1 grms minimum screening.

Acceptance Random Vibration Environment	
Frequency	Acceptance Level (1)
20 Hz	0.002 g^2/Hz
20 – 40 Hz	11.5 dB/Octave
40 – 80 Hz	0.028 g^2/Hz
80 – 100 Hz	4.8 dB/Octave
100 – 200 Hz	0.04 g^2/Hz
200 – 2000 Hz	– 5.70 dB/Octave
2000 Hz	0.0005 g^2/Hz
Overall Level:	3.7 grms
Duration:	60 seconds per axis
Orientation:	Three mutually perpendicular axes

Note:
(1) Unit Under Test shall be mounted to the vibration test support strut, TD–1F98740–1ATP1. The acceptance vibration levels shall be imposed at the support structure test fixture clevis fittings.

RATIONALE:

With respect to the test levels, the requirement for qualification is to envelop the acceptance test level plus test tolerances. This ensures that the qualification test is always equal to, or greater than, the acceptance level at all control frequency bands considering actual test tolerances from one unit test to another. The minimum acceptance test level and spectrum is defined in Figure 5–2 to be a 6.1 grms environment. This is defined as a basic minimum acceptance level (regardless of the predicted flight level) needed to ensure that an adequate environmental stress screen is applied for percipitating workmanship defects. In the case of the heat exchanger, the 6.1 grms minimum workmanship environment exceeds the maximum predicted flight level. Therefore, the acceptance test requirement for the heat exchanger would be the minimum 6.1 grms environment. This would drive the minimum qualification level to something above 6.1 grms (typically 3 dB above, or about 8.6 grms) in order to ”envelope the acceptance level and spectrum plus test tolerances” as required. The 6.1 grms environment has its basis primarily in the screening of electronics boxes. The heat exchanger ORU is primarily fluid equipment with two electronic control boxes in it. The electronics control units received an acceptance random vibration test at the component level prior to being integrated into the ORU. Their component test level actually exceeds the 6.1 grms minimum level and thus, the electronics are adequately screened for workmanship at a lower level of assembly. In order to minimize the risk of unnecessary damage to a good ORU, it was decided that the acceptance test level could be lower than 6.1 grms levels and still provide adequate screening of the ORU assembly (since its critical components were adequately screened at lower assembly levels).

In summary, the concern was unnecessary damage to the ORU by following the strict requirements. It was decided that for the heat exchanger, it could be adequately qualified for flight and screened for workmanship by lowering the acceptance level (thus lowering the required qualification level) and by mounting it to its flight support strut rather than a rigid test fixture. These two actions together, again, were to avoid unnecessary damage or failure of the hardware.

PG1–244:

ITEM:

Heat Exchanger ORU Part Number 1F28940–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.

Paragraph 4.2.5.4. Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

Qualification Random Vibration Testing without Power On and monitoring at the Heat Exchanger ORU Assembly is permitted.

RATIONALE:

Monitoring the Heat Exchanger ORU ECUs for intermittence would require operating NH3 valves which could potentially cause damage. The Heat Exchanger ORU Qualification ECU is electrically energized and monitored for intermittence during its component qualification random vibration test. Therefore, the need to perform Qualification Random Vibration Testing with Power On and Monitoring of the ECUs during the Heat Exchanger ORU Assembly test is mitigated.

PG1–245:

ITEM:

Heat Exchanger ORU Part Number 1F28940–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.

Paragraph 5.1.4.4. Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

Acceptance Random Vibration Testing without Power On and monitoring at the Heat Exchanger ORU Assembly is permitted.

RATIONALE:

Monitoring the Heat Exchanger ORU ECUs for intermittence would require operating NH3 valves which could potentially cause damage. The Flight ECUs are electrically energized and monitored for intermittence during their component acceptance random vibration testing. Therefore, the need to perform Acceptance Random Vibration Testing with Power On and Monitoring of the ECUs during the Heat Exchanger ORU test is mitigated.

PG1–246:

ITEM:

Pump Module ORU Part Number 1F96100

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Description. The test method employed shall have sensitivity and accuracy consistent with the maximum allowed leak rate.

EXCEPTION:

The Pump Module ORU specification leakage requirement shall not be verified by test.

RATIONALE:

ORU leakage testing is performed using the detector probe (“sniffing”) technique; however, the detector probe was not calibrated. In addition, utilization of the detector probe technique only provides a qualitative indication of leakage at the probed locations and cannot verify quantitatively total ORU leakage as defined in the applicable development specification. Also, of the total 48 welds made as part of ORU manufacturing/assembly, one weld is totally inaccessible and one weld is partially inaccessible; therefore, they cannot be fully leak tested after manufacture. Neither of these two welds are fracture critical. These welds experience a proof pressure test to verify their structural integrity.

External leak testing of the Pump Module ORU for qualification verifies design integrity of the ORU assembly. The Pump Module ORU is a welded assembly that contains no seals to permeate through except the Quick Disconnects, which are tested extensively at the component level. Components receive acceptable leak tests at the component–level prior to being assembled into the ORU, so only the ORU welds need to be tested. These welds are orbital tube welds performed by trained welders to strict weld schedules. Experience has shown that "bad" welds are extremely rare, and, in those cases, a gross leak is apparent. Therefore, there is low risk that the Pump Module has any significant leaks.

The ORU was pressurized to 500 psig, and each of the tube welds that were made during the assembly of the ORU (except for the vent lines) were tested for leakage. Pass/fail was determined by the sum of the leakage from all the tube welds and components. During the qualification test program, the ORU was leak tested after Proof Pressure, Acoustic, and Thermal Vacuum testing.

PG1–247:

ITEM:

Pump Module ORU Part Number 1F96100

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Description. The test method employed shall have sensitivity and accuracy consistent with the maximum allowed leak rate.

EXCEPTION:

The Pump Module ORU specification leakage requirement shall not be verified by test.

RATIONALE:

ORU leakage testing is performed using the detector probe ("sniffing") technique; however, the detector probe was not calibrated. In addition, utilization of the detector probe technique only provides a qualitative indication of leakage at the probed locations and cannot verify quantitatively total ORU leakage as defined in the applicable development specification. Also, of the total 48 welds made as part of ORU manufacturing/assembly, one weld is totally inaccessible and one weld is partially inaccessible; therefore, they cannot be fully leak tested after manufacture. Neither of these two welds are fracture critical. These welds experience a proof pressure test to verify their structural integrity.

External leak testing of the Pump Module ORU verifies workmanship of the ORU assembly. The Pump Module ORU is a welded assembly that contains no seals to permeate through except the Quick Disconnects, which are tested extensively at the component level. Components receive acceptable leak tests at the component–level prior to being assembled into the ORU, so only the ORU welds need to be tested. These welds are orbital tube welds performed by trained welders to strict weld schedules. Experience has shown that "bad" welds are extremely rare and, in those cases, a gross leak is apparent. Therefore, there is low risk that the Pump Module has any significant leaks.

The first two flight units have been tested with ammonia as part of the Multiple Element Integration Tests (MEIT), and no measurable ammonia leakage has been observed with these units during MEIT. The remaining units will be tested with an improved detector calibration and leak detection technique described below.

The leakage rate from a known leak source of 9.0E –06 scc per sec of Helium is incorporated in the leak test procedure. This provides a means to compare the leakage rate from each Pump Module ORU test point against a known standard leakage rate base line, and also provides a safety factor of 2 versus the maximum allowable leakage rate. The ORU is pressurized to 500 psig, and each of the tube welds that were made during the assembly of the ORU (except for the vent lines) are tested for leakage. Pass/fail is determined by the leakage measurement of each tube weld where previously pass/fail was determined by the sum of the leakage from all the tube welds and components. In addition, the change incorporates the requirement to use the same leakage probe speed movement across, and distance from, each ORU test point as was used when the probe was moved across the calibrated capillary leak source during calibration.

Because of obstructions, two of the tube welds cannot be fully accessed for leak testing with the leak detector adapter. One of these is situated on the ammonia inlet side of the accumulator such that it is not conducive to testing any part of the weld circumference with the leak detector adapter. This weld joint will be tested by inserting the sniffer probe as close as possible to each weld and monitoring the probe response for 30 minutes. The surrounding architecture directly above these welds is shaped like a dome such that any leakage from the weld will be captured and thus surely detected by the 30–minute probe monitoring. This detection, of course, will not be a calculated value since the calibration reference to the capillary leak source is dependent on distance and scan rate from the weld. If the probe detects the presence of helium in this situation it will reliably indicate leakage and a requirement to replace the tube weld. The remaining tube weld whose weld circumference cannot be fully accessed for leak testing is approximately 75 percent accessible. After testing the exposed weld circumference portions for leakage, the probe will be positioned at the weld/structure interference point and monitored for 2 minutes. In this situation the probe can be positioned directly at the weld; therefore 2 minutes of monitoring will be adequate.

PG1–248:

ITEM:

Ammonia Flow Meter Part Number 1F40070

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3, Test Levels and Duration. The test duration shall be sufficient to detect any significant leakage.

EXCEPTION:

The ammonia flow meter qualification external leak tests were conducted for 5 minutes each.

RATIONALE:

The pressurized portion of the ammonia flow meter (the flow sensor) is a straight section of seamless 316L CRES tubing. There are no seals to permeate through, no penetrations, and no welds. The leak testing performed verifies the integrity of the flow sensor. The flow sensor is bagged and sniffed (i.e., a helium accumulation leak test) using the helium leak detector to determine if a leak is present. A 500–psid pressure differential creates a positive pressure for helium to leak through the tubing. A 5–minute test time is sufficient for the flow sensor design since leaks through the tube will be immediate (no permeable seals). The measured rate during the Qualification Leak testing was 3.8E–07 scc per sec Helium compared to a requirement for 1.32E–04 scc per sec Helium. The intent of the SSP 41172 requirement has been met.

PG1–249:

ITEM:

Ammonia Flow Meter Part Number 1F40070

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3, Test Levels and Duration. The test duration shall be sufficient to detect any significant leakage.

EXCEPTION:

The ammonia flow meter acceptance external leak tests were conducted for 5 minutes each.

RATIONALE:

The pressurized portion of the ammonia flow meter (the flow sensor) is a straight section of seamless 316L CRES tubing. There are no seals to permeate through, no penetrations, and no welds. The leak testing performed verifies the integrity of the flow sensor. The flow sensor is bagged and sniffed (i.e., a helium accumulation leak test) using the helium leak detector to determine if a leak is present. A 500–psid pressure differential creates a positive pressure for helium to leak through the tubing. A 5–minute test time is sufficient for the flow sensor design since leaks through the tube will be immediate (no permeable seals). The maximum measured rate recorded during the Acceptance Leak testing was 6.0E–06 scc per sec Helium compared to a requirement for 1.32E–04 scc per sec Helium. The intent of the SSP 41172 requirement has been met.

PG1–250:

ITEM:

Pump and Control Valve Package Part Number 1F96451

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be no less than 4 hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The PCVP qualification external leak tests were conducted for 20 minutes with an external pressure of less than 0.05 Torr.

RATIONALE:

External leak testing of the PCVP verifies the design and workmanship of the PCVP assembly. The PCVP is a welded assembly which contains no seals to permeate through. The leak testing performed verifies the integrity of the welds in the unit. The leak tests are performed in a vacuum chamber. A 500–psid pressure between the unit and chamber creates a positive pressure for helium to leak through the welds if a “bad” weld exists (no seals to external environment). The chamber pressure of 0.05 Torr is sufficient to bring the Varian Mass Spectrometer Leak Detector on–line to perform the leak test. After a period of 20 to 22 minutes, the external leakage rate is recorded. A 20–minute test time duration is sufficient since leaks through welds will be immediate (no permeable seals). The measured rate during the Qualification Leak testing was 5.4E–07 scc per sec Helium compared to a requirement for 4E–04 scc per sec Helium. Therefore, as seal (weld) integrity will have been verified, the intent of the SSP 41172 requirement is met.

PG1–251:

ITEM:

Pump and Control Valve Package Part Number 1F96451

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be no less than 4 hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The PCVP acceptance external leak tests were conducted for 20 minutes with an external pressure of less than 0.05 Torr.

RATIONALE:

External leak testing of the PCVP verifies the workmanship of the PCVP assembly. The PCVP is a welded assembly which contains no seals to permeate through. The leak testing performed verifies the integrity of the welds in the unit. The leak tests are performed in a vacuum chamber. A 500–psid pressure between the unit and chamber creates a positive pressure for helium to leak through the welds if a “bad” weld exists (no seals to external environment). The chamber pressure of 0.05 Torr is sufficient to bring the Varian Mass Spectrometer Leak Detector on–line to perform the leak test. After a period of 20 to 22 minutes, the external leakage rate is recorded. A 20–minute test time duration is sufficient since leaks through welds will be immediate (no permeable seals). The maximum measured rate recorded during the Acceptance Leak testing was 8.0E–06 scc per sec Helium compared to a requirement for 4E–04 scc per sec Helium. Therefore, as seal (weld) integrity will have been verified, the intent of the SSP 41172 requirement is met.

PG1–252:

ITEM:

Ammonia Tank Assembly Part Number 1F28801–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.
Paragraph 4.2.11.2E, Method V. Leakage shall be detected using an appropriate method.

EXCEPTION:

The Ammonia Tank Assembly specification leakage requirement shall not be verified by test.

RATIONALE:

Since the chosen method of leak detection is the sniffer probe, the total leakage rate around a weld cannot be accurately determined. Only a localized leakage can be determined. A calibration of the sniffer probe shall be performed using a known leak source of 2.2E–06 scc per sec of GHe. This provides a means to compare the leakage rate from each Ammonia Tank Assembly test point against a known standard leakage rate baseline with a factor of safety of two. Each of the 47 tube welds that were made during the assembly of the ORU are tested for leakage using this calibration.

Because of obstructions, 9 of the 47 tube welds cannot be fully accessed for leak testing with the leak detector adapter. Two of these 9 tube welds are situated on the Nitrogen inlet side of the Ammonia tank, such that they are not conducive to testing any part of the weld circumference with the leak detector adapter. These weld joints will be tested by inserting the sniffer probe above each weld and monitoring the leak detector response for 30 minutes. If the probe is too short to be placed above the weld, an additional Tygon tube installed on the probe will be used. The surrounding architecture directly above these welds is shaped like a dome such that any leakage from the weld will be captured, yielding an acceptable confidence level that a leak will be detected by the 30–minute probe monitoring. This detection, of course, will not be a calculated value since the calibration reference to the capillary leak source is dependent on distance and scan rate from the weld. If the probe detects the presence of Helium in this situation, it will reliably indicate leakage and a requirement to replace the tube weld. In the remaining 7 of the 9 tube welds whose weld circumference cannot be fully accessed for leak testing, 4 are approximately 75 percent accessible and 3 are approximately 50 percent accessible. After testing the exposed weld circumference portions of these 7 welds, the probe will be positioned at the weld/structure interference point and monitored for 2 minutes. Two minutes of monitoring will yield an acceptable level of confidence that the inaccessible portion of the weld is not leaking more than the allowable limit.

In this revision, ORU pass–or–fail status is determined by the leakage measurement of each tube weld. Previously, the ORU pass–or–fail status was determined by the sum of the leakage from all the tube welds and components. In addition, this change incorporates the requirement to use the same probe scanning rate across and distance from each Ammonia Tank Assembly test point as was used when the probe was moved across the calibrated capillary leak source during calibration. Since Orbital Tube Welds, by experience, are either good (no leak above Helium background) or bad (gross leak), this revised method yields an acceptable level of confidence in the ORU leakage rate, despite that it is a semiquantitative measurement of ORU leakage.

PG1–253:

ITEM:

Ammonia Tank Assembly Part Number 1F28801–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.

EXCEPTION:

The Ammonia Tank Assembly specification leakage requirement shall not be verified by test. Acceptance of flight hardware will be verified by test via a pass/fail detection criteria.

RATIONALE:

Since the chosen method of leak detection is the sniffer probe, the total leakage rate around a weld cannot be accurately determined. Only a localized leakage can be determined. A calibration of the sniffer probe shall be performed using a known leak source of 2.2E–06 scc per sec of GHe. This provides a means to compare the leakage rate from each Ammonia Tank Assembly test point against a known standard leakage rate baseline with a factor of safety of two. Each of the 47 tube welds that were made during the assembly of the ORU are tested for leakage using this calibration.

Because of obstructions, 9 of the 47 tube welds cannot be fully accessed for leak testing with the leak detector adapter. Two of these 9 tube welds are situated on the Nitrogen inlet side of the Ammonia tank, such that they are not conducive to testing any part of the weld circumference with the leak detector adapter. These weld joints will be tested by inserting the sniffer probe above each weld and monitoring the leak detector response for 30 minutes. If the probe is too short to be placed above the weld, an additional Tygon tube installed on the probe will be used. The surrounding architecture directly above these welds is shaped like a dome such that any leakage from the weld will be captured, yielding an acceptable confidence level that a leak will be detected by the 30–minute probe monitoring. This detection, of course, will not be a calculated value since the calibration reference to the capillary leak source is dependent on distance and scan rate from the weld. If the probe detects the presence of Helium in this situation, it will reliably indicate leakage and a requirement to replace the tube weld. In the remaining 7 of the 9 tube welds whose weld circumference cannot be fully accessed for leak testing, 4 are approximately 75 percent accessible and 3 are approximately 50 percent accessible. After testing the exposed weld circumference portions of these 7 welds, the probe will be positioned at the weld/structure interference point and monitored for 2 minutes. Two minutes of monitoring will yield an acceptable level of confidence that the inaccessible portion of the weld is not leaking more than the allowable limit.

In this revision, ORU pass–or–fail status is determined by the leakage measurement of each tube weld. Previously, the ORU pass–or–fail status was determined by the sum of the leakage from all the tube welds and components. In addition, this change incorporates the requirement to use the same probe scanning rate across and distance from each Ammonia Tank Assembly test point as was used when the probe was moved across the calibrated capillary leak source during calibration. Since Orbital Tube Welds, by experience, are either good (no leak above Helium background) or bad (gross leak), this revised method yields an acceptable level of confidence in the ORU leakage rate, despite that it is a semiquantitative measurement of ORU leakage.

PG1–254:

ITEM:

Solar Array Rotary Joint Trundle Bearing Assembly Part Number 5846485

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.4, Supplementary Requirements. Functional tests shall be conducted, as a minimum, at the maximum operating temperature plus 20 degrees F (11.1 degrees C) and at the minimum operating temperature minus 20 degrees F (11.1 degrees C) during the first and last operating cycle after the dwell and after return of the component to ambient temperature.

EXCEPTION:

The SARJ Trundle Bearing Assembly will not be functionally tested as an EVA replaceable mechanism at thermal extremes during the Qualification Thermal Vacuum Test. Analysis and an engineering demonstration performed on flight–like hardware below the thermal extreme prediction of –31 degrees F is provided in lieu of this requirement.

RATIONALE:

This exception pertains to the Trundle Bearing Assembly as an unloaded mechanism. A review of the design reveals that:

(1) primary functions of this assembly do meet SSP 41172 requirements;
(2) analysis shows that during EVA replacement there are adequate clearances between all moving parts relative to one another;
(3) no greases or other thermal vacuum sensitive lubricants are used;
(4) a flight–like unit (flight unit downgraded due to moisture exposure) passed cold box testing 13 degrees F below Qualification limits of – 51 degrees F;
(5) removal and replacement was demonstrated under a load 52 percent higher than the predicted worst–case condition (320 lbs. compression); and,
(6) All EVA Bolt torques for the acceptance tests were at ambient conditions, and the demonstrations at cold temperatures while under load were within family.

PG1–255:

ITEM:

Solar Array Rotary Joint Trundle Bearing Assembly Part Number 5846485

SSP 41172 REQUIREMENT:

Paragraph 5.1.1, Functional Test, Component Acceptance.
Paragraph 5.1.1.3, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The SARJ Trundle Bearing Assembly will not be functionally tested as an EVA replaceable mechanism at thermal extremes during the Thermal Extreme Test performed in lieu of an Acceptance Thermal Vacuum Test (see PG1–76). Analysis and an engineering demonstration performed on flight–like hardware below the thermal extreme prediction of –31 degrees F is provided in lieu of this requirement.

RATIONALE:

This exception pertains to the Trundle Bearing Assembly as an unloaded mechanism. A review of the design reveals that:

(1) primary functions of this assembly do meet SSP 41172 requirements;
(2) analysis shows that during EVA replacement there are adequate clearances between all parts moving relative to one another;
(3) no greases or other thermal vacuum sensitive lubricants are used;
(4) a flight–like unit (flight unit downgraded due to moisture exposure) passed cold box testing 13 degrees F below Qualification limits (– 51 degrees F);
(5) removal and replacement was demonstrated under a load 52 percent higher than the predicted worst–case condition (320 lbs. compression); and,
(6) All EVA Bolt torques for the acceptance test were at ambient conditions, and the demonstrations at cold temperatures while under load were within family.

PG1–256:

ITEM:

Solar Alpha Rotary Joint Launch Restraint Mechanism Part Number 1F83193–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Solar Alpha Rotary Joint Launch Restraint Mechanism will not undergo a Qualification Thermal Vacuum Test.

RATIONALE:

The SARJ Launch Restraint is a simple clamp device that consists of only a few parts. The primary features are made up of the same materials with the same coefficient of thermal expansion. Due to the same materials used in the SARJ Launch Restraint Clamshell and the Trunnions, the coefficient of thermal expansion and contraction is the same; therefore, no thermally induced binding will occur. Detailed thermal analysis as documented in MDC 0H1298, SARJ Launch Restraint (SLR) Thermal/Tolerance Analysis, mandatory inspection of the primary features to insure drawing conformance, and successful ground installation (proper running torque measurements and final preload along with proper gap measurements) will insure successful on–orbit removal.

PG1–257:

ITEM:

Segment–to–Segment Attach System Striker Assembly Part Number 1F70572–1
Segment–to–Segment Attach System Latch EVA Extension Part Numbers 1F61303–1, 1F70164–1, and 1F61275–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Segment–to–Segment Attach System Striker Assembly and Segment–to–Segment Attach System Latch EVA Extension will not undergo a Qualification Thermal Vacuum Test. This includes thermal extreme testing at qualification levels performed in lieu of thermal vacuum testing.

RATIONALE:

The Striker Assembly is not sensitive to thermal environments due to generous clearances and Aluminum construction. The Latch EVA Extension is not sensitive to cold extremes due to the interfacing parts having similar coefficient of thermal expansion and contraction, and drawing tolerance analysis that show clearance at both cold and hot extremes. Thus, little value is obtained via thermal testing at qualification levels.

PG1–258:

ITEM:

Heat Rejection Subsystem (HRS) Radiators Boeing Part Number 1F40032–1 (LMMFC Part Number 83–39400)

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Qualification external leakage testing shall not require verification by test.

RATIONALE:

Other than the four QDs per radiator which mate the radiator fluid inlet and outlet to flex–hoses, all other fluid line connections on the radiators are welded connections. The welded components of the radiators consist of either solid tubing for manifolds or flow tubes, or flexible metal bellows hoses which connect the radiator panels. There are no seals in any of this hardware. During build of the HRS radiators, several external leakage tests are conducted in order to verify “good” welds. In addition to the incremental build leakage tests, the individual welds at each step are inspected using dye–penetrant per MIL–STD–6866, Type I, Method A, Sensitivity level 3, and x–ray per MIL–STD–453 methods, as well as a proof pressure test before proceeding.

For the base plumbing, the individual radiator tubes, and the flex–hose assemblies, at the incremental build levels there is a helium leak test done with a sniffer probe, with a test requirement limit of 3.0 E–07 scc per sec helium, which may not have been verifiable by the test setup. The leak test is done at 525 +/– 25 psig with 100 percent helium. After completion of assembly, there is an ORU level leakage test performed in the same manner, but sniffing only those joints that are accessible at the ORU level, with a test requirement limit of 3.0 E–07 scc per sec helium. The total ORU allowable leakage rate is 2.0 E–02 scc per sec helium.

It should be noted that since the detection method is the sniffer probe, the total leakage rate of the ORU, or of a particular weld, cannot be accurately determined, but rather only a localized leakage can be detected. This is due to the fact that although the leak detector is capable of detecting a leak in the 3.0 E–07 scc per sec range, detection at this level is only possible if the leak is directly exposed to the vacuum system of the leak detector. This rate is not achievable when accounting for all of the helium dispersive factors that come into play when using a sniffer probe, but rather the detectable limit then falls into the E–04 to E–05 range. Because only a localized rate can be determined, it becomes difficult to relate this number to verification of the overall ORU allowable leakage limit. Only by having an estimated number of welds, and rationing the ORU leakage rate down to the individual joint, can any indication be made that the unit total leakage rate will not be exceeded, provided that the localized limit is still in the detectable range of the sniffing device. With an estimated 500 welds per radiator, to exceed the total ORU allowable leakage rate of the HRS Radiators, every weld would have to be leaking at a rate of approximately 4.0 E–05 scc per sec. This leakage rate is likely to be detectable using this sniffer probe method.

In summary, leakage tests during build were conducted, with a requirement of 3.0 E–07 scc per sec per weld joint, and an actual detectable limit in the E–04 to E–05 range. If any readings of the sniffer probe registered above 3.0 E–07, then the weld in question would be reworked, re–inspected, and re–tested until it passed the individual weld joint test. With all joints in the radiator having a maximum leak rate just below the detectable limit of the sniffer probe (the acceptable test criteria), the average measurable leakage rate is estimated to be below 4.0 E–05 scc per sec per weld. As such, the radiator total leakage rate of 2.0 E–02 scc per sec has been met.

Each ORU also undergoes an ammonia pressure drop test and is pressurized with ammonia to 500 psig for a period of approximately seven days for each independent flow path. The ammonia quick disconnects are not installed during this test due to restrictions on testing the QDs, but all other fluid systems are included in this test. The characteristic ammonia odor was never detected for any of the flight units tested, nor was the lab ammonia detection alarm system activated.

Given the level of inspectability of all of the welds on the ORU during build, as well as the incremental build leakage tests, the welds on the radiator ORU have been shown to be adequate for preventing leakage.

PG1–259:

ITEM:

Heat Rejection Subsystem (HRS) Radiators Boeing Part Number 1F40032–1 (LMMFC Part Number 83–39400)

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Acceptance external leakage testing shall not require verification by test.

RATIONALE:

Other than the four QDs per radiator which mate the radiator fluid inlet and outlet to flex–hoses, all other fluid line connections on the radiators are welded connections. The welded components of the radiators consist of either solid tubing for manifolds or flow tubes, or flexible metal bellows hoses which connect the radiator panels. There are no seals in any of this hardware. During build of the HRS radiators, several external leakage tests are conducted in order to verify good welds. In addition to the incremental build leakage tests, the individual welds at each step are inspected using dye–penetrant per MIL–STD–6866, Type I, Method A, Sensitivity level 3, and x–ray per MIL–STD–453 methods, as well as a proof pressure test before proceeding.

For the base plumbing, the individual radiator tubes, and the flex–hose assemblies, at the incremental build levels, there is a helium leak test done with a sniffer probe, with a test requirement limit of 3.0 E–07 scc per sec helium, which may not have been verifiable by the test setup. The leak test is done at 525 +/– 25 psig with 100 percent helium. After completion of assembly, there is an ORU level leakage test performed in the same manner, but sniffing only those joints that are accessible at the ORU level, with a test requirement limit of 3.0 E–07 scc per sec helium. The total ORU allowable leakage rate is 2.0 E–02 scc per sec helium.

It should be noted that since the detection method is the sniffer probe, the total leakage rate of the ORU, or of a particular weld, cannot be accurately determined, but rather only a localized leakage can be detected. This is due to the fact that although the leak detector is capable of detecting a leak in the 3.0 E–07 scc per sec range, detection at this level is only possible if the leak is directly exposed to the vacuum system of the leak detector. This rate is not achievable when accounting for all of the helium dispersive factors that come into play when using a sniffer probe, but rather the detectable limit then falls into the E–04 to E–05 range. Because only a localized rate can be determined, it becomes difficult to relate this number to verification of the overall ORU allowable leakage limit. Only by having an estimated number of welds, and rationing the ORU leakage rate down to the individual joint, can any indication be made that the unit total leakage rate will not be exceeded, provided that the localized limit is still in the detectable range of the sniffing device. With an estimated 500 welds per radiator, to exceed the total ORU allowable leakage rate of the HRS Radiators, every weld would have to be leaking at a rate of approximately 4.0 E–05 scc per sec. This leakage rate is likely to be detectable using this sniffer probe method.

In summary, leakage tests during build were conducted, with a requirement of 3.0 E–07 scc per sec per weld joint, and an actual detectable limit in the E–04 to E–05 range. If any readings of the sniffer probe registered above 3.0 E–07, then the weld in question would be reworked, re–inspected, and re–tested until it passed the individual weld joint test. With all joints in the radiator having a maximum leak rate just below the detectable limit of the sniffer probe (the acceptable test criteria), the average measurable leakage rate is estimated to be below 4.0 E–05 scc per sec per weld. As such, the radiator total leakage rate of 2.0 E–02 scc per sec has been met.

Each ORU also undergoes an ammonia pressure drop test and is pressurized with ammonia to 500 psig for a period of approximately seven days for each independent flow path. The ammonia quick disconnects are not installed during this test due to restrictions on testing the QDs, but all other fluid systems are included in this test. The characteristic ammonia odor was never detected for any of the flight units tested, nor was the lab ammonia detection alarm system activated.

Given the level of inspectability of all of the welds on the ORU during build, as well as the incremental build leakage tests, the welds on the radiator ORU have been shown to be adequate for preventing leakage.

PG1–260:

ITEM:

Flexhose Rotary Coupler ORU (FHRC) Part Number 5839202–501

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When there is no dedicated qualification test article and all production articles are intended for flight usage, the test shall be the same…" This requires that all fluid or propulsion equipment including the Flexhose Rotary Coupler ORU to undergo a protoflight thermal cycle test in accordance with 6.1.1B.

EXCEPTION:

The FHRC ORUs shall not experience protoflight thermal cycle testing.

RATIONALE:

FHRC ORU Serial Numbers 1001 and 1002 underwent thermal cycle testing at the FHRC ORU assembly level and thermal vacuum testing at the Thermal Radiator Rotary Joint (TRRJ) assembly level. This Design, Development, Test, and Evaluation (DDT&E) test approach was previously approved via exception PG1–07. Since there will be no TRRJ–level thermal vacuum test for the spare FHRCs (Serial Numbers 1003 and 1004), a protoflight thermal vacuum test has been added to the FHRC ORU level test program. Future thermal vacuum testing of any FHRC ORU shall occur at the FHRC assembly level.

The purpose of thermal cycle testing is to screen electrical and electronic components for latent manufacturing defects. The only electronics of concern in the FHRC ORU are within the Power Data Transfer Assembly (PDTA) built and tested by Honeywell. The PDTA component goes through Acceptance Thermal Cycle testing across a greater operational temperature range than the current FHRC test plan (–55 degrees F to 150 degrees F at the PDTA level, versus –55 degrees F to 105 degrees F at the FHRC level). This PDTA–level acceptance thermal cycle test is in compliance with SSP 41172 test requirements and is sufficient to screen the FHRC electronics for workmanship. Performing thermal cycle testing at the FHRC ORU level of assembly would add little, if any, additional workmanship screening benefits, and the Program cost and schedule impacts of performing the test at this level of assembly are not technically warranted. Thermal Vacuum testing at the FHRC ORU assembly level is sufficient to ensure that the ORU assembly will function as required at the appropriate temperature extremes.

PG1–261:

ITEM:

Radiator Beam Valve Module (RBVM) Part Number 1F28980 (Honeywell Part Number 3750098)

SSP 41172 REQUIREMENT:

Paragraph 5.1.6, Pressure Test, Component Acceptance.
Paragraph 5.1.6.3, Test Levels. ALL REQUIREMENTS.

EXCEPTION:

Acceptance proof pressure verification of the internal Radiator Beam Valve Module ball valves shall not be required by test.

RATIONALE:

Per Specification Control Drawing 1F28980, the RBVM maximum operating pressure is 500 psi. The flight unit ORUs were proof pressure tested to 800 +40 to –0 psi to provide the required factor of safety of 1.5 in accordance with SSP 41172 and SSP 30559. However, subsequent analysis and testing resulting from a RBVM Functional Configuration Audit/Physical Configuration Audit issue have identified that from a line pressure of 500 psi, the RBVM 2–port and 3–port valve cavities can reach pressures of 950 psi, with shaft seal cavities reaching pressures up to 1620 psi. Thus, the valves have not experienced a proof pressure test to the required safety factor at these locations.

The valve cavities of the 2–port and 3–port RBVM valves will experience a maximum operational pressure of 950 psi when the RBVM inlet ammonia line pressure is at 500 psi due to fluid becoming trapped between the ball seals of the valve, and then experiencing thermal transients. In addition, the two shaft seals of each valve have a similar possibility of fluid entrapment. Between these two seals, assuming that some leakage occurs (the smallest leak measured during flight unit ATP was used in calculations for analysis), a maximum operating pressure of 1620 psi will be reached. Because none of the flight units were tested under this transient condition to proof levels with a factor of safety of 1.5, a burst test was conducted on a qualification unit for risk mitigation with pressure applied as applicable to the valve cavity. (Pressure could not be directly applied to the area between shaft seals.)

This qualification unit was taken to an internal valve fluid pressure of 5000 psi as limited by test equipment, and a 5–minute dwell was held. After pressure was relieved from the unit, it was noted to have permanent deformation in the form of bulging cavity and port walls. However, no fluid escaped from the shaft seals of the valve.

Structural analysis of the valve housing material indicates that although there is a fairly low yield strength, the ultimate pressure capability of the valve will allow for deformation to accumulate before any rupture occurs. By setting the ultimate strength at the established 5000 psi level obtained, and scaling the ultimate strength to the yield strength of the housing material, it was determined that the test results demonstrate that the valves have a design capability of operating up to 1187 psi, from a proof–pressure perspective. This value exceeds, for the main valve cavity, the established maximum operating pressure of 950 psi, thus indicating that this design is capable of withstanding these valve cavity pressures.

Similarly, for the shaft seal area, a structural analysis was conducted to determine its burst and proof capabilities. Since the same mechanical properties apply as for the housing, it was determined that yield strength is the limiting factor. However, since the valve housing is robust around the shaft seal area, analysis calculated that this area of the valve is capable of operating to a maximum pressure greater than 5700 psi (based on yield strength). This is in excess of the calculated maximum operating pressure of 1620 psi in the shaft seal area of the valve. As such, the valve has acceptable strength for these pressures.

Thus, the RBVM may be exempted from an acceptance proof pressure test in accordance with program requirements at the noted locations.

PG1–262:

ITEM:

Radiator Beam Valve Module (RBVM) Part Number 1F28980 (Honeywell Part Number 3750098)

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Qualification leakage verification of the internal Radiator Beam Valve Module ball valves shall not be performed with the component pressurized at the maximum operating pressure.

RATIONALE:

Per Specification Control Drawing 1F28980, the RBVM maximum operating pressure is 500 psia. The flight units assemblies were leakage tested to 500 psia at lower level testing, and 450 +20 to –10 psia during ATP, to test in accordance with SSP 41172. However, subsequent analysis and testing resulting from a RBVM Functional Configuration Audit/Physical Configuration Audit issue have identified that from a line pressure of 500 psi, thermal transients in the RBVM 2–port and 3–port valve cavities will cause pressures of 950 psi, with shaft seal cavities reaching pressures up to 1620 psi. Thus, the valves have not experienced a leakage test to the required pressure at these locations.

In the main valve cavity, increased fluid pressure will result in a decrease of the interface pressure of the ball seals, tending to increase their leakage in the reverse direction, which is a desired effect in this condition. For the 3–port valve in the supply line, this increased cavity pressure will increase the interface pressure of the vent port seal, tending to decrease its leakage to space vacuum, and minimizing the leakage concern of additional pressure on this seal. The effect of increased ball seal leakage in the reverse direction is to limit the maximum pressure obtained in the valve cavity, as verified by testing on qualification units. This testing, conducted as reverse pressurization leakage testing on 4 ball valves, indicated that at a maximum reverse differential pressure of 450 psid, an individual ball seal would leak at a rate of 1.0 E–1 scc per sec of Helium. With a MOP of 500 psi, this creates a new maximum pressure within the valve of 950 psi. Note that with two ball seals in parallel, the weakest seal in the reverse direction will be the one to control maximum cavity pressure. Since this situation creates a reverse differential pressure across the ball seals, and they have only been tested for leakage in the forward direction, minimal data is available regarding their leakage characteristics in the forward direction after experiencing reverse pressure cycling. Additional forward leakage past the ball seals will not be detrimental to the system operational performance, and the likelihood that additional seal leakage will result in loss of ammonia from the system is small. This was determined by inspection of the number of seals in various valve positions that would have to have increased leakage before ammonia loss (above specification) is noted. In addition, the seal vendor has indicated that the pressure differential cycling that the seals will experience is not likely to change their forward sealing leakage capability.

The two shaft seals in series in each valve will have a pressure buildup between them from thermal transients that results in a maximum of 1620 psi in the shaft seal cavity. This number was determined analytically, and was derived assuming some reverse leakage back into the ball valve cavity, as well as external leakage to the RBVM ORU internal area. It should be noted that this case cannot be tested on the flight units, as there is no access to pressure the shaft seal cavity area in the expected operating modes and inspect for leakage. Higher shaft seal cavity pressure will tend to decrease the sealing capability of the primary shaft seal as it relieves in the reverse direction, and increase the capability of the secondary shaft seal in sealing to space vacuum. With no testing conducted in which the pressure in the shaft seal cavity was known, leakage characteristics of the shaft seals after exposure to the expected pressure differentials is unknown. Again, though, the seal vendor has indicated that the pressure differential cycling that the seals will experience is not likely to change their forward sealing leakage capability. So, the worst case effect of pressure buildup is additional external ammonia leakage above specification levels. If the external leakage does increase, it will not affect system operations other than a quicker than expected loss of ammonia inventory.

Thus, the RBVM may be exempted from qualification leakage testing at the maximum operating pressure within the valve cavities in accordance with SSP 41172 requirements.

PG1–263:

ITEM:

Radiator Beam Valve Module (RBVM) Part Number 1F28980 (Honeywell Part Number 3750098)

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Acceptance leakage verification of the internal Radiator Beam Valve Module ball valves shall not be performed with the component pressurized at the maximum operating pressure.

RATIONALE:

Per Specification Control Drawing 1F28980, the RBVM maximum operating pressure is 500 psia. The flight units assemblies were leakage tested to 500 psia at lower level testing, and 450 +20 to –10 psia during ATP, to test in accordance with SSP 41172. However, subsequent analysis and testing resulting from a RBVM Functional Configuration Audit/Physical Configuration Audit issue have identified that from a line pressure of 500 psi, thermal transients in the RBVM 2–port and 3–port valve cavities will cause pressures of 950 psi, with shaft seal cavities reaching pressures up to 1620 psi. Thus, the valves have not experienced a leakage test to the required pressure at these locations.

In the main valve cavity, increased fluid pressure will result in a decrease of the interface pressure of the ball seals, tending to increase their leakage in the reverse direction, which is a desired effect in this condition. For the 3–port valve in the supply line, this increased cavity pressure will increase the interface pressure of the vent port seal, tending to decrease its leakage to space vacuum, and minimizing the leakage concern of additional pressure on this seal. The effect of increased ball seal leakage in the reverse direction is to limit the maximum pressure obtained in the valve cavity, as verified by testing on qualification units. This testing, conducted as reverse pressurization leakage testing on 4 ball valves, indicated that at a maximum reverse differential pressure of 450 psid, an individual ball seal would leak at a rate of 1.0E–1 scc per sec Helium. With a MOP of 500 psi, this creates a new maximum pressure within the valve of 950 psi. Note that with two ball seals in parallel, the weakest seal in the reverse direction will be the one to control maximum cavity pressure. Since this situation creates a reverse differential pressure across the ball seals, and they have only been tested for leakage in the forward direction, minimal data is available regarding their leakage characteristics in the forward direction after experiencing reverse pressure cycling. Additional forward leakage past the ball seals will not be detrimental to the system operational performance, and the likelihood that additional seal leakage will result in loss of ammonia from the system is small. This was determined by inspection of the number of seals in various valve positions that would have to have increased leakage before ammonia loss (above specification) is noted. In addition, the seal vendor has indicated that the pressure differential cycling that the seals will experience is not likely to change their forward sealing leakage capability.

The two shaft seals in series in each valve will have a pressure buildup between them from thermal transients that results in a maximum of 1620 psi in the shaft seal cavity. This number was determined analytically, and was derived assuming some reverse leakage back into the ball valve cavity, as well as external leakage to the RBVM ORU internal area. It should be noted that this case cannot be tested on the flight units, as there is no access to pressure the shaft seal cavity area in the expected operating modes and inspect for leakage. Higher shaft seal cavity pressure will tend to decrease the sealing capability of the primary shaft seal as it relieves in the reverse direction, and increase the capability of the secondary shaft seal in sealing to space vacuum. With no testing conducted in which the pressure in the shaft seal cavity was known, leakage characteristics of the shaft seals after exposure to the expected pressure differentials is unknown. Again, though, the seal vendor has indicated that the pressure differential cycling that the seals will experience is not likely to change their forward sealing leakage capability. So, the worst case effect of pressure buildup is additional external ammonia leakage above specification levels. If the external leakage does increase, it will not affect system operations other than a quicker than expected loss of ammonia inventory.

Thus, the RBVM may be exempted from acceptance leakage testing at the maximum operating pressure within the valve cavities in accordance with SSP 41172 requirements.

PG1–264:

ITEM:

Nitrogen Tank Part Number 1F96201

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be no less than 4 hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The Nitrogen tank qualification external leakage tests were conducted for 30 minutes each.

RATIONALE:

The Tank is a welded assembly (with a composite overbraid for strength) which contains no seals for leaks to permeate through. The Leak testing performed verifies the integrity of the welds in tanks and was performed in a Vacuum chamber. The specification requirement (1F96201, paragraph 3.2.1.2) is not greater than 1E–06 scc per sec of Helium at 3000 psid. The tanks were pressurized with 3000 psig of Helium and the chamber is evacuated to 1E–04 torr. The external leak test was performed three times during the qualification testing (after Pressurized Volume (proof pressure), Random Vibration, and Pressure Cycling) for 30 minutes for each test. SSP 41172, paragraph 4.2.11.2 requires leak checks performed prior to initiation of and following the completion of component qualification thermal and vibration. The stabilized readings recorded were on the order of 3.2E–09 scc per sec of Helium.

A 30 minute test time is sufficient since leaks through welds will be practically immediate (no permeable seals). The Helium Mass Spectrometer Leak Detector was monitored for the 30 minutes for indications of leakage. Thus, the intent of SSP 41172 requirement (reference paragraph 4.2.11.3, ”The test duration shall be sufficient to detect any significant leakage”) has been met as the seal (weld) integrity has been verified.

Comparative leak rates of similar tanks made using the same construction methods were found to be similar to those found for the ISS tanks manufactured by Lincoln Composites. Because the methods used to process the tank through manufacturing caused the filament–wound tank to be crazed, permeability is not an issue through the filament overwrap, since leaks will be detected quickly. Leaks that have occurred in similarly designed tanks were detected immediately. Also, although the leak test was for a duration of 30 minutes, in order to reach the leak check pressure of 3000 psig Helium, the tank pressurization process takes a few hours to allow for cooling of the Helium gas during pressurization. Therefore, the tank will experience pressurized gas for a period of time much longer than 30 minutes, providing a longer duration for actual leakage to occur.

PG1–265:

ITEM:

Nitrogen Tank Part Number 1F96201

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be no less than 4 hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The Nitrogen tank acceptance external leak test was conducted for 30 minutes.

RATIONALE:

The Tank is a welded assembly (with a composite overbraid for strength), which contains no seals for leaks to permeate through. The Leak testing performed verifies the integrity of the welds in tanks and was performed in a Vacuum chamber. The specification requirement (1F96201, paragraph 3.2.1.2) is not greater than 1E–06 scc per sec of Helium at 3000 psid. The tanks were pressurized with 3000 psig of Helium and the chamber is evacuated to 1E–04 torr. The external leak test was performed once during the acceptance testing for 30 minutes. The stabilized readings recorded were on the order of 8E–08 scc per sec of Helium.

A 30 minute test time is sufficient since leaks through welds will be practically immediate (no permeable seals). The Helium Mass Spectrometer Leak Detector was monitored for the 30 minutes for indications of leak. Thus, the intent of SSP 41172 requirement (reference paragraph 4.2.11.3, ”The test duration shall be sufficient to detect any significant leakage”) has been met as seal (weld) integrity has been verified.

Comparative leak rates of similar tanks made using the same construction methods were found to be similar to those found for the ISS tanks manufactured by Lincoln Composites. Because the methods used to process the tank through manufacturing caused the filament–wound tank to be crazed, permeability is not an issue through the filament overwrap, since leaks will be detected quickly. Leaks that have occurred in similarly designed tanks were detected immediately. Also, although the leak test was for a duration of 30 minutes, in order to reach the leak check pressure of 3000 psig Helium, the tank pressurization process takes a few hours to allow for cooling of the Helium gas during pressurization. Therefore, the tank will experience pressurized gas for a period of time much longer than 30 minutes, providing a longer duration for actual leakage to occur.

PG1–266:

ITEM:

Nitrogen Tank ORU Part Number 1F96000

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Description. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.

EXCEPTION:

The Nitrogen Tank ORU specification leakage requirement shall not be verified by test.

RATIONALE:

ORU leakage testing is performed using the detector probe (“sniffing”) technique; however, the detector probe was not calibrated. In addition, utilization of the detector probe technique only provides a qualitative indication of leakage at the probed locations and cannot verify quantitatively total ORU leakage as defined in the applicable development specification. Also, of the 28 welds made as part of the ORU manufacturing/assembly, two welds are only 75 percent accessible. However, these welds are located on the vent line and do not need to be leak checked. All welds do experience a proof pressure test to verify their structural integrity.

External leak testing of the Nitrogen Tank ORU for qualification verifies design integrity of the ORU assembly. The Nitrogen Tank ORU is a welded assembly that contains no seals to permeate through except the Quick Disconnects, which are tested extensively at the component level. Components receive acceptable leak tests at the component–level prior to being assembled into the ORU, so only the ORU welds need to be tested. These welds are orbital tube welds performed by trained welders to strict weld schedules. Experience has shown that "bad" welds are extremely rare and, in those cases, a gross leak is apparent. Therefore, there is low risk that the Nitrogen Tank ORU has any significant leaks.

The ORU was pressurized to 3000 psia on the high pressure side and 390 psia on the low pressure side, then each of the tube welds that were made during the assembly of the ORU (except for the vent lines) were tested for leakage. Pass/fail was determined by the sum of the leakage from all the tube welds and components. During the qualification test program, the ORU was leak tested after Proof Pressure, Acoustic, and Thermal Vacuum testing.

PG1–267:

ITEM:

Nitrogen Tank ORU Part Number 1F96000

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Description. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.

EXCEPTION:

The Nitrogen Tank ORU specification leakage requirement shall not be verified by test.

RATIONALE:

ORU leakage testing is performed using the detector probe ("sniffing") technique; however, the detector probe was not calibrated. In addition, utilization of the detector probe technique only provides a qualitative indication of leakage at the probed locations and cannot verify quantitatively total ORU leakage as defined in the applicable development specification. Also, of the 28 welds made as part of the ORU manufacturing/assembly, two welds are only 75 percent accessible. However, these welds are located on the vent line and do not need to be leak checked. All welds do experience a proof pressure test to verify their structural integrity.

External leak testing of the Nitrogen Tank ORU verifies workmanship of the ORU assembly. The Nitrogen Tank ORU is a welded assembly that contains no seals to permeate through except the Quick Disconnects, which are tested extensively at the component level. Components receive acceptable leak tests at the component–level prior to being assembled into the ORU, so only the ORU welds need to be tested. These welds are orbital tube welds performed by trained welders to strict weld schedules. Experience has shown that "bad" welds are extremely rare and, in those cases, a gross leak is apparent. Therefore, there is low risk that the Nitrogen Tank ORU has any significant leaks.

The first two flight units have been tested as part of the Multiple Element Integration Tests and no measurable Nitrogen leakage has been observed with these units during performed testing. The remaining units will be tested with an improved detector calibration and leak detection technique described below.

The leakage rate from a known leak source equal to or less than 1.5E–05 scc per sec Helium is incorporated in the leak test procedure. This provides a means to compare the leakage rate from each Nitrogen Tank Assembly test point against a known standard leakage rate baseline. The ORU is pressurized to 3000 psia on the high pressure side and 390 psia on the low pressure side, and each of the tube welds that were made during the assembly of the ORU (except for the vent lines) are tested for leakage. Pass/fail is determined by the leakage measurement of each tube weld where previously pass/fail was determined by the sum of the leakage from all the tube welds and components. In addition, the change incorporates the requirement to use the same leakage probe speed movement across, and distance from, each ORU test point as was used when the probe was moved across the calibrated capillary leak source during calibration.

PG1–268:

ITEM:

Gas Pressure Regulator Valve Part Number 1F40058

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the components specified maximum allowable leak rate.
Paragraph 4.2.11.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be no less than 4 hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The Gas Pressure Regulator Valve qualification external leakage tests were conducted for 30 minutes each.

The Gas Pressure Regulator Valve port leakage tests (inlet to outlet, vent, and relief) were conducted for 5 minutes.

RATIONALE:

The Gas Pressure Regulator Valve is an all–welded assembly with no seals for leaks to permeate through. For external leak testing, the inlet, outlet, and vent ports are pressurized and the relief port is capped. No seals are verified in this test. The external leak testing performed verifies the integrity of the welds in the assembly and is performed in a Vacuum chamber. The specification requirement (1F40058, paragraph 3.2.1.4) is not greater than 1E–06 scc per sec of Helium at 3000 psid. The assembly was pressurized with 3000 psig of Helium and the chamber was evacuated to 1E–04 torr. The external leak test was performed 3 times during the qualification testing for 30 minutes each test.

A 30 minute test time is sufficient since leaks through welds will be practically immediate (no permeable seals). The Helium Mass Spectrometer Leak Detector was monitored for the entire 30 minutes for indications of leak. Thus, the intent of SSP 41172 requirement (reference paragraph 4.2.11.3, ”The test duration shall be sufficient to detect any significant leakage”) has been met as seal (weld) integrity has been verified.

Port leakage tests were performed 10 times during the qualification test, following shock, vibration, and valve actuation, and several times during thermal cycling. The specification requirements (1F40058, paragraph 3.2.1.4) are not greater than 2E–04 scc per sec of Helium for the vent port and relief port and not greater than 2E–03 scc per sec of Helium inlet to outlet. These tests were performed each time for five minutes. The maximum leak rate measured during the qualification program was on the order of 1.8E–06 scc per sec Helium. Development leak tests were performed early in the program by Allied Signal to verify valve seat leakage. Vespel seats were specifically selected for their low leakage characteristics. The five–minute duration was determined to be adequate to establish the leakage rate. The duration is certainly adequate to show that there is no gross leakage problem. Thus, the intent of SSP 41172 requirement (reference. paragraph 4.2.11.3 "The test duration shall be sufficient to detect any significant leakage") has been met.

On orbit, once the Active Thermal Control System is initiated, the Gas Pressure Regulator Valve is isolated from the nitrogen tank on one side and from the ammonia tank on the other side. Even if leakage was an order of magnitude higher than specified from the Gas Pressure Regulator Valve vent and relief, only a small volume of nitrogen would be lost.

PG1–269:

ITEM:

Gas Pressure Regulator Valve Part Number 1F40058

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the components specified maximum allowable leak rate.
Paragraph 5.1.7.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be no less than 4 hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The Gas Pressure Regulator Valve acceptance external leakage tests were conducted for 30 minutes each.

The Gas Pressure Regulator Valve port leakage tests (inlet to outlet, vent, and relief) were conducted for 5 minutes.

RATIONALE:

The Gas Pressure Regulator Valve is an all–welded assembly with no seals for leaks to permeate through. For external leak testing, the inlet, outlet, and vent ports are pressurized and the relief port is capped. The external leak testing performed verifies the integrity of the welds in the assembly and is performed in a Vacuum chamber. The specification requirement (1F40058, paragraph 3.2.1.4) is not greater than 1E–06 scc per sec of Helium at 3000 psid. The assemblies were pressurized with 3000 psig of Helium and the chamber is evacuated to 1E–04 torr. The external leak test was performed twice during the acceptance testing, after proof pressure and as a final test for 30 minutes each test. The stabilized readings recorded for flight unit Serial Number 103 were less than 3.2E–07 scc per sec of Helium.

A 30 minute test time is sufficient since leaks through welds will be practically immediate (no permeable seals). The Helium Mass Spectrometer Leak Detector was monitored for the entire 30 minutes for indications of leak. The intent of SSP 41172 requirement (reference. paragraph 5.1.7.3, "The test duration shall be sufficient to detect any significant leakage") has been met as seal (weld) integrity has been verified.

Port leakage tests were performed several times during the acceptance test, before and after random vibration and thermal cycling testing as well as at the end of testing. The specification requirements (1F40058, paragraph 3.2.1.4) are not greater than 2E–04 scc per sec of Helium for the vent port and relief port and not greater than 2E–03 scc per sec of Helium inlet to outlet. The final leakage readings for flight unit Serial Number 103 were 4.2E–07, 3.8E–07, and 2.4E–04 scc per sec of Helium, respectively. These tests were performed each time for 5 minutes. Development leak tests were performed early in the program by Allied Signal to verify valve seat leakage. Vespel seats were specifically selected for their low leakage characteristics. The 5–minute duration was determined to be adequate to establish the leakage rate. The duration is certainly adequate to show that there is no gross leakage problem. Thus, the intent of SSP 41172 requirement (reference. paragraph 5.1.7.3 "The test duration shall be sufficient to detect any significant leakage") has been met.

On orbit, once the Active Thermal Control System is initiated, the Gas Pressure Regulator Valve is isolated from the nitrogen tank on one side and from the ammonia tank on the other side. Even if leakage was an order of magnitude higher than specified from the Gas Pressure Regulator Valve vent and relief, only a small volume of nitrogen would be lost.

PG1–270:

ITEM:

Unpressurized Cargo Carrier Attach System (UCCAS) Part Number 1F70156–1 (hinges)
Payload Attach System (PAS) Part Number 1F70157–1 (hinges)

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Thermal Vacuum Test will not be performed on the hinges for the UCCAS and PAS mechanical assemblies.

RATIONALE:

A qualification thermal vacuum test will not be performed on the hinges for the UCCAS and PAS mechanical assemblies. Instead, a thermal analysis will be performed in lieu of testing. The hinges are a simple hinge joint that allows the UCCAS and PAS platforms to rotate during deployment. During assembly, the UCCAS and PAS structural assemblies are centered with respect to each other and the hinges. Therefore, at ambient temperature, a nominal position is attained with no appreciable tolerances. Rotation of the assemblies at ambient temperature is verified during assembly when each UCCAS/PAS platform is articulated between the stowed and deployed positions.

A detailed thermal analysis as documented in Memorandum A3–J092–AAM–M–0100764, Revision A, was performed on the ability of the system to deploy over the qualification thermal environment of –140 degrees F to + 160 degrees F. The results of the thermal analysis show that a minimum 0.001 to 0.005 inches of net diametric clearance exists at the extreme tolerance range for the maximum bolt versus minimum hole dimensions with regards to the Yoke/Platform and Platform/Longeron hinges, respectively. Thermal effects are negligible since both the bolt and associated bushings are made of compatible or identical materials. In the axial direction, a minimum 0.068 to 0.005 inches of clearance exists at the extreme tolerance range for the same hinges. Again, the thermal effects are negligible since the components are made of compatible or identical materials. A positive clearance of 0.032 inches is also shown for effects due to misalignment between the two hinges.

The omission of qualification thermal vacuum testing is acceptable because the referenced analysis shows that the UCCAS and PAS platforms are found to be deployable over the qualification thermal range of –140 degrees F to + 160 degrees F with the hinge rotation points having positive diametric and axial clearances. In addition, the as–built data that provides additional margin for axial clearances and required deployment of the assemblies in ambient conditions provides significant additional risk reduction.

PG1–271:

ITEM:

On–orbit Restraint Assembly Part Number 1F82080–1 (Left–Handed) and 1F82088–1 (Right–Handed)

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same…” This requires protoflight functional, thermal vacuum, and random vibration testing to be performed.

EXCEPTION:

The On–orbit Restraint will not experience Protoflight Functional, Thermal Vacuum, or Random Vibration Testing.

RATIONALE:

The On–orbit Restraint is a “simple” moving mechanical assembly comprised of nine captive EVA bolts and one On–orbit adjustable interface. Design margin has been demonstrated by conservative thermal expansion and contraction, strength, and tolerance analysis related to all operational requirements using worst–case environments.

Performance is established, and workmanship defects screened, by a comprehensive verification approach relying on hardware demonstration, analysis, and inspection. This includes (1) strength and vibro–acoustic analysis with a factor of safety of 2.0 documented in MDC02H0987, (2) On–orbit environmental combined mechanical and thermal tolerance analysis for fit and function documented in MDC02H1009, (3) as–built inspections of critical design features that govern performance (used in analysis report MDC02H1009), and (4) fit checks of two of the four On–orbit Restraints to the ITS P1 flight element as described in MDC02H0999.

PG1–272:

ITEM:

Adjustable Grapple Bar Part Number 1F82020–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same…" This requires protoflight functional, thermal vacuum, and random vibration testing to be performed.

EXCEPTION:

The Adjustable Grapple Bar will not experience Protoflight Functional, Thermal Vacuum, or Random Vibration Testing.

RATIONALE:

The Adjustable Grapple Bar is a "simple" moving mechanical assembly comprised of a sliding strut, plunger locking pin, and single EVA bolt used for controlling gap when installed on an ORU. Design margin has been demonstrated by conservative thermal expansion and contraction, strength, and tolerance analysis related to all operational requirements using worst–case environments.

Performance is established, and workmanship defects screened, by a comprehensive verification approach relying on hardware demonstration, analysis, and inspection. This includes (1) strength and vibro–acoustic analysis with a factor of safety of 2.0 documented in MDC02H0987, (2) On–orbit environmental combined mechanical and thermal tolerance analysis for fit and function documented in MDC02H1004, (3) as–built inspections of critical design features that govern performance, and (4) fit checks of the Adjustable Grapple Bar to one spare of each ORU that uses an Adjustable Grapple Bar for On–orbit translation as described in MDC02H1000 and MDC02H1001.

APPENDIX B PG–2 APPROVED EXCEPTIONS

The following is a list of exceptions to this document taken by Product Group 2. The exceptions to this document in no way eliminates the Contractor’s responsibility for showing compliance to the sections 3.2 through 3.7 of the applicable specification.

PG2–02a:

ITEM:

Structural Certification of PG–2 Hardware

SSP 41172 REQUIREMENT:

Structural Testing.

DEVIATION:

DDP–145 Component Acceptance Test

RATIONALE:

DDP–145

PG2–02b:

ITEM:

Protoflight Beta Gimbal Assembly Qualification unit

SSP 41172 REQUIREMENT:

Protoflight Testing.

DEVIATION:

DDP–146

RATIONALE:

DDP–146

PG2–02d:

ITEM:

Qualification Tests of Remote Power Control Modules (RPCM)

SSP 41172 REQUIREMENT:

Qualification Testing.

DEVIATION:

Delete Two Qualification Tests of RPCMs

RATIONALE:

DDP–172

PG2–02e:

ITEM:

Functional Tests of Wings 3 – 8

SSP 41172 REQUIREMENT:

Acceptance Testing.

DEVIATION:

Reduce Functional Tests of Wings 3 – 8

RATIONALE:

DDP–173

PG2–02f:

ITEM:

Tests of Wings 3 – 8

SSP 41172 REQUIREMENT:

Acceptance Testing.

DEVIATION:

Delete Thermal Vacuum Tests of Wings 3 – 8

RATIONALE:

DDP–174

PG2–03:

ITEM:

Rocketdyne Radiator

SSP 41172 REQUIREMENT:

Table 5–1, Random Vibration.

DEVIATION:

Vibration Acceptance Testing of the Rocketdyne Radiator is waived.

RATIONALE:

Originally approved by NASA Space Station Freedom Program. Note (9) of Table 5–1 was written to waive the Vibration Acceptance Testing of the Rocketdyne Radiator.

PG2–06:

ITEM:

Radiator Part Number RE1894

SSP 41172 REQUIREMENT:

Paragragh 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.

EXCEPTION:

During Qualification Thermal Vacuum and Thermal Cycling Testing, the Radiator may experience less than 20 degrees F thermal margin beyond the acceptance thermal testing.

RATIONALE:

Since items containing fluids may experience freezing conditions, 20 degrees F margin may not be accomplished during qualification testing.

PG2–08:

ITEM:

PFCS

SSP 41172 REQUIREMENT:

Paragraphs 4.2.5.4 and 5.1.4.4, Supplementary Requirements.

DEVIATION:

The PFCS electrical and electronic parts will not be electrically energized and monitored during qualification and acceptance random vibration tests.

RATIONALE:

The electronic boxes within the PFCS will be electrically energized and monitored during separate random vibration tests.

PG2–09:
(This exception has been superceded by PG2–89)

ITEM:

Rocketdyne Radiator

SSP 41172 REQUIREMENT:

Paragraphs 4.2.2, Thermal Vacuum Test; 4.2.3, Thermal Cycling; and 5.1.2, Thermal Vacuum Test, Component Acceptance, and Figure 4–1, Component Thermal/Vacuum Qualification Test.

DEVIATION:

The complete Rocketdyne Radiator will not undergo Thermal Vacuum (T/V) and Thermal Cycling (T/C).

RATIONALE:

The complete Rocketdyne Radiators will be not be Thermal Vacuum and Thermal Cycling tested. Fluids will be flowed through the Radiator to conduct the thermal test.

PG2–11:

ITEM:

Beta Gimbal Bearing/Motor Roller Ring Module (BMRRM)

SSP 41172 REQUIREMENT:

Table 5–1, Component Acceptance Tests, Thermal Vacuum of Moving Mechanical Assembly, and Electrical/Electronic Assembly.

DEVIATION:

The Beta Gimbal BMRRM units 3–8 will not be Thermal Vacuum Acceptance Tested. BMRRM does not contain any Printed Wiring Boards or classic electronics, only connectors and cables.

RATIONALE:

BMRRM moving parts do not have close tolerances and can be effectively inspected. Electrical parts are environmentally tested at the component level.

PG2–13:

ITEM:

Beta Gimbal Assembly

SSP 41172 REQUIREMENT:

Paragraphs 4.3.1.4, Supplementary Requirements (redlined out of draft SSP 41172), and 6.1, Use of Qualification Assemblies For Flight (Protoflight).

DEVIATION:

Since the Beta Gimbal is fully qualified except for structural test, subsequent flight units will not be protoflight tested structurally and are considered qualified by initial qualification testing.

RATIONALE:

Beta Gimbal per DDP 146 Protoflight Beta Gimbal Assembly Qualification unit will undergo protoflight static structural testing per SSP 30559. All other environmental testing will be to full qualification test levels and subsequent flight units will not be protoflight structural tested.

PG2–14:

ITEM:

IEA

SSP 41172 REQUIREMENT:

Paragraphs 4.3.1.4, Supplementary Requirements (redlined out of draft SSP 41172), and 6.1, Use of Qualification Assemblies For Flight (Protoflight)

DEVIATION:

Since the IEA is fully qualified except for structural test, subsequent flight units will not be protoflight tested structurally and are considered qualified by initial qualification testing.

RATIONALE:

IEA DDP 145 Structural Certification of PG–2 Hardware unit will undergo protoflight static structural testing per SSP 30559. All other environmental testing will be to full qualification test levels and subsequent flight units will not be protoflight structural tested.

PG2–15:

ITEM:

Bearing/Motor Roller Ring Module

SSP 41172 REQUIREMENT:

Paragraphs 4.2.5.4 and 5.1.4.4, Supplementary Requirements.

DEVIATION:

The BMRRM electrical parts will not be electrically energized and monitored during the qualification and acceptance random vibration tests.

RATIONALE:

There are no electronic boxes within the BMRRM.

PG2–19:

ITEM:

All hardware subject to Acceptance Minimum Component Random Vibration Workmanship Screening Test Level.

SSP 41172 REQUIREMENT:

Figure 5–2, Component Random Vibration Workmanship Screening Test Level

DEVIATION:

Reduction of Test Levels at the 20 Hz and 2000 Hz frequency range may be required on a case by case basis.

RATIONALE:

Minimum Component Random Vibration Workmanship Screening Test Levels are too high at the 20 Hz and 2000 Hz frequency range and may require a reduction on a case by case basis. Testing equipment will have difficulty maintaining control at the extreme frequency ranges depending on the weight of the test fixture and test item.

PG2–21:
(This exception has been superceded by PG2–89)

ITEM:

Radiator

SSP 41172 REQUIREMENT:

Table 5–1, Component Acceptance Tests, Thermal Vacuum.

DEVIATION:

No Thermal Vacuum Acceptance Testing of Radiator.

RATIONALE:

Table 5–1, Component Acceptance Tests, Thermal Vacuum, Note (6).

PG2–23:

ITEM:

Fluid Quick Disconnect Coupling (FQDC)

SSP 41172 REQUIREMENT:

Table 5–1 and Figure 5–1.

DEVIATION:

Acceptance Thermal Cycling tests are not performed on the FQDC.

RATIONALE:

Clarification of Requirement

PG2–24:

ITEM:

Fluid Quick Disconnect Coupling

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.

DEVIATION:

FQDC Thermal Vacuum Acceptance Test will be at a less vacuum and temperature transition rate than required.

RATIONALE:

Subcontractors equipment limitations.

PG2–25:

ITEM:

Batteries

SSP 41172 REQUIREMENT:

Figures 4–1 and 5–1, Component Thermal Tests Qualification and Acceptance.

DEVIATION:

Thermal Vacuum and Thermal Cycling does not include temperatures margins

RATIONALE:

Battery limitations.

PG2–27:

ITEM:

Corona monitoring

SSP 41172 REQUIREMENT:

Paragraphs 4.2.2.1, Thermal Vacuum Test, Component Qualification, and 5.1.2.2, Thermal Vacuum Test, Component Acceptance.

DEVIATION:

It is assumed that the test requirements to monitor for arcing and corona and to assure that multipacting does not occur can be tailored to use an alternate technique.

RATIONALE:

Corona monitoring during Thermal Vacuum Testing. Clarification of need to tailor requirement to avoid the impact of repetitive testing and test equipment cost.

PG2–28:

ITEM:

Solar Array Wing (SAW)

SSP 41172 REQUIREMENT:

Paragraph 4.2.12, Electromagnetic Compatibility Test, Component Qualification.

DEVIATION:

The radiated emissions from the SAW shall be allowed to exceed the RE02 requirements of SSP 30237.

RATIONALE:

The Motor Drive Assembly (MDA) is more efficient than the specification requires. Requiring the MDA to meet this requirement would result in a less efficient system. The SAW is on its own bus; therefore, it can not disrupt other systems with the radiation.

PG2–29:

ITEM:

Qualification Baterry Charge/Discharge Unit (BCDU) Battery system for EMC Qualification

SSP 41172 REQUIREMENT:

Table 4–1, Component Qualification Test.

DEVIATION:

The BCDU/Battery system EMC qualification will be conducted using two Engineering Model (EM) BCDUs and Qualification Battery ORUs.

RATIONALE:

The fidelity of the EM BCDUs is sufficient to enable verification of the overall EMI/EMC requirements.

PG2–30:

ITEM:

Battery Thermal Vacuum Acceptance Testing – 3 Qualification Units and 48 Flight Units

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance, and Table 5–1, Component Acceptance Tests.

DEVIATION:

Eliminate thermal vacuum testing for Battery ORUs QM–02 (third Qualification Battery ORUs thru FM–48).

RATIONALE:

The thermal vacuum testing conducted on QM–00 and QM–01 will demonstrate compliance with performance requirements for ORUs populated with Semi–Automatic Functional Test and EPI Battery cells. Battery ORU units QM–02 thru QM–04 will undergo thermal vacuum testing while installed in the Qualification testing of Battery ORUs. All qualification and flight Battery Signal Conditioner and Control Modules (BSCCM) will undergo thermal cycling testing and thermal vacuum prior to installation in the Battery ORUs.

PG2–31:

ITEM:

Photovoltaic Radiator (PVR)

SSP 41172 REQUIREMENT:

Qualification and Acceptance Test Matrixes.

DEVIATION:

Delete the Thermal/Vacuum testing of stowed PVR ORUs. This includes Qualification unit and three flight units. Note: Plumbrook testing of Flight Unit 1 will perform these tests.

RATIONALE:

The PVR meets test requirements for electrical systems at the component level prior to installation. Analysis of results from the Plumbrook T/V. T/C testing on Flight Unit 1 will detect workmanship and material defects.

PG2–32:

ITEM:

PFCS Random Vibration Acceptance Testing

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Component Acceptance Test Matrix.

DEVIATION:

Delete Random Vibration Acceptance Testing for Qualification and Flight Units

RATIONALE:

Several Hamilton Standard subsystems for the National Space Transportation System are accepted without random vibration test and no recordable nonconformance has been traced to workmanship errors detectable by vibration testing. All electronic components are subjected to random vibration testing prior to installation in the PFCS.

PG2–33:

ITEM:

PFCS Response Limiting Test Fixture

SSP 41172 REQUIREMENT:

Random Vibration Test.

DEVIATION:

Delete requirement to use Response Limiting test fixture.

RATIONALE:

Test fixture is not required (will use notching).

PG2–34:

ITEM:

PFCS Thermal Cycling Acceptance Test

SSP 41172 REQUIREMENT:

Thermal Cycling Test, Component Acceptance.

DEVIATION:

Delete 4 of the 8 required Thermal Cycling Test for Flight/Qualification and Flight Units.

RATIONALE:

The first and eight Thermal Cycles are the most stringent. These cycles and two additional cycles provide adequate screening for the PFCS flight units.

PG2–36:

ITEM:

Solar Array Wing

SSP 41172 REQUIREMENT:

Component Acceptance Test, Acoustic Vibration.

DEVIATION:

The SAW will not be acoustic acceptance tested after Wings 1 and 2.

RATIONALE:

Most components are vibration or acoustic tested at the component level before the wing level acoustic test. The components not tested at a lower level are structure that is easily inspected.

PG2–37:

ITEM:

Solar Array Wing

SSP 41172 REQUIREMENT:

Component Acceptance Test, Note 1.

DEVIATION:

The SAW mast/canister functional test will not be performed between the environmental acceptance tests. A functional test will be performed prior to the first environmental test and following the last environmental test.

RATIONALE:

Regardless of where any damage would occur, the defect would be discovered and corrected prior to flight.

PG2–38:

ITEM:

Battery Thermal Vacuum Test Qualification and Acceptance

SSP 41172 REQUIREMENT:

Thermal Vacuum testing: paragraphs 4.2.2, Thermal Vacuum Test, Component Qualification, and 5.1.2, Thermal Vacuum Test, Component Acceptance.

DEVIATION:

The Qualification requirement for 140 degrees F delta and 100 degrees F acceptance test requirement will not be accomplished.

RATIONALE:

The qualification test will encompass four cycles, three of which will be nonoperating –13 degrees F to + 86 degrees F, the fourth cycle will be conducted with the battery operating +32 degrees F to +50 degrees F. The acceptance test will be one cycle +7 degrees F to +66 degrees F. Battery Temperatures are limited by design of the battery cells.

PG2–39:

ITEM:

Solar Array Wing Motor Drive Assembly Part Number 5843318

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperatures during qualification thermal vacuum and thermal cycle testing under nonoperating conditions for the Solar Array Wing Motor Drive Assembly were –67 degrees F. The test provides 0 degrees F qualification margin beyond the as–tested minimum nonoperating conditions during acceptance testing.

RATIONALE:

The lack of demonstrated qualification thermal margin for nonoperating temperatures is minimized through:

A. Numerous qualification tests of space–quality electronics to –65 degrees F (operational) indicate temperatures of this level do not degrade these types of electronics.

B. A flight fidelity MDA was subjected to a cold survival test to the following conditions (reference Tecstar Report 2990243, MDA Thermal Survival Test), with 2–hour soaks at each temperature under the following conditions:

   (1) One cycle from ambient to –67 degrees F

   (2) One cycle from ambient to –87 degrees F

   (3) One cycle from ambient to –108 degrees F

   (4) Six cycles from ambient to –128 degrees F

After each cycle of exposure to these temperature extremes, the flight–fidelity MDA passed an ambient functional test. Additionally, there is a manual EVA backup means for deployment/retraction of MDA–driven SAW mechanisms in the event that the MDA electronics fail.

PG2–40:

ITEM:

Solar Array Wing Motor Drive Assembly Part Number 5843318

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperature during qualification thermal vacuum and thermal cycle testing under operating conditions for the Solar Array Wing Motor Drive Assembly was –67 degrees F. The test provides 18 degrees F qualification margin beyond the as–tested minimum operating condition during acceptance testing.

The maximum temperature during qualification thermal vacuum and thermal cycle testing under nonoperating conditions for the Solar Array Wing Motor Drive Assembly was +185 degrees F. The test provides 18 degrees F qualification margin beyond the as–tested maximum nonoperating condition during acceptance testing.

The maximum temperature during qualification thermal vacuum and thermal cycle testing under operating conditions for the Solar Array Wing Motor Drive Assembly was +86 degrees F. The test provides 18 degrees F qualification margin beyond the as–tested maximum operating condition during acceptance testing documented in PG2–137.

RATIONALE:

Cognizant NASA, Boeing, and Lockheed Martin engineers consider the 2 degrees F non–compliance on qualification margin to be inconsequential particularly in view of MDA thermal test data. Review of the MDA thermal test data indicates a capability to control test tolerances to a greater fidelity than the SSP 41172–required 5.4 degrees F (typically, less than 3.6 degrees F). This ability to control actual test temperature tolerances to a tighter fidelity supplements the exception of a reduced qualification margin of 18 degrees F.

PG2–41:

ITEM:

Solar Array Wing Motor Drive Assembly Part Number 5843318

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3B. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance levels during the cold portion of the cycle.

EXCEPTION:

The minimum temperature during acceptance thermal vacuum testing under operating conditions for the Solar Array Wing Motor Drive Assembly was –49 degrees F.

RATIONALE:

The SAW was designed to Space Station Freedom environments and its components were accepted to temperature ranges computed for the Freedom environment. The International Space Station environment is more severe than the Freedom environment and the on–orbit temperature of several components exceeds their acceptance test temperature ranges. The SAW MDAs are expected to reach more severe temperatures (Space Station Revision D+ analysis predicted a minimum operating thermal environment of – 80 degrees F and a minimum nonoperating thermal environment of – 94 degrees F) than the temperature range to which they were accepted. If the SAW MDAs do not function after exposure to on–orbit cold temperature, then an EVA tool will be used to manually retract and deploy the SAW.

Additionally, a flight fidelity MDA was subjected to a cold survival test to the following conditions (reference Tecstar Report 2990243, MDA Thermal Survival Test), with 2–hour soaks at each temperature under the following conditions:

A. One cycle from ambient to –67 degrees F

B. One cycle from ambient to –87 degrees F

C. One cycle from ambient to –108 degrees F

D. Six cycles from ambient to –128 degrees F

After each cycle of exposure to these temperature extremes, the flight–fidelity MDA passed an ambient functional test.

PG2–42:

ITEM:

Solar Array Wing Motor Drive Assembly Part Number 5843318

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.3B. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

The minimum temperature during acceptance thermal cycle testing under operating conditions for the Solar Array Wing Motor Drive Assembly was –49 degrees F.

RATIONALE:

The SAW was designed to Space Station Freedom environments and its components were accepted to temperature ranges computed for the Freedom environment. The International Space Station environment is more severe than the Freedom environment and the on–orbit temperature of several components exceeds their acceptance test temperature ranges. The SAW MDAs are expected to reach more severe temperatures (Space Station Revision D+ analysis predicted a minimum operating thermal environment of – 80 degrees F and a minimum nonoperating thermal environment of – 94 degrees F) than the temperature range to which they were accepted. If the SAW MDAs do not function after exposure to on–orbit cold temperature, then an EVA tool will be used to manually retract and deploy the SAW.

Additionally, a flight fidelity MDA was subjected to a cold survival test to the following conditions (reference Tecstar Report 2990243, MDA Thermal Survival Test), with 2–hour soaks at each temperature under the following conditions:

A. One cycle from ambient to –67 degrees F

B. One cycle from ambient to –87 degrees F

C. One cycle from ambient to –108 degrees F

D. Six cycles from ambient to –128 degrees F

After each cycle of exposure to these temperature extremes, the flight–fidelity MDA passed an ambient functional test.

PG2–43:

ITEM:

SAW Mast Canisters Part Number 5818235: Serial Numbers 541FLT004, 541FLT005, 541FLT006, 541FLT007, 541FLT008, 541FLT009, and 541FLT010

SSP 41172 REQUIREMENT:

Paragraph 5.1.3.3B. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

Acceptance thermal cycle temperature test levels for the SAW Mast Canisters will be 142 +/–5 degrees F (hot) and –92 +/–5 degrees F (cold).

RATIONALE:

SSP 41172 component thermal cycle acceptance test requires that the component be subjected to the maximum and minimum predicted service temperatures. The mast canisters were designed for environmental temperatures predicted for the Space Station Freedom Program. New thermal predictions indicate temperature extremes outside those predicted for Freedom. The mast canisters have been successfully acceptance thermal cycling tested to the Freedom thermal predictions of 142 +/–5 degrees F (hot) and –92 +/–5 degrees F (cold). The new International Space Station thermal environments are 193 +/–5 degrees F (hot) and –127 +/–5 degrees F (cold). Mast canisters already delivered will not be acceptance thermal cycle tested to the International Space Station temperature extremes at the component level.

The primary purpose for acceptance testing at the component level is to screen the hardware for workmanship and to demonstrate acceptable performance at the service environment extremes prior to delivery or installation of the hardware into a higher level of assembly where failure would have greater cost and schedule impacts. Since some of the mast canisters are already delivered, no significant benefit would be realized by rescreening at the component level. The delivered mast canisters will be subjected to protoflight temperature extremes (maximum and minimum predicted +/– 10 degrees F margin) during the wing level protoflight tests on Solar Arrays 1 and 3. This will be sufficient to demonstrate adequate workmanship and performance at the service environment extremes for these units prior to flight.

AEC Able, the Mast Canister Manufacturer, has performed tests on the Bray 601 lubricates to –144 degrees F (Mast Roller Thermal Vacuum Test, Report Number AEC – 93518R904). The test demonstrated that the lubricant worked, and no excessive torque, grease migration, or other unfavorable characteristics were observed on the Mast Canister rollers. The Freedom requirement (28 degrees F) was the basis of the Lockheed–Martin Missiles and Space (LMMS) analysis. The initial review (meet or exceed) assessment did not show significant departures. Only during a more in–depth analysis did the problem surface. The lubricant test is just one of a series of reviews, analyses, and tests that AEC Able is going to perform to insure the hardware. LMMS will be incorporating that information into their verification paperwork.

AEC Able is performing flex batten component verification to the new temperature (–101 degrees F to +170 degrees F). Mast Canister Flight Unit 8, Serial Number 541FLT011, will be protoflight (maximum and minimum predicted +/– 10 degrees F margin) tested to the ISS thermal environments of 203 +/–5 degrees F (hot) and –137 +/–5 degrees F (cold).

PG2–44:

ITEM:

Battery Charge/Discharge Unit Part Number RE1807

SSP 41172B REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.

(1) Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The Qualification Random Vibration Test for the Battery Charge/Discharge Unit shall be performed to the levels indicated as follows:

Frequency Range (Hz)	Power Spectral Density
20	0.02 g^2/Hz
20–45	3.0 dB/Octave
45–100	–3.9 dB/Octave
100–140	0.016 g^2/Hz
140–200	13.6 dB/Octave
200–500	0.08 g^2/Hz
500–2000	–3.8 dB/Octave
2000	0.014 g^2/Hz
Composite	8.72 grms

RATIONALE:

Reduced input in frequency bands that have high box resonance and increased input in frequency bands where there is attenuation inside the ORU. Reduced risk of exceeding component structural margins while maintaining an adequate level of qualification testing.

PG2–45:

ITEM:

Battery Charge/Discharge Unit Part Number RE1807

SSP 41172B REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance

(1) Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The Acceptance Random Vibration Test for the Battery Charge/Discharge Unit shall be performed to the levels indicated as follows:

Frequency Range (Hz)	Power Spectral Density
20	0.01 g^2/Hz
20–45	3.0 dB/Octave
45–100	–3.9 dB/Octave
100–140	0.008 g^2/Hz
140–200	15.5 dB/Octave
200–500	0.05 g^2/Hz
500–2000	–4.3 dB/Octave
2000	0.007 g^2/Hz
Composite	6.65 grms

RATIONALE:

Reduced input in frequency bands that have high box resonance and increased input in frequency bands where there is attenuation inside the ORU. Reduced risk of exceeding component structural margins while maintaining an adequate level of acceptance testing.

PG2–46:

ITEMS:

Remote Power Controller Module, Type I Part Number R077416
Remote Power Controller Module, Type II Part Number R077417
Remote Power Controller Module, Type III Part Number R077418
Remote Power Controller Module, Type IV Part Number R072702
Remote Power Controller Module, Type V Part Number R077419
Remote Power Controller Module, Type VI Part Number R077420

SSP 41172B REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.

(1) Paragraph 5.1.2.2, Test Description. The components that are required to operate during ascent, descent, and depressurization/repressurization shall be operating and monitored for arcing and corona during the initial reduction of pressure to the specified lowest levels.

EXCEPTION:

The RPCMs will undergo depressurization without power applied and without monitoring for corona discharge during the Acceptance Thermal Vacuum Test.

RATIONALE:

Physical analysis (supported by testing of the first flight RPCM, Type V Part Number R077419 Serial Number X751150 on April 16, 1997) has determined that corona discharge cannot occur in a Type V RPCM during nominal or transient operations. Therefore, the subject testing is not needed for production unit acceptance.

PG2–47:

ITEMS:

Power Supply A through E Part Number R075730
Switchgear Control Assembly Part Number R072526
Voltage Divider Relay Driver Assembly Part Numbers R078224 and R078226
Remote Power Controller Module, Type I Part Number R077416
Remote Power Controller Module, Type II Part Number R077417
Remote Power Controller Module, Type III Part Number R077418
Remote Power Controller Module, Type IV Part Number R072702
Remote Power Controller Module, Type V Part Number R077419
Remote Power Controller Module, Type VI Part Number R077420

SSP 41172B REQUIREMENT:

Paragraph 5.1.8, Burn–In Test, Component Acceptance

(1) Paragraph 5.1.8.3C, Duration. The total operating time for electronic and electrical component burn–in shall be 300 hours, including the operating time during any testing while operating.

EXCEPTION:

The duration of the Component Acceptance Burn–In Test shall be modified by the accelerated burn–in formula indicated. This formula allows a combination of room temperature burn–in hours to be combined with accelerated burn–in hours (accomplished at a higher temperature) to produce a “burn–in equivalent” of 300 hours.

For example, an accelerated burn–in at 115 degrees F provides an acceleration factor of 5.83. Thus, every hour of operation at 115 degrees F is equivalent to 5.83 hours of testing at 72 degrees F.

RATIONALE:

The accelerated burn–in can be characterized by:

$$F = \exp[(Ea/K)(1/Ta - 1/Tbi)]$$

where
F = Acceleration Factor
E = Activation energy (eV) (0.3 to 1.2 typical)
K = Boltzmann’s constant = 8.625E–05 eV/K
Tn = Device temperature during normal operation (degrees K)
Tbi = Device temperature during burn–in (degrees K)

For an activation energy of 0.6 eV (selected based on the makeup of the electronic components utilized) and an accelerated burn–in temperature of 115 degrees F, an acceleration factor of 5.83 is obtained. In other words, every hour of operation at 115 degrees F is equivalent to 5.83 hours of testing at 72 degrees F.

Burn–in temperatures may be tailored for each item indicated.

PG2–48:

ITEM:

Battery ORU Part Number RE1804

SSP 41172 REQUIREMENT:

Paragraph 5.1.4.3: Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 5.1.4.3B: A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime Contractor.

EXCEPTION:

The Battery ORU will be tested to an Acceptance Random Vibration Spectrum of 3.9 grms.

RATIONALE:

The Battery ORU receives an adequate workmanship screen for its integrated components via ORU– and module–level acceptance testing. These includes 12–cycle Thermal Cycle performance verification, 6–cycle Orbital Rate Capacity Test, 6–cycle Charge Retention Test, Reference Capacity Test, and the Acceptance Random Vibration screen at the ORU level, and a 16–cycle Thermal Cycle performance verification and 1–cycle Thermal Vacuum performance verification at the BSCCM module level.

The BSCCM is the only electrical/electronic component integrated into the Battery ORU. Acceptance Random Vibration screening at this level would have limited benefit as the construction of the BSCCM, with 2 Circuit Card Assemblies fastened to a webbed case at 12 locations, would not be sufficiently excited to precipitate workmanship defects. A failure of the BSCCM does not cause the Battery ORU to be nonoperable; the Battery ORU continues degraded mode operation with lost of ORU temperature and cell pressure data. BCDU functionality continues using redundant Battery ORU.

PG2–49:

ITEMS:

Remote Power Controller Module, Type I Part Number R077416
Remote Power Controller Module, Type II Part Number R077417
Remote Power Controller Module, Type III Part Number R077418
Remote Power Controller Module, Type IV Part Number R072702
Remote Power Controller Module, Type V Part Number R077419
Remote Power Controller Module, Type VI Part Number R077420

SSP 41172 REQUIREMENT:

Paragraph 5.1.4.3: Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 5.1.4.3B: A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime Contractor.

EXCEPTION:

RPCM Acceptance Random Vibration Spectrum does not meet SSP 41172 minimum workmanship criteria below 30 Hertz and greater than 500 Hertz in the X and Y axes. This exception will allow RPCM Acceptance Random Vibration Spectrum for the X and Y axes as follows:

ACCEPTANCE (5.9 grms)	
20 Hz	.00625 G^2/Hz
20–64 Hz	+6.0 dB/Oct
64–366 Hz	.0625 G^2/Hz
366–2,000 Hz	–7.5 dB/Oct
2,000 Hz	0.0009 G^2/Hz

RATIONALE:

Significant RPCM response frequencies are in the frequency range where the RPCM Acceptance Random Vibration Spectrum exceeds the minimum SSP 41172 Workmanship Criteria; therefore, the RPCMs received acceptable workmanship screening. There are no significant RPCM response modes at the frequencies where the noncompliances exist; and therefore, the risk of undetected workmanship defects is considered minimal.

The natural frequencies lie at 150 Hertz, 473 Hertz, 673 Hertz, and 1262 Hertz. Workmanship screening exceeding the minimum requirements in Figure 5–2 was accomplished for the primary and secondary frequencies (150 Hertz and 473 Hertz). The energy levels above 500 Hertz provide an adequate workmanship screen for the design due to the fact that the fundamental response modes at the remaining natural frequencies (673 Hertz and 1262 Hertz) are not critical.

PG2–50:

ITEM:

Battery Charge/Discharge Unit Part Number RE1807

SSP 41172 REQUIREMENT:

Paragraph 4.2.3.3, Test Levels and Duration. The transitions shall be at a rate no less than 1.0 degree F (0.56 degrees C) per minute.

EXCEPTION:

The temperature transition rate from hot–to–cold shall be no less than 0.4 degree F per minute.

RATIONALE:

As permitted by SSP 41172, Qualification Thermal Cycling test was completed under Thermal Vacuum conditions. However, the conductance of the flight fin plate limited the maximum ramp rate to 0.6 degree F per minute. The test method did provide adequate workmanship ramp rates as the temperature transition from cold–to–hot did exceed 1 degree F per minute. High power circuits experienced temperature transition rates exceeding 1.6 degrees F minute due to self–heating. Additionally, the high power circuits experienced a temperature sweep of 260 degrees F from hot–to–cold and the lower dissipation circuits experienced a temperature sweep of 140 degrees F.

PG2–51:

ITEM:

Battery Charge/Discharge Unit Part Number RE1807

SSP 41172 REQUIREMENT:

Paragraph 5.1.3.3, Test Levels and Duration. The transitions between low and high temperatures shall be at a rate no less than 1.0 degree F (0.56 degrees C) per minute.

EXCEPTION:

The temperature transition rate from hot–to–cold shall be no less than 0.4 degree F per minute.

RATIONALE:

As permitted by SSP 41172, Acceptance Thermal Cycling test was completed under Thermal Vacuum conditions. However, the conductance of the flight fin plate limited the maximum ramp rate to 0.6 degree F per minute. The test method did provide adequate workmanship ramp rates as the temperature transition from cold–to–hot did exceed 1 degree F per minute. High power circuits experienced temperature transition rates exceeding 1.6 degrees F minute due to self–heating. Additionally, the high power circuits experienced a temperature sweep of 260 degrees F from hot–to–cold and the lower dissipation circuits experienced a temperature sweep of 140 degrees F.

PG2–52:

ITEM:

Sequential Shunt Unit Part Number RE1806

SSP 41172 REQUIREMENT:

Paragraph 4.2.3.3C, Test Levels and Duration. The transitions shall be at a rate no less than 1.0 degree F (0.56 degrees C) per minute.

EXCEPTION:

The thermal ramp rate during thermal cycle hot–to–cold transitions shall be at a rate no less than 0.5 degrees F per minute.

RATIONALE:

Thermal cycle testing of the Sequential Shunt Unit is performed in the Subcontractor thermal vacuum test chamber. The testing is dependent on the radiative cooling of the shrouds in the thermal vacuum chamber. However, the physical properties of these shrouds limit the ramp rate on hot–to–cold transition to no greater than 0.6 degrees F per minute.

The Sequential Shunt Unit did experience thermal ramp rates from 1.0 to 1.6 degrees F per minute during every thermal cycle cold–to–hot transition. The unit withstood a temperature sweep of approximately 200 degrees F (to maximize the hot–to–cold thermal transitions) which far exceeds the minimum thermal sweep. Therefore, the test method optimizes the thermal ramp rate of the unit given the facility's limitation and is an adequate workmanship screen.

PG2–53:

ITEM:

Sequential Shunt Unit Part Number RE1806

SSP 41172 REQUIREMENT:

Paragraph 5.1.3.3C, Test Levels and Duration. The transitions shall be at a rate no less than 1.0 degree F (0.56 degrees C) per minute.

EXCEPTION:

The thermal ramp rate during thermal cycle hot–to–cold transitions shall be at a rate no less than 0.5 degrees F per minute.

RATIONALE:

Thermal cycle testing of the Sequential Shunt Unit is performed in the Subcontractor thermal vacuum test chamber. The testing is dependent on the radiative cooling of the shrouds in the thermal vacuum chamber. However, the physical properties of these shrouds limit the ramp rate on hot–to–cold transition to no greater than 0.6 degrees F per minute.

The Sequential Shunt Unit did experience thermal ramp rates from 1.0 to 1.6 degrees F per minute during every thermal cycle cold–to–hot transition. The unit withstood a temperature sweep of approximately 200 degrees F (to maximize the hot–to–cold thermal transitions) which far exceeds the minimum thermal sweep. Therefore, the test method optimizes the thermal ramp rate of the unit given the facility's limitation and is an adequate workmanship screen.

PG2–54:

ITEM:

Solar Array Wing Motor Drive Assembly Part Number 5843317

SSP 41172 REQUIREMENT:

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

The SAW MDA shall not be electrically energized and monitored during the qualification random vibration test.

RATIONALE:

The Solar Array Wing Motor Drive Assembly contains a fail safe brake that will be released when power is applied under the current design. Applying power during the random vibration test will put the motor in a nonflight configuration during the test and potentially cause damage to the motor. This is not a desired condition during qualification random vibration testing. These motors are not powered–on and functioned during the flight vibration environments. Additionally, the MDAs have a limited on–orbit life. The Mast Canister MDAs have an on–orbit life of 10 cycles each for a maximum of 26 minutes (13 minutes to deploy and 13 minutes to retract). The PV Blanket and Containment Box MDAs have an on–orbit life of 10 cycles each for a maximum of 1 minute 20 seconds (20 seconds to unlatch, 20 seconds to tension, 20 seconds to untension and 20 seconds to relatch). Thus, the additional risk associated with this exception is limited.

PG2–55:

ITEM:

Solar Array Wing Motor Drive Assembly Part Number 5843317

SSP 41172 REQUIREMENT:

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

The SAW MDA shall not be electrically energized and monitored during the acceptance random vibration test.

RATIONALE:

The SAW MDA contains a fail safe brake that will be released when power is applied under the current design. Applying power during the random vibration test will put the motor in a nonflight configuration during the test and potentially cause damage to the motor. This is not a desired condition during acceptance random vibration testing. These motors are not powered–on and functioned during the flight vibration environments. Additionally, the MDAs have a limited on–orbit life. The Mast Canister MDAs have an on–orbit life of 10 cycles each for a maximum of 26 minutes (13 minutes to deploy and 13 minutes to retract). The PV Blanket and Containment Box MDAs have an on–orbit life of 10 cycles each for a maximum of 1 minute 20 seconds (20 seconds to unlatch, 20 seconds to tension, 20 seconds to untension and 20 seconds to relatch). Thus, the additional risk associated with this exception is limited.

PG2–56:

ITEMS:

Remote Power Controller Module, Type I Part Number R077416
Remote Power Controller Module, Type II Part Number R077417
Remote Power Controller Module, Type III Part Number R077418
Remote Power Controller Module, Type IV Part Number R072702
Remote Power Controller Module, Type V Part Number R077419
Remote Power Controller Module, Type VI Part Number R077420
Sequential Shunt Unit Part Number RE1806
Battery Charge/Discharge Unit Part Number RE1807
Electronic Control Unit - Part Number R072341
Plasma Contactor Unit - Part Number R078480
Power Electronic Unit - Part Number R076855
DC Switching Unit - Part Number R072610
DC/DC Converter Unit Internal Part Number R076500
DC/DC Converter Unit External Part Number R076522
DC/DC Converter Unit High Power Part Number R079903
Main Bus Switching Unit Part Number R072591
Local Data Interface - Part Number R072491
Initialization Diode Assembly - Part Number R078486
Switchgear Control Assembly - Part Number R072526
Power Supply A/E Part Number R075730

SSP 41172 REQUIREMENT:

Paragraph 5.1.3.3, Test Levels and Duration. Ambient pressure shall be used.

EXCEPTION:

Acceptance Thermal Cycling may be conducted at less than ambient pressure.

RATIONALE:

Thermal Cycling at reduced pressure is at least or more rigorous than testing conducted at ambient pressure as less air density decreases heat transfer through convection, resulting in a more stressful thermal test. Also, Qualification Thermal Cycle Testing is permitted at less than ambient pressure. Note: The minimum temperature transition rate of 1 degree F per minute is required.

PG2–57:

ITEMS:

Remote Power Controller Module, Type I Part Number R077416
Remote Power Controller Module, Type IV Part Number R072702
Remote Power Controller Module, Type V Part Number R077419
Remote Power Controller Module, Type VI Part Number R077420

SSP 41172 REQUIREMENT:

Paragraph 4.2.3.2, Test Description: ALL REQUIREMENTS

Paragraph 4.2.3.3B, Test Levels and Duration, Temperature. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

Paragraph 4.2.3.3C, Test Levels and Duration, Duration. The dwell time at the high and low levels shall be long enough to obtain internal thermal equilibrium.

Paragraph 4.2.3.4, Supplementary Requirements. Functional tests shall be conducted, as a minimum, at the maximum predicted temperature plus 20 degrees F (11.1 degrees C) and at the minimum predicted temperature minus 20 degrees F (11.1 degrees C) during the first and last thermal cycles and after return of the component to ambient.

EXCEPTION:

A. No thermocouple was placed on the Qualification units during Qualification Thermal Cycle Testing.

B. The units were not operated at full power and did not achieve the maximum predicted temperature extremes, and therefore did not demonstrate the 20 degrees F margins.

C. Dwell times were insufficient to achieve internal thermal equilibrium.

RATIONALE:

Delta Qualification Thermal Vacuum Testing conducted on all Qualification units with thermocouples on the units did achieve the maximum predicted temperature extremes and 20 degrees F margins with dwell times sufficient to produce internal thermal equilibrium. Hot and cold restarts were also conducted at the Qualification temperature extremes. Thermal Vacuum Testing is more stringent than ambient testing because the RPCM depends on conduction to reject heat. Although fewer cycles are utilized in the Delta Thermal Vacuum Testing, the qualification of the design is adequate. Since the primary purpose of conducting a greater number of cycles during ambient testing is to stress the EEE parts via the thermal ramp rate, this was previously accomplished during the as–run Qualification Thermal Cycling Test.

PG2–58:

ITEMS:

Remote Power Controller Modules, Type I Part Number R077416 Serial Numbers: C020002, C020003, C235321, X751346, X751347, and X800558

Remote Power Controller Modules, Type II Part Number R077417 Serial Numbers: C109000, X751457, X751458, X751459, C024154, and B968160

Remote Power Controller Modules, Type III Part Number R077418 Serial Numbers: C108998, B968161, X751349, X751350, X751351, X751461, and X751462

Remote Power Controller Modules, Type IV Part Number R072702 Serial Numbers: C113568, C171123, C250617, X751463, X751464, B968202, and B968203

Remote Power Controller Modules, Type V Part Number R077419 Serial Numbers: C024243, C024244, C024245, C024246, C113615, C206340, C235298, X751390, X800390, X800391, X800392, B956621, C024088, C024090, C024091, C024092, C024094, C024095, C024096, C024097, C024098, C024100, C024239, C024240, C024241, C024242, C051591, C070987, C070988, C070989, C070990, X751149, X751150, X751151, X751153, X751154, X751155, X751156, X751157, X751158, X751159, X751160, X751389, X751391, X751392, X751393, X751394, X751395, X751422, X751423, X751424, X751427, X751428, X751432, X751433, X751434, X751435, X751436, X751437, X751438, X751439, Z075729, Z076719, Z076720, Z076721, Z076722, Z076724, Z093969, Z093970, C206338, and C206337

Remote Power Controller Modules, Type VI Part Number R077420 Serial Numbers: C024235, C171178, X800489, X800490, C024236, X751232, X751431, Z063025, Z087176, Z087177, Z087178, and C251951.

SSP 41172 REQUIREMENT:

THERMAL VACUUM TEST, COMPONENT ACCEPTANCE

Paragraph 5.1.2.2, Test Description: ALL REQUIREMENTS

Paragraph 5.1.2.3B, Test Levels and Duration, Temperature: The component shall be at the maximum predicted temperature during the hot portion of the cycle and at the minimum predicted temperature during the cold portion of the cycle.

Paragraph 5.1.2.3C, Test Levels and Duration, Duration: The component shall undergo a dwell period of at least one hour or a time sufficient for the component to reach internal thermal equilibrium as established by qualification testing, whichever is greater, at both the high and low temperature extremes with power off and then turned on.

Paragraph 5.1.2.4, Supplementary Requirements: Functional tests shall be conducted at the maximum and minimum predicted temperature levels during the first and last operating cycles after the dwell and after return of the component to ambient temperature.

THERMAL CYCLING TEST, COMPONENT ACCEPTANCE

Paragraph 5.1.3.2, Test Description: ALL REQUIREMENTS

Paragraph 5.1.3.3B, Test Levels and Duration, Temperature: The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

Paragraph 5.1.3.3C, Test Levels and Duration, Duration: The dwell time at the high and low levels shall be long enough to obtain internal thermal equilibrium.

Paragraph 5.1.3.4, Supplementary Requirements: Functional tests shall be conducted during the first and last operating thermal cycles after the dwell at the maximum and minimum predicted operating temperatures and after return of the component to ambient.

EXCEPTION:

A. No thermocouple was placed on the unit.

B. The temperature extremes achieved during Acceptance Thermal Cycle and Acceptance Thermal Vacuum Testing did not reach the minimum and maximum predicted values.

C. Full–power testing was not conducted with sufficient dwell time to achieve Internal Thermal Equilibrium.

D. Cold and Hot Re–start tests were conducted at less extreme temperatures than the minimum and maximum predicted values.

RATIONALE:

The SSP 41172 noncompliances of RPCM Acceptance Thermal Cycle/Thermal Vacuum Testing result in minimal risk of undetected workmanship defects.

During thermal screening, it is typical to find more problems at low temperatures than at high temperatures and the low temperatures achieved were very close to the 5.4 degree F test tolerance. No failure modes have been identified for which a quantifiable reduction in acceptance screening strength can be correlated to the reduced hot temperature test limits. Furthermore, it is unlikely that the absolute maximum (worst case) predicted temperature for any of the sub–components will ever occur during the operational lifetime of the units. Worst–case scenarios used for such maximum predictions generally combine several extreme conditions, each of which have a low probability of occurrence, and have a combined probability which is extremely low. Such factors include: all channels being utilized, maximum power output, sustained duration of demand, and all under maximum environmental temperature conditions. Prudent engineering practice would preclude the design of a system in which RPCMs would be used without de–rating of the component units and/or redundancy.

Thermal Analysis was conducted to evaluate whether RPCM Acceptance Thermal Testing would detect a manufacturing defect such as failure to install thermal conductive paste. The results of that investigation concluded that the design of the RPCM is sufficiently robust that such a manufacturing anomaly would result in no out–of–specification condition and no degradation in RPCM performance.

Although a reduction in screening effectiveness due to not achieving the maximum predicted temperature cannot be quantified, it is believed that it constitutes minimal risk to the Program particularly when cost and schedule impacts resulting from re–testing are considered.

PG2–59:

ITEM:

Beta Gimbal Assembly (BGA) Electronic Control Unit (ECU) Part Number R072341

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification

Paragraph 4.2.2.3, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

Paragraph 4.2.2.4, Supplementary Requirements. Functional tests shall be conducted, as a minimum, at the maximum operating temperature plus 20 degrees F (11.1 degrees C) and at the minimum operating temperature minus 20 degrees F (11.1 degrees C) during the first and last operating cycle after the dwell and after return of the component to ambient temperature.

EXCEPTION:

The BGA ECU will demonstrate a thermal margin of minimum acceptance temperature minus 14 degrees F for operation and startup during component qualification thermal vacuum tests.

RATIONALE:

The BGA ECU contains electronics comprised of EEE and S–rated parts with an operational minimum temperature of –65 degrees F for their intended life cycle with power on. Analytical assessments of changes to the on–orbit thermal sink temperatures for the ECU, including minimum effective dissipation revisions, resulted in a decrease of operational margin by –6 degrees F to –51 degrees F. To obtain full qualification margin, the ECU assembly would require exposure to -71 degrees F which could result in degraded performance and potential overstress to the internal EEE parts.

PG2–60:

ITEM:

Beta Gimbal Assembly (BGA) Electronic Control Unit (ECU) Part Number R072341

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification

Paragraph 4.2.3.3, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

Paragraph 4.2.3.4, Supplementary Requirements. Functional tests shall be conducted, as a minimum, at the maximum predicted temperature plus 20 degrees F (11.1 degrees C) and at the minimum predicted temperature minus 20 degrees F (11.1 degrees C) during the first and last thermal cycles and after return of the component to ambient.

EXCEPTION:

The BGA ECU will demonstrate a thermal margin of minimum acceptance temperature minus 14 degrees F for operation and startup during component qualification thermal cycle tests.

RATIONALE:

The BGA ECU contains electronics comprised of EEE and S–rated parts with an operational minimum temperature of –65 degrees F for their intended life cycle with power on. Analytical assessments of changes to the on–orbit thermal sink temperatures for the ECU, including minimum effective dissipation revisions, resulted in a decrease of operational margin by –6 degrees F to –51 degrees F. To obtain full qualification margin, the ECU assembly would require exposure to -71 degrees F which could result in degraded performance and potential overstress to the internal EEE parts.

PG2–61:

ITEMS:

Rocketdyne Truss Attachment System (RTAS)
Z1 Bolt Assembly Part Number R074940–1
S6/P6 Nut Assembly Part Numbers R078813–1, R078813–11, and R078813–21
S5/P5 Bolt Assembly Part Numbers R074940–1, R074940–11, and R074940–12
S4/P4 Nut Assembly Part Numbers R078860–1, R078860–11, R078860–21, and R078860–31

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Qualification Thermal Vacuum Test is replaced by a Qualification Thermal Cycle Test.

RATIONALE:

The RTAS is a moving mechanical assembly without electronic or electrical components. A Qualification Thermal Cycle Test from –100 degrees F to 135 degrees F is performed on this hardware at the component level to certify the design. One RTAS bolt and nut pair is functioned in a Human Thermal Vacuum test from –100 degrees F to 135 degrees F up to fully–loaded conditions (385 ± 39 in–lbs.) which verifies performance on–orbit. Additionally, inspection is used to verify that the assembly clearances are adequate as predicted by the component level analysis.

PG2–62:

ITEMS:

Rocketdyne Truss Attachment System (RTAS)
Z1 Bolt Assembly Part Number R074940–1
S6/P6 Nut Assembly Part Numbers R078813–1, R078813–11, and R078813–21
S5/P5 Bolt Assembly Part Numbers R074940–1, R074940–11, and R074940–12
S4/P4 Nut Assembly Part Numbers R078860–1, R078860–11, R078860–21, and R078860–31

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Acceptance Thermal Vacuum Test on the RTAS will not be performed.

RATIONALE:

The RTAS is a moving mechanical assembly without electronic or electrical components. One RTAS bolt and nut pair is functioned in a Human Thermal Vacuum test from –100 degrees F to 135 degrees F up to fully–loaded conditions (385 ± 39 in–lbs.) which verifies performance on–orbit. Additionally, inspection is used to verify that the assembly clearances are adequate as predicted by the component–level analysis.

PG2–63:

ITEMS:

Rocketdyne Truss Attachment System (RTAS)
Z1 Bolt Assembly Part Number R074940–1
S6/P6 Nut Assembly Part Numbers R078813–1, R078813–11, and R078813–21
S5/P5 Bolt Assembly Part Numbers R074940–1, R074940–11, and R074940–12
S4/P4 Nut Assembly Part Numbers R078860–1, R078860–11, R078860–21, and R078860–31

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

Paragraph 4.2.3.4, Supplementary Requirements. Functional tests shall be conducted, as a minimum, at the maximum predicted temperature plus 20 degrees F (11.1 degrees C) and at the minimum predicted temperature minus 20 degrees F (11.1 degrees C) during the first and last thermal cycles and after return of the component to ambient.

EXCEPTION:

The RTAS shall have a margin of 5 degrees F beyond the worst–case on–orbit maximum temperature predicted by ISS Assembly Sequence Revision D+ during Qualification Thermal Cycle testing.

RATIONALE:

The RTAS is a moving mechanical assembly without electronic or electrical components. A Qualification Thermal Cycle Test from –100 degrees F to 135 degrees F is performed on this hardware at the component level to certify the design. ISS Assembly Sequence Revision B analysis yields a maximum temperature of 125 degrees F at its location on P6**,** and ISS Assembly Sequence Revision D+ analysis yields a maximum temperature of 130 degrees F at its location on P6. However, additional qualification testing is not warranted as analysis indicates there is both sufficient clearance and strength margins for the fine alignment hardware elements.

At the P6 location, the critical clearance at maximum temperature is the contingency race spherical inside diameter to the contingency ball outside diameter clearance. At the predicted Revision B maximum temperature, analysis indicates the clearance will be 0.00084 inches; additionally, the analysis further derives an adequate clearance at a conservatively high temperature of 140 degrees F of 0.000792 inches. Also, the internal components with the smallest strength margins were assessed at a temperature of 140 degrees F. The component with the least margin, the Primary Ball Bearing, still would retain a margin of safety on ultimate of 0.08 and a margin of safety on yield of 0.23.

Finally, the ISS Assembly Sequence Revision D+ maximum temperature of 130 degrees F would occur at a combination of the XPOP flight–configuration, hot–biased environments, high beta angle (+/– 75 degrees), and a narrow range of attitudes. The high beta angle alone is expected to occur less than 3 percent per year (12 days per year). The likelihood that this combination would occur where these worst–case clearances and strength margins are applicable is remote. Thus, the Qualification program, when combined with the analysis indicated, is adequate to certify the design.

PG2–64:

ITEM:

Bearing Motor and Roll Ring Module (BMRRM) Part Number R074030–11

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. The test duration in each of the three orthogonal axes shall be three times the expected flight exposure time to the maximum predicted level and spectrum or three times the component random vibration acceptance test time if that is greater, but not less than three minutes per axis.

EXCEPTION:

The duration of the BMRRM Random Vibration Test shall be one minute at qualification levels.

RATIONALE:

The BMRRM used for this qualification test is a flight unit (flight set # 8). The duration per axis of the BMRRM qualification random vibration qualification was reduced to one minute to minimize life degradation of the BMRRM unit and possible damage to the flight hardware. Total cumulative time in the vibration environment was two and one–half minutes per axis after including the lower PSD levels experienced prior to the qualification level test that ensured that the test setup was properly characterized. Plans do account for an inspection and, if needed, refurbishments with a suitable reverification test for flight.

PG2–65:

ITEM:

Battery Charge/Discharge Unit Part Number RE1807

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The qualification thermal margin at the maximum and minimum temperatures shall be no less than 15 degrees F.

RATIONALE:

As permitted by SSP 41172, the qualification thermal tests (thermal vacuum and thermal cycle) were performed under vacuum conditions. The BCDU qualification thermal tests was performed to temperature limits of –31 degrees F to 109 degrees F (netting a 140 degrees F temperature swing). To comply with the SSP 41172 qualification thermal margin requirements, the acceptance test temperature limits should have been –11 degrees F to 89 degrees F (netting a 100 degrees F temperature swing). However, after the BCDU qualification tests were completed, SSP 41172 compliance discussions were held with the NASA Test and Verification Control Panel which resulted in Loral and Boeing–Canoga Park modifying the acceptance thermal test temperature limits to alleviate concerns complying with the minimum 100 degrees F temperature swing requirement. As a result, the acceptance test temperature limits were widened to –16.6 degrees F to 94.3 degrees F (netting a 110.9 degrees F temperature swing). The modified temperature limits ensure compliance with the 100 degrees F temperature swing requirement when worst case (+/– 5.4 degrees F) test temperature tolerances are applied (i.e. With worst case test tolerances applied, the test could be performed from –11.2 degrees F to 88.9 degrees F, yielding a 100.1 degree F temperature swing). Unfortunately, the modified temperature limits compromise the 20 degrees F qualification margin by 5 degrees F at both the minimum and maximum temperatures.

However, the temperatures that are utilized do lead to a more conservative acceptance test screening of the flight BCDUs and also ensures that the 100 degree F temperature swing is met under worst case test temperature conditions. The test causes no risk of damage or overstress applied to any of the internal components. This is supported by the fact that during acceptance testing, a 15–degree F margin is maintained between the BCDU qualification and acceptance test temperatures.

PG2–66:

ITEM:

Battery Subassembly ORU Part Number RE1804

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:
A. The maximum predicted flight level and spectrum minus 6 dB, but not less than a level derived from a 135 dB overall acoustic environment (whose spectrum is defined by NSTS 21000–IDD–ISS, paragraph 4.1.1.5).

EXCEPTION:

The Acceptance Random Vibration Test of the Battery Subassembly ORU in the Z–axis shall be performed to the following levels:

Frequency Range (Hz)	Power Spectral Density	Tolerance (+/– dB)	Comment
20	0.001250	3.0	
64	0.013256	3.0	
68	0.010858	3.0	Notched
84 to 112	0.001700	3.0	Notched
152	0.025000	3.0	Notched
450	0.025000	3.0	
500	0.019200	3.0	
2000	0.000600	3.0	

Composite = 3.87 Grms. 94.4 percent of AT level

RATIONALE:

Battery Subassembly ORU hardware, QM–00 and QM–01, were successfully tested at Qualification Random Vibration test levels in the flight configuration without notching. This initial qualification random vibration testing of these Battery Subassembly ORUs displayed an equipment control problem in the high frequencies that was caused by motion of the rear Acme post. To allow the vibration test equipment to control within tolerances, to reduce test time, and the associated hardware fatigue life consumption (by 40 to 90 percent), the rear acme post was bolted directly to the vibration fixture for acceptance testing performed on QM–02 and subsequent flight Battery Subassembly ORUs hardware. This configuration change (reference Test Configuration Variation PG–2–160) prevented the Acme post from moving and interfering with the equipment control. As a result, more energy was transmitted into the hardware under test and response limiting occurred.

As flight hardware would not contain response limiting accelerometers and to prevent possible damage to the flight hardware, a notched random vibration Power Spectral Density was developed from response limiting accelerometers located on the ORU base plates of QM–02 through QM–04 Battery Subassembly ORUs during their AT screening. With the rear post locked and the test input notched below maximum predicted flight levels less 6 dB, the internal response is approximately equal to the internal response with the rear post unlocked as in the flight configuration.

An exception to SSP 41172 for the Battery Signal Conditioning and Control Module (BSCCM) Acceptance Random Vibration test environment has previously been approved (reference SSCN 1933).

PG2–67:

ITEM:

Battery Signal Conditioning and Control Module (BSCCM) Part Number E006400

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The BSCCM will not undergo a Qualification Thermal Vacuum test.

RATIONALE:

The BSCCM underwent an Acceptance Thermal Vacuum Test from –31 degrees F to 122 degrees F. However, the BSCCM did not undergo an independent Qualification Thermal Vacuum Test though, as part of a higher–level Battery Subassembly ORU Qualification Thermal Vacuum Test, the BSCCM did experience temperatures from –13 degrees F to 86 degrees F. These temperatures do not envelop the independent Acceptance Thermal Vacuum Test performed.

The BSCCM did undergo an Acceptance Thermal Cycle Test from –31 degrees F to 122 degrees F. Also, the BSCCM underwent a Qualification Thermal Cycle Test from –51 degrees F to 142 degrees F. As indicated, the electronics have been stressed with 20 degrees F temperature margin during the Qualification program under ambient pressure conditions.

The BSCCM is a low–power electronics box with maximum power dissipation of 8.4 Watts and nominal power dissipation of 3.9 Watts. At these levels, any temperature increase of the internal BSCCM electronics during operation under vacuum conditions would be small compared to temperatures reached at atmospheric pressure. Thus, the Qualification Thermal Cycle Test of the BSCCM provides sufficient evidence that the BSCCM is designed with sufficient thermal margins.

Additionally, the failure of a BSCCM does not result in failure of its Battery ORU functionality.

PG2–68:

ITEM:

Plasma Contactor Unit (PCU) Part Number R078480–11

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Description and Alternatives. Component leak checks shall be made prior to initiation of, and following the completion of, component qualification thermal and vibration tests.

EXCEPTION:

No Leakage Test will be performed between the Qualification Vibration Test and the Qualification Thermal Vacuum Test.

RATIONALE:

A final post–environmental Leakage Test will be conducted which assures that the component successfully passed all environmental testing. When the chamber is drawn down to vacuum prior to Qualification Thermal Vacuum Testing, any gross leakage would be detected as a difficulty of achieving and/or maintaining vacuum. Performance of the Leakage Test in accordance with 4.2.11.2 would require that Xenon be purged from the component, which is both expensive and time–consuming. Although conducting the Leakage Test between the Qualification Vibration and Qualification Thermal Vacuum Test would simplify the root cause investigation should a failure occur, the removal of this instance of a leak test contributes little risk to the International Space Station Program.

PG2–69:

ITEM:

Plasma Contactor Unit (PCU) Part Number R078480–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Description and Alternatives. The component leak checks shall be made before and after exposure to each environmental acceptance test.

EXCEPTION:

No Leakage Test will be performed between the Acceptance Vibration Test and the Acceptance Thermal Vacuum Test.

RATIONALE:

A final post–environmental Leakage Test will be conducted which assures that the component successfully passed all environmental testing. When the chamber is drawn down to vacuum prior to Acceptance Thermal Vacuum Testing, any gross leakage would be detected as a difficulty of achieving and/or maintaining vacuum. Performance of the Leakage Test in accordance with 5.1.7.2 would require that Xenon be purged from the component, which is both expensive and time–consuming. Although conducting the Leakage Test between the Acceptance Vibration and Acceptance Thermal Vacuum Test would simplify the root cause investigation should a failure occur, the removal of this instance of a leak test contributes little risk to the International Space Station Program.

PG2–70:

ITEM:

NH3 Accumulator Part Number SV809903–5

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The NH3 Accumulator will not undergo a Qualification Thermal Vacuum test.

RATIONALE:

The Z1 NH3 Accumulator electronics and moving components are internal to the accumulator. The operating environment of these components would be unaffected by external pressure during a vacuum test. Thus, a Qualification Thermal Vacuum test of the Accumulator is not required as the worst–case potential stress would be encompassed during the Qualification Thermal Cycle test.

PG2–71:

ITEM:

NH3 Accumulator Part Number SV809903–5

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Qualification.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The NH3 Accumulator will not undergo an Acceptance Thermal Vacuum test.

RATIONALE:

The Z1 NH3 Accumulator electronics and moving components are internal to the accumulator. The operating environment of these components would be unaffected by external pressure during a vacuum test. Thus, an Acceptance Thermal Vacuum test of the Accumulator is not required as the worst–case potential stress would be encompassed during the Acceptance Thermal Cycle test.

PG2–72:

ITEM:

NH3 Accumulator Part Number SV809903–5

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The Qualification Thermal Cycle test for the Z1 NH3 accumulator was performed to the same temperature spectrum as the Acceptance Thermal Cycle test.

RATIONALE:

During Acceptance Thermal Cycle testing, the Z1 NH3 accumulators completed 8 Thermal Cycles over the temperature range of –85 degrees F to 120 degrees F. During Qualification Thermal Cycle testing, the accumulator completed 32 thermal cycles over the identical temperature range. Thus, during the Qualification program, 20 degrees F thermal margin was not exhibited in accordance with 4.2.3.3B.

The minimum nonoperational temperature the Z1 NH3 accumulators will be exposed is – 60 degrees F. The operational temperature extremes the accumulators will experience are – 2 degrees F to 105 degrees F. Under both qualification and acceptance, all accumulators were functionally tested at – 67 degrees F and 120 degrees F. The only electrical devices within the accumulator are potentiometers. Four flight potentiometers were successfully actuated (cable pull test) at – 85 degrees F to verify acceptable cable spring and potentiometer greases at cold temperatures. Additionally, all potentiometers underwent five nonoperational thermal cycles from – 67 degrees F to 257 degrees F (–65 degrees C to 125 degrees C) in accordance with MIL–STD–202 Condition 107, Method B. Thus, through all testing performed, the NH3 Accumulator and its internal potentiometer design can be deemed adequately qualified for performed acceptance tests.

PG2–73:

ITEM:

Beta Gimbal Assembly (BGA) Part Number R075800

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Beta Gimbal Deployment Transition Structure (4–Bar Deployment Structure) internal to the Beta Gimbal Assembly was not included in the Qualification Thermal Vacuum Test.

RATIONALE:

The Beta Gimbal Deployment Transition Structure (4–Bar Deployment Structure) consists of four aluminum bars and a cross bar. The four aluminum bars are attached to the Integrated Equipment Assembly via the Cam Clevis. The 4–Bar Deployment Structure is the only moving mechanism. The team conducted a separate thermal cycle test on the Cam Clevis prior to the Qualification Thermal Vacuum test of the BGA. As a result, the Beta Gimbal Deployment Transition Structure hardware was not included in the BGA Qualification Thermal Vacuum Test.

However, the Thermal Cycle Test of the Cam Clevis is a more stringent test as the unit under test was actuated to lock and unlock position at each temperature limit to demonstrate performance under extreme conditions. This operational capability could not be demonstrated in the Qualification Thermal Vacuum test without significant complexity to operate the mechanism. Additionally, the 4–Bar Deployment Mechanism have undergone testing under vacuum conditions as part of Test Case 5 of Human Thermal Vacuum testing at NASA JSC.

PG2–74:

ITEM:

Sequential Shunt Unit Part Number RE1806

SSP 41172 REQUIREMENT:

Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Response Limiting of the Vibration Profile shall be allowed on the SSU Qualification Unit QM01. Procedures shall be modified to ensure post–test data evaluation of control accelerometer to ensure the response–limited input to the SSU shall not be less than the maximum predicted flight level.

RATIONALE:

Response limiting is required to limit the internal vibration responses to preclude exceeding the design margins and potentially damaging the SSU. SSU Vibration test data exhibit a unit–to–unit variation such that uniformity with a tailored notched profile for testing cannot been established that would provide an adequate test. The stated restrictions on control input limitations ensure that flight levels are not violated to provide an adequate workmanship screen.

PG2–75:

ITEM:

Sequential Shunt Unit Part Number RE1806

SSP 41172 REQUIREMENT:

Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Response limiting of the Vibration Profile shall be allowed on all SSU Flight Units. Procedures shall be modified to ensure post–test data evaluation of control accelerometers to ensure the response–limited input to the SSUs shall not be less than six dB below the maximum predicted flight level.

With response limiting, the SSU will be required to meet an overall energy level of 6.1 grms +/– 3.0 dB.

RATIONALE:

Response limiting is required to limit the internal SSU vibration responses to preclude exceeding the design margins and potentially damaging the units. SSU Vibration test data exhibit a unit–to–unit variation such that uniformity with a tailored notched profile for testing cannot been established that would provide an adequate test. The stated restrictions on control input limitations ensure that flight levels are not violated to provide an adequate workmanship screen.

PG2–76:

ITEMS:

Power Supply A/E Part Number R075730

Switchgear Controller Assembly Part Numbers R072526, R078224, and R078226

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.

Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Switchgear Controller Assembly and the Power Supply A/E shall not undergo a qualification random vibration test at the component level.

RATIONALE:

The design certification approach for both the Switchgear Controller Assembly and the Power Supply A/E is documented in the Boeing – Canoga Park Master Verification Plan (RI/RD94–637). Therein is stated that though electrical functional, random vibration, thermal cycle, and burn–in testing is performed as a confidence test at the Switchgear Controller Assembly and the Power Supply A/E level, these components are qualified and acceptance tested as part of the Main Bus Switching Unit (Part Number R072591) or the DC Switching Unit (Part Number R072610) ORU. Concerns remain that the qualification random vibration test at the ORU level does not adequately encompass random vibration testing performed on the Flight Switchgear Controller Assemblies and Power Supplies A/E during the confidence testing and acceptance testing phases.

During development testing, an Engineering Model Switchgear Controller Assembly did undergo random vibration testing at this component level (reference EID–01479). During testing, the Engineering Model Switchgear Controller Assembly experienced overall random vibration levels of 16.5 grms in all three axes for three minutes per axis as follows:

Flight SCA Random Vibration Confidence Test Levels		Engineering Model SCA Random Vibration Development Test Levels	
Frequency	Level	Frequency	Level
20 Hz	$0.01G^2/Hz$	20 Hz	$0.05G^2/Hz$
20 – 80 Hz	+3 dB/octave	20 – 64 Hz	+6 dB/octave
80 – 350 Hz	$0.04G^2/Hz$	64 – 366 Hz	$0.5\ G^2/Hz$
350 – 2000 Hz	–3 dB/octave	366 – 2000 Hz	–7.5 dB/octave
2000 Hz	$0.007G^2/Hz$	2000 Hz	$0.0072G^2/Hz$
Overall	6.1 Grms	Overall	16.5 Grms
Duration	1 minute/axis	Duration	3 minutes/axis

The Engineering Model SCA Random Vibration Levels effectively qualifies for the Flight SCA Random Vibration Confidence Test Levels.

During development testing (noted in EID–05181) of an Engineering Model Main Bus Switching Unit, a triaxial accelerometer and a Y–axis accelerometer was placed on the ORU baseplate at the Power Supply A/E location. The triaxial accelerometer recorded random vibration levels of 12.1 grms for the X–axis, 16.3 grms for the Y–axis, and 32.2 grms for the Z–axis. The Y–axis accelerometer recorded 12.3 grms for the Y–axis. These are equal to or greater than the MBSU ORU level of 12.1 grms recorded by the control accelerometer at the Acme interface. Therefore, the Power Supply A/E is effectively qualified at the ORU level as the stimulus recorded during this testing is as great or greater than recorded on the ORU during Qualification Vibration tests.

PG2–77:

ITEM:

Plasma Contactor Unit (PCU) Part Number R078480–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Qualification.

Paragraph 5.1.7.2, Test Description and Alternatives. The component leak checks shall be made before and after exposure to each environmental acceptance test.

EXCEPTION:

No Leakage Test on the High–Pressure Side of the Plasma Contactor Unit will be performed after completion of Environmental Testing.

RATIONALE:

Leakage testing on the high–pressure side of the Plasma Contactor unit will not propagate workmanship failures that would not have been screened in prior proof and leak testing. As a result, the performance of this test would contribute little additional benefit. Leakage testing on the low–pressure side of the Plasma Contactor Unit will be conducted after all environmental testing via SSP 41172 Method VI Leakage methodology. This will include evacuating the Plasma Contactor Unit internal components downstream of the PCU regulator with the Heater Controller Assembly Swagelok fitting capped.

PG2–78:

ITEM:

Pump Flow Control Subassembly (PFCS) Part Number RE2814

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3C, Test Level and Duration. A minimum of three temperature cycles shall be used.

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3C, Test Levels and Duration. The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total.

EXCEPTION:

A minimum of one hot thermal vacuum cycle shall be used.

RATIONALE:

Review of original qualification thermal vacuum and thermal cycle testing of the PFCS deemed both tests to be inadequate for the hot temperature condition. PFCS Qualification Test Report SVSHER 1818370 (dated 18 September 1997) documents results of this testing. After evaluation of the impacts to fully repeat thermal vacuum and thermal cycle qualification testing, it was deemed that a one–cycle hot thermal vacuum test would be performed to demonstrate the PFCS ability to perform under worst–case predicted hot on–orbit conditions. This test was performed successfully. PFCS Delta Qualification Test Report SVHSER19573 (dated 30 March 99) documents results of this testing.

PG2–79:

ITEM:

Pump Flow Control Subassembly Part Number RE2814, Serial Numbers 00003 through 00008

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Thermal Vacuum Testing shall not be required during the first acceptance test of these Serial Numbers. Any subsequent Acceptance Thermal Testing of Serial Numbers 00003 – 00008 shall be in accordance with SSP 41172, Acceptance Thermal Vacuum Test requirements.

RATIONALE:

PFCS Serial Numbers 0003 through 0008 shall have Electronic component and PFCS ORU–level Acceptance Thermal Cycle testing to 100 degree F minimum sweep requirements used for workmanship screening substituted in lieu of Acceptance Thermal Vacuum testing. A PFCS Conductive Path Analysis (E.M. CSS–P–EM–414) and a design build record review were performed. The results establish that the PFCS operation in a worst–case ISS thermal environment is well within the design margins for all electronic components of this ORU. This approach of test backed by analysis for PFCS Serial Numbers 00003–00008 is sufficient for acceptance of these ORUs.

PFCS Serial Numbers 0009–0012 will be Thermal Vacuum tested to ISS equivalent environments in accordance with SSP 41172 requirements.

PG2–80:

ITEM:

Pump Flow Control Subassembly Part Number RE2814

SSP 41172 REQUIREMENT:

Paragraph 4.2.10, Pressure Test, Component Qualification.
Paragraph 4.2.10.2C, Test Description. Ultimate Pressure: For such items as pressure vessels, pressure lines, and fittings, the temperature of the component shall be consistent with the critical–use temperature, and the component shall be pressurized to design ultimate pressure or greater. The internal pressure shall be applied at a uniform rate such that stresses resulting from shock loading are not imposed. Ultimate testing shall not be performed on actual flight articles.

EXCEPTION:

An ultimate pressure test in accordance with SSP 41172 on the PFCS ORU shall not be performed. The Accumulator Pressure Vessel and Fluid Quick Disconnects Couplings components of PFCS shall be subjected to ultimate pressure testing to qualification level requirements of SSP 41172.

RATIONALE:

Typically, testing of fluid lines is not performed due to sufficient margin for such items. A structural PFCS Burst Analysis and Test Results (E.M. CSS–P–EM–392, dated 8/31/99) shows sufficient margin for the worst–case PFCS component (Brazed Manifold, Part Number SV809925) with a minimum margin of safety of 0.02 based on a factor of safety of 1.1 at a burst pressure of 1465 psi.

PG2–81:

ITEM:

Solar Array Wing Blanket Box Motor Drive Assembly Manual Backup Assembly, Part Number 5836788

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Qualification Thermal Vacuum Testing of the Solar Array Wing Blanket Box Motor Drive Assembly Manual Backup Assembly is replaced by thermal testing at ambient pressure with dwell periods of at least one hour at each temperature extreme.

RATIONALE:

There are no vacuum sensitive components in the mechanism. Since the validity of the test is not reduced, the increased cost of performing the test at vacuum is not necessary.

The manual backup assembly is a mechanical device composed primarily of aluminum and steel with a mass less than 10 lbs. Since the dwell period is intended to insure that the unit has reached a state of internal thermal equilibrium at the required temperature condition, a small mass made of thermally conductive materials will reach temperature quickly. A one–hour dwell period during thermal testing is sufficient to reach equilibrium and, therefore, a minimal risk.

PG2–82:

ITEM:

Beta Gimbal Assembly Roll Ring Assembly Part Number RE1822–02

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (Minimum design temperature) during the cold portion of the cycle.

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The Beta Gimbal Assembly Roll Ring Assembly minimum qualification test temperature is –65 degrees F.

RATIONALE:

The minimum predicted Beta Gimbal Assembly Roll Ring Assembly on–orbit operating temperature is –48 degrees F.

The BGA qualification unit has experienced thermal testing at a minimum operating temperature of –67 degrees F. During this test, the minimum measured Roll Ring Assembly temperature was also –67 degrees F; therefore there is only 19 degrees F qualification margin on the Roll Ring Assembly instead of the required 20 degrees F. SSP 41172 section 3 allows a maximum test temperature tolerance of +/– 5.4 degrees F. The 1 degree F non–compliance is well within this allowed tolerance and is therefore acceptable.

PG2–83:

ITEM:

Beta Gimbal Assembly Roll Ring Assembly Part Number RE1822–02

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3B, Test Levels and Duration. The component shall be at the maximum predicted temperature during the hot portion of the cycle and at the minimum predicted temperature during the cold portion of the cycle.

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

The Beta Gimbal Assembly Roll Ring Assembly minimum acceptance temperature is –45 degrees F.

RATIONALE:

The minimum predicted Beta Gimbal Assembly Roll Ring Assembly on–orbit operating temperature is –48 degrees F; however, all flight Roll Ring Assemblies have been acceptance thermal tested at –45 degrees F.

The first two BGA flight units experienced minimum acceptance thermal test temperatures of –45 degrees F (flight BGAs 3 through 8 do not receive any thermal acceptance testing in accordance with SSP 41172, PG2–11). During this testing, the measured temperature of the Roll Ring Assemblies was –45 degrees F. Therefore, there is a 3 degree F non–compliance between the acceptance test temperature and the worst–case on–orbit predicted environment.

The minimum predicted BGA operating temperature on the P6 element of –48 degrees F is at extreme high beta angles for XVV flight attitudes between Stages 4A through 13A when P6 is located on the Z1 and do not occur for Mean Propulsive Attitudes. High beta angles (60 to 75 degrees) occur for less than 20 days per year.

SSP 41172 section 3 allows a maximum test temperature tolerance of +/– 5.4 degrees F. The 3–degree non–compliance is within this allowed tolerance level and is therefore acceptable.

PG2–84:

ITEM:

SAW Mast Canister Manual Override Mechanism AEC–Able Part Number 541K600

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

No acceptance thermal vacuum or acceptance thermal cycle testing will be performed on SAW Mast Canister Manual Override Mechanism via the EVA interface.

RATIONALE:

The Flight Wing 1 through Flight Wing 8 mast canisters did undergo thermal acceptance tests, during which the mechanism was backdriven by the Motor Drive Assembly. All rotating surfaces were exercised except a bushing (No. 11 on AEC–Able Drawing 541K600). When the drive is shifted from automatic to manual, a bushing on the pinion gear rotates inside a counterbore of the input shaft. A tolerance stackup analysis was performed for this interface with the effects of maximum predicted delta T (260+ degrees F). The results indicate sufficient margin at temperature to prevent binding.

Additionally, the start–up torque qualification test (AEC–Able 1059D1520) performed at the component level provides a high confidence in the strength of the mechanism driving through the EVA interface. The test consisted of applying 160 in–lbs. of torque to the manual override idler gear in a stalled condition. This was followed by a functional test which simulated deployment and retraction for 50 cycles each direction at ambient, and three cycles each at hot, cold, and ambient temperatures. Performance of the unit was unaffected by the start–up torque loads as evidenced by the nearly identical performance measured during the initial and final ambient functional tests. The high gear efficiencies at temperature indicate sufficient torque margin for the EVA drive.

PG2–85:

ITEM:

SAW Blanket Box Manual Backup Assembly Lockheed Missile and Space Systems Part Number 5836788

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

No acceptance thermal vacuum or acceptance thermal cycle testing will be performed on the SAW Blanket Box Manual Backup Assembly via the EVA interface.

RATIONALE:

A tolerance stackup analysis was performed for every interface of the SAW latch mechanism that includes the manual backup assembly. The effects of maximum predicted delta T (170 degrees F from ambient to cold) were calculated for every interface.

Additionally, existing test data, manufacturing processes, and assembly processes were reviewed to augment the tolerance analyses. A review of this information by NASA, Boeing, and Lockhead Missile and Space Systems Structures and Mechanisms engineers concluded there was sufficient design margin in the manual backup assembly to allow for potential workmanship defects in the flight units.

PG2–86:

ITEM:

Direct–Current to Direct–Current Converter Unit – Heat Pipe Part Number R079903–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.2. Thermal Vacuum Test, Component Qualification.

Paragraph 4.2.2.3B. Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperature of an electrical functional test during qualification thermal vacuum testing shall be – 31 degrees F.

RATIONALE:

The minimum on–orbit thermal specification requirement for DDCU instrumentation accuracy is –11 degrees F. The qualification thermal vacuum test included successful completion of an electrical functional test at –31 degrees F. However, during acceptance thermal vacuum testing, an electrical functional test on Serial Numbers X8821267 and X8821268 was successfully performed at –45 degrees F although it was not required below –11 degrees F. In accordance with SSP 41172, since an electrical functional test was conducted at –45 degrees F on the flight units identified, an electrical functional test is required at –65 degrees F during qualification testing to establish a margin of 20 degrees F for this particular test.

During qualification testing, a cold restart test was performed at –65 degrees F. This test includes a start–up in Monitor–Hi mode from a –65 degrees F thermal equilibrium state. The Monitor–Hi start–up test includes an abrupt start–up and sustained operation at full–power; therefore, it is considered to be thermally more severe than the electrical functional test during which all circuits are cycled briefly through various operational modes.

As the design was qualified for the DDCU instrumentation accuracy specification requirement and the cold restart test was performed at – 65 degrees F, the risk associated with this exception is minimal and no additional qualification testing is necessary to qualify the design for the performed electrical functional test.

PG2–87:

ITEM:

DC–to–DC Converter Unit (DDCU–E/HP) Part Number R076522

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.

Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The maximum temperature during the qualification thermal vacuum and thermal cycle testing of the DC–to–DC Converter Unit–E/HP shall be 201 degrees F.

RATIONALE:

The DDCU–E qualification unit has been successfully tested under its required 3 kW continuous load in thermal cycle and thermal vacuum environments at 201 degrees F average baseplate temperature. Subtracting the required 20 degrees F qualification margin this qualifies the DDCU–HP for a maximum operating temperature of 181 degrees F. The current worst case predicted DDCU–HP operating temperature for Assembly Sequence Rev–D+ is 193 degrees F with the spare SASA. This occurs under a continuous 3 kW loading condition, during a 5–6 day period each year when the beta angle exceeds –68 degrees, and only if the station is in certain orientations on the outer envelope of the TEA domain. The associated qualification temperature for this condition would be 213 degrees F.

Both DDCU–HP units delivered to KSC were originally fully acceptance tested and delta acceptance tested (one Thermal Vacuum cycle) at an average baseplate temperature of 188 degrees F, which was based on the Rev–B worst case conditions and updated thermal model. The associate qualification temperature for this condition would have been 208 degrees F.

The DDCU–E/HP has demonstrated full functional operation at 201 degrees F after accumulating 18.8 hours at the required 3 kW operating at 208 degrees F. SSP 41172 Qualification and Acceptance Environmental Test Requirements requires that the component test temperature be at the maximum acceptance limits plus a margin of 20 degrees F. This has been accomplished except that the normal Electrical Functional Test fails when a power–cycled event (off/on) is conducted when the average baseplate temperature was increased from 201 degrees F to 208 degrees F. The DDCU–HP will operate above 201 degrees F during qualification testing but the Electrical Functional Test will fail when the power cycled event (off/on) is conducted; therefore, the DDCU–HP can only be qualification tested to 201 degrees F which results in only a 13 degrees F margin.

	Maximum Operating Temperature	Qualification Temperature	Qualification Margin
Current Qualification	181 degrees F	201 degrees F	20 degrees F
Required Revision B (Approve with Operational Constraint to 188 degrees F)	188 degrees F	201 degrees F	13 degrees F
Required Rev D+	193 degrees F	201 degrees F	8 degrees F

DDCU–E/HP Thermal Testing (Approximate Average Baseplate Temperatures)

	Thermal Cycling		Thermal Vacuum	
Original Acceptance	8 Cycles	138 degrees F	1 Cycle	138 degrees F
First Delta Acceptance	8 Cycles	181 degrees F	1 Cycle	181 degrees F
Delta Acceptance			1 Cycle	188 degrees F
Original Qualification	24 Cycles	203 degrees F	3 Cycles	203 degrees F
Delta Qualification	3 Cycles	201 degrees F	3 Cycles	195 degrees F
Delta Qual. Retest			3 Cycles	201 degrees F

Operation of the DDCU at baseplate temperatures greater than 181 degrees F is not detrimental to the hardware itself. The DDCU–E/HP has demonstrated full functional operation at 201 degrees F. No adverse impacts on life or performance are expected at a baseplate temperature of 188 degrees F.

PG2–88:

ITEM:

Photovoltaic Radiator Flex Hoses Part Number 83–36860

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Thermal Vacuum Test shall not be performed on the Photovoltaic Radiator Flex Hoses.

RATIONALE:

The Photovoltaic Radiator flex hoses are not equipped with any electrical or electronic components or any materials that are sensitive to Thermal Vacuum environments. The geometric changes due to exposing the flex hoses to the Thermal Vacuum temperature range with the simultaneous application of an external vacuum are insignificant and will not result in any physical interference. Flex Hose Drawing 83–36860 shows minimum clearances around the flex hoses of approximately 1/4–inch. Since the flex hoses are Inconel 718 material and their maximum OD is approximately 1 inch, then their dimensional change for a 300 degree temperature change is 0.002 inch (1 inch x 300 degree F x 6.6 x 10^{-6} in/in/degree F), which is insignificant relative to any clearances.

Thermal Vacuum environment–induced changes in the torque required to bend and straighten PVR flex hoses will not result in a PVR deploy/retract torque above what a PVR motor can supply. All PVR motors will provide sufficient torque on–orbit to overcome:

(a) Differences between the PVR Qualification Thermal Vacuum test and the on–orbit applications; and
(b) Unit to unit variations.

LMMFC Report 3–47300/1999DIR–004 addresses these two concerns showing the motor has ample design margin. Key analysis from the report is summarized as follows:

The worst case condition for motor torque is when the PVR is in a cold environment. During the PVR Qualification Thermal Vacuum test at Plum Brook, the maximum measured torque required to deploy the PVR was 1531 in–lbf at –110 degrees F and 250 psi. The warmest temperature when the PVR motor torque was measured was during Pre–test checkout when the PVR was at ambient temperature. During this pre–test checkout, the measured torque to deploy the PVR was 1302 in–lbf. Assuming an increase in torque of 10 percent for unit to unit variations, the maximum torque required would be 1684 in–lbf. For conservative purposes, the roller friction on–orbit was assumed to be 25 percent of the ground roller friction instead of zero, which results in a torque reduction of 545 in–lbf. Also, hinge joint friction on–orbit would be reduced by 50 percent since the ground roller friction induces hinge loads which would not be experienced on–orbit. The lower hinge joint friction results in a reduction in torque of 86 in–lbf. The PVR Qualification Thermal Vacuum test at Plum Brook was conducted with uncovered flexhoses. Analysis based on test data shows the torque required to deploy a PVR with uncovered flexhoses is 78.2 percent of the torque required to deploy it with covered flexhoses. Adjusting the Plum Brook maximum measured torque for these considerations results in a maximum on–orbit torque requirement for PVR deployment of 823 in–lbf.

Procurement specification 304–PVR–006F–1 requires a minimum torque of 2290 in–lbf. Conservatively reducing this torque to 90 percent of the minimum value required results in the minimum torque available being 2061 in–lbf.

Comparison of the maximum torque required to the minimum torque available shows the motors have a minimum design margin of 1238 in–lbf or 150 percent as shown below. This large design margin based on conservative assumptions shows no Acceptance Thermal Vacuum testing of the flex hoses is necessary to show a PVR can be successfully deployed or retracted on–orbit.

PVR Motor Torque Design Margin	
Item Analyzed/Tested	Motor Torque (in–lbf)
Ground:	
Plum Brook Qualification Thermal Vacuum Test	1531 (maximum measured)
Plum Brook + 10 percent for unit to unit variation	1684
On–orbit Adjustments for Differences from Ground:	
Roller Friction	–545
Hinge Friction	–86
Covers	–230 = (78.2–100)/100*(1684–545–86)
Maximum On–orbit required	823 = (1684–545–86–230)
Minimum On–orbit torque available	2061* (90 percent of the minimum value required by the procurement specification 304–PVR–006F–1)
Design Margin	1238 = (2061–823) (150 percent of the Maximum on–orbit required torque)
* Torque measured during component Qualification Thermal Vacuum Testing was approximately 2400 in–lbf at the cold temperature and approximately 2800 in–lbf at the hot temperature.	

Workmanship Screening Performed

The flex hoses are screened for workmanship with a proof pressure test to 880 psi (4.4 times launch pressure) and passed dye penetrant and x–ray inspection of the welds.

The following exception supercedes PG2–09 and PG2–21. PG2–89:

ITEM:

Photovoltaic Radiator Part Number RE1894 and its mechanical components as follows:
The Radiator Deploy/Retract Mechanism Part Number 83–36884–101
The EVA Drive Part Number 2941062–1/–501
The Winch Mechanism Part Number 83–42110–1
The Cinches Part Number 83–42090–107
General Category Part Number 83–42–012

SSP 41172 REQUIREMENT:

Paragraph 6.1 Use of Qualification Assemblies for Flight (Protoflight). Subsequent assemblies/components shall be subjected to identical protoflight tests.

EXCEPTION:

Only one Photovoltaic Radiator (and all its mechanical components) shall be subjected to protoflight thermal vacuum testing. Subsequent units shall not require protoflight thermal vacuum testing.

RATIONALE:

Protoflight Photovoltaic Radiator, Serial Number 0001, successfully underwent thermal vacuum testing at NASA–Plumbrook. This test successfully verified the Photovoltaic Radiator design to function under simulated on–orbit thermal vacuum conditions.

A Coefficient of Thermal Expansion analysis with thermal effects in LMMFC Document No. 3–47300/2000R–006 shows the Photovoltaic Radiator will function as designed under “worst combination” of tolerances, assembly tolerances, adjustment tolerances, and thermal extremes.

The report documents both an integrated analysis of the Photovoltaic Radiator mechanical systems and individual analysis of mechanical interfaces that were used to determine the sensitivity of the Photovoltaic Radiator to the combination of dimensional variability and thermal deformations. This mechanical assessment of the Photovoltaic Radiator was divided into five major sections:

The Radiator Deploy/Retract Mechanism,
The EVA Drive,
The Winch Mechanism,
The Cinches, and a
General Category.

Under each category, specific mechanical components are identified. These subsections are then further broken down into individual components and interfaces. Where appropriate, the actual mathematical analysis was included.

The analysis of the Photovoltaic Radiator mechanical systems and individual analyses of mechanical interfaces was performed to determine sensitivity of the Photovoltaic Radiator to the combination of dimensional variability and thermal deformation. The analysis process involved determining the clearance margins for all mechanical interfaces with nominal part dimensions, alignments, and adjustments. Then the worst–case clearance margins were determined for ambient conditions under the extremes of part tolerance variation, alignment, and adjustment. Finally, worst–case clearance margins under the extremes of part tolerance variation, alignment, and adjustment, including the effects of the thermal extremes of worst–case hot, cold, and gradient temperatures on mechanical interface alignment, were also calculated.

This assessment addressed every line item identified in the mechanical assessment matrix. Overall findings show that all identified issues (that had identified detrimental effects on deploy and retract operations) were resolved using detail analysis and/or flight hardware as–built measurements.

Even though analysis alone cannot totally replace the manufacturing screening assurance provided by Thermal Vacuum Testing, the analysis shows that the mechanism is robust and insensitive to thermal and manufacturing tolerance because it has relatively large clearance margins. This analysis, combined with the fact that ambient functional deployment tests were conducted, indicates that the risk of functional failure due to manufacturing/assembly out–of–tolerance conditions is very low.

PG2–90:

ITEM:

Beta Gimbal Assembly (BGA) Part Number R075800

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The fully–assembled Beta Gimbal Assembly will not undergo a qualification thermal vacuum test.

RATIONALE:

The Beta Gimbal Assembly and its internal Electronic Control Unit experiences Qualification Thermal Vacuum test environments independently, with a functional test of the fully–assembled unit performed at ambient conditions.

The root concern is whether drift of the Beta Gimbal Assembly or the Electronic Control Unit electrical parameters under thermal extremes would result in anomalous performance. The following shows that drift is either non–existent under thermal conditions or that acceptable drift (i.e. off nominal, but still meeting specification) will not effect the Beta Gimbal Assembly operation.

All individual hardware components meet specified requirements under thermal extremes as indicated in the following documents:

(1) Beta Gimbal Assembly (excluding Electronic Control Unit) EID–03897

(2) Roll Ring subassembly, Honeywell Report #DR55–T–404–23–004.

(3) Electronic Control Unit, EID–03846 for qualification unit and EID–03835 for the Engineering Model ECU used in the Beta Gimbal Assembly Qualification Test (this later report also describes the differences between the Engineering Model and Flight Electronic Control Units).

There are two primary functions where the Electronic Control Unit and Beta Gimbal Assembly interface:

(1) Anti–rotation latch function involving the anti–rotation latch (RE2757) which includes the following hardware interfaces:

(a) Limit switches

(b) 15 V power supply

(2) Positioning function involving the motor (RE2739) and resolver (part of RE1822) which includes the following hardware interfaces:

(a) Motor current lines (3, 1 for each phase)

(b) Resolver excitation and cosine/sin feedback.

The following shows that drift under thermal is acceptable when applied to the specific hardware interfaces:

(1) Anti–rotation latch limit switch interface: Limit switch resistances (<10 ohms closed, >10 megaohms open) are tested under thermal conditions to meet requirements (as indicated in the latch acceptance data packages). The Electronic Control Unit is tested under thermal conditions with the maximum specification resistances (i.e. 10 ohm and 10 megaohms) to ensure that the limit switch function works even if both pieces of hardware are at thermal extremes.

(2) Anti–rotation latch power (voltage and current): The anti–rotation latch will work under voltage variances due to the heating strip design (lower voltages result in longer time frames, but will still operate). During Beta Gimbal Assembly thermal vacuum testing, the latch was powered under a minimum specification voltage condition (i.e. longest time lines) of 13.5 Vdc and passed. See EID–03897 for time lines and description of the low voltage condition. The Electronic Control Unit is tested for voltage output under thermal extremes as indicated in EID–03846 and EID–03835. As shown in these two reports, the Electronic Control Unit minimum voltage under thermal extremes was always greater than voltage applied to the latch under thermal extremes. Also, there is a lot of margin in terms of current. The latch consumes approximately 1.20 amperes under 13.5 volts, while the Electronic Control Unit is required to have the capability and is tested under thermal conditions to meet a minimum 1.8 amperes.

(3) Motor current, voltage, and phasing: Motor torque availability is proportional to the current output availability of the Electronic Control Unit. Current output availability of the Electronic Control Unit is dependent on motor resistance, voltage input, and internal Electronic Control Unit voltage drop. The nominal torque produced 403 in–lb. Under worst case (thermal, voltage, and resistance extremes) predictions of minimum Electronic Control Unit output of 1.2 amperes, the minimum torque produced by the Beta Gimbal Assembly motor is 348 in–lb (see EID–05139, section 7.2.5). The minimum of 348 in–lb motor torque exceeds the maximum friction of the system (peak drag torque of less than 50 in–lb). During thermal extreme, low voltage input extreme, and resistance (80–ohm) extreme testing, the Electronic Control Unit output was always greater than the needed 1.2 amperes (always above 1.35 amperes) as indicated in EID–03846. Therefore, under worst case thermal, voltage, and life, the Electronic Control Unit/Motor combination torque is greater then the minimum requirements.

(4) Resolver: Resolver error effects pointing accuracy, latching capability, and motor torque.

(a) Pointing accuracy is as indicated in EID–05139, Table 6. This allocation table includes maximum errors for resolver winding, resolver temperature, convector (under temperature extreme), and wind–up for a total of 0.305 degrees. Even with these thermal extreme errors, the Beta Gimbal Assembly will meet the 1.0–degree pointing accuracy with a 0.289 margin (see Table 6). For resolver winding and temperature, see the Roll Ring Honeywell Report # DR55–T–404–32–004. For the convector error under temperature, see EID–03846. Both of these items meet requirements under temperature.

(b) Latching capability: This is a subset of pointing accuracy. The resolver is aligned to the latch 1 (one) location. During thermal vacuum testing, this position was monitored during every latching attempt and varied by no more than the 0.02 degrees between extremes. Allowable is 0.7 degrees error; therefore, margin exists under thermal extremes.

(c) Motor torque: Phasing motor torque has a dependency on resolver error. Torque will drop in power based on the equation: Cos(error*32). At 0.305 degrees error, the result is 98.6 percent torque or a drop of 1.4 percent torque (about 5 in–lb) under thermal extremes. Subtracted from the 348 in–lb minimum in (3) above, the result is 343 in–lb; therefore, sufficient margin remains.

PG2–91:

ITEM:

Beta Gimbal Assembly (BGA) Part Number R075800

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The fully–assembled Beta Gimbal Assembly will not undergo an acceptance thermal vacuum test.

RATIONALE:

The Beta Gimbal Assembly and its internal Electronic Control Unit experiences Acceptance Thermal Vacuum test environments independently, with a functional test of the fully–assembled unit performed at ambient conditions.

The root concern is whether drift of the Beta Gimbal Assembly or the Electronic Control Unit electrical parameters under thermal extremes would result in anomalous performance. The following shows that drift is either non–existent under thermal conditions or that acceptable drift (i.e. off nominal, but still meeting specification) will not effect the Beta Gimbal Assembly operation.

All individual hardware components meet specified requirements under thermal extremes as indicated in the following documents:

(1) Beta Gimbal Assembly (excluding Electronic Control Unit) EID–03897

(2) Roll Ring subassembly, Honeywell Report #DR55–T–404–23–004.

(3) Electronic Control Unit, EID–03846 for qualification unit and EID–03835 for the Engineering Model ECU used in the Beta Gimbal Assembly Qualification Test (this later report also describes the differences between the Engineering Model and Flight Electronic Control Units).

There are two primary functions where the Electronic Control Unit and Beta Gimbal Assembly interface:

(1) Anti–rotation latch function involving the anti–rotation latch (RE2757) which includes the following hardware interfaces:

(a) Limit switches

(b) 15 V power supply

(2) Positioning function involving the motor (RE2739) and resolver (part of RE1822) which includes the following hardware interfaces:

(a) Motor current lines (3, 1 for each phase)

(b) Resolver excitation and cosine/sin feedback.

The following shows that drift under thermal is acceptable when applied to the specific hardware interfaces:

(1) Anti–rotation latch limit switch interface: Limit switch resistances (<10 ohms closed, >10 megaohms open) are tested under thermal conditions to meet requirements (as indicated in the latch acceptance data packages). The Electronic Control Unit is tested under thermal conditions with the maximum specification resistances (i.e. 10 ohm and 10 megaohms) to ensure that the limit switch function works even if both pieces of hardware are at thermal extremes.

(2) Anti–rotation latch power (voltage and current): The anti–rotation latch will work under voltage variances due to the heating strip design (lower voltages result in longer time frames, but will still operate). During Beta Gimbal Assembly thermal vacuum testing, the latch was powered under a minimum specification voltage condition (i.e. longest time lines) of 13.5 Vdc and passed. See EID–03897 for time lines and description of the low voltage condition. The Electronic Control Unit is tested for voltage output under thermal extremes as indicated in EID–03846 and EID–03835. As shown in these two reports, the Electronic Control Unit minimum voltage under thermal extremes was always greater than voltage applied to the latch under thermal extremes. Also, there is a lot of margin in terms of current. The latch consumes approximately 1.20 amperes under 13.5 volts, while the Electronic Control Unit is required to have the capability and is tested under thermal conditions to meet a minimum 1.8 amperes.

(3) Motor current, voltage, and phasing: Motor torque availability is proportional to the current output availability of the Electronic Control Unit. Current output availability of the Electronic Control Unit is dependent on motor resistance, voltage input, and internal Electronic Control Unit voltage drop. The nominal torque produced 403 in–lb. Under worst case (thermal, voltage, and resistance extremes) predictions of minimum Electronic Control Unit output of 1.2 amperes, the minimum torque produced by the Beta Gimbal Assembly motor is 348 in–lb (see EID–05139, section 7.2.5). The minimum of 348 in–lb motor torque exceeds the maximum friction of the system (peak drag torque of less than 50 in–lb). During thermal extreme, low voltage input extreme, and resistance (80–ohm) extreme testing, the Electronic Control Unit output was always greater than the needed 1.2 amperes (always above 1.35 amperes) as indicated in EID–03846. Therefore, under worst case thermal, voltage, and life, the Electronic Control Unit/Motor combination torque is greater then the minimum requirements.

(4) Resolver: Resolver error effects pointing accuracy, latching capability, and motor torque.

(a) Pointing accuracy is as indicated in EID–05139, Table 6. This allocation table includes maximum errors for resolver winding, resolver temperature, convector (under temperature extreme), and wind–up for a total of 0.305 degrees. Even with these thermal extreme errors, the Beta Gimbal Assembly will meet the 1.0–degree pointing accuracy with a 0.289 margin (see Table 6). For resolver winding and temperature, see the Roll Ring Honeywell Report # DR55–T–404–32–004. For the convector error under temperature, see EID–03846. Both of these items meet requirements under temperature.

(b) Latching capability: This is a subset of pointing accuracy. The resolver is aligned to the latch 1 (one) location. During thermal vacuum testing, this position was monitored during every latching attempt and varied by no more than the 0.02 degrees between extremes. Allowable is 0.7 degrees error; therefore, margin exists under thermal extremes.

(c) Motor torque: Phasing motor torque has a dependency on resolver error. Torque will drop in power based on the equation: Cos(error*32). At 0.305 degrees error, the result is 98.6 percent torque or a drop of 1.4 percent torque (about 5 in–lb) under thermal extremes. Subtracted from the 348 in–lb minimum in (3) above, the result is 343 in–lb; therefore, sufficient margin remains.

PG2–92:

ITEM:

Pump Flow Control Subassembly Part Number RE2814

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3, Test Level and Duration.

B. Method II. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The external test pressure shall be 0.050 Torr (6.67 Pa) or less, and the duration of the test shall be fifteen minutes minimum.

RATIONALE:

The maximum allowable leakage rate for the PFCS is 4E–04 standard cubic centimeters per second of helium. The chamber pressure is critical for vacuum drying of unit under test potential leak paths and to obtain the proper level of mass spectrometer sensitivity. 0.05 Torr is sufficient for vacuum drying and it permits a sufficient level of mass spectrometer sensitivity for the leak rate specified for the PFCS.

The purpose of the four–hour test requirement is to ensure the unit under test has been positively pressurized for a sufficient duration to achieve steady state flow through all leak paths. The PFCS leak test procedure and test facility performance dictate that the PFCS is under high positive pressure (greater than 240 psia) for approximately eleven hours prior to the first leak rate measurement. The time is sufficient to achieve a steady state flow rate through all possible leaks.

PG2–93:

ITEM:

Pump Flow Control Subassembly Part Number RE2814

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3, Test Level and Duration.

B. Method II. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The external test pressure shall be 0.050 Torr (6.67 Pa) or less, and the duration of the test shall be fifteen minutes minimum.

RATIONALE:

The maximum allowable leakage rate for the PFCS is 4E–04 standard cubic centimeters per second of helium. The chamber pressure is critical for vacuum drying of unit under test potential leak paths and to obtain the proper level of mass spectrometer sensitivity. 0.05 Torr is sufficient for vacuum drying and it permits a sufficient level of mass spectrometer sensitivity for the leak rate specified for the PFCS.

The purpose of the four–hour test requirement is to ensure the unit under test has been positively pressurized for a sufficient duration to achieve steady state flow through all leak paths. The PFCS leak test procedure and test facility performance dictate that the PFCS is under high positive pressure (greater than 240 psia) for approximately eleven hours prior to the first leak rate measurement. The time is sufficient to achieve a steady state flow rate through all possible leaks.

PG2–94:

ITEM:

Pump Flow Control Subassembly Part Number RE2814, Serial Number 00007

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3, Test Level and Duration.

B. Method II. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The external test pressure shall not be documented and the duration of the test shall be ten minutes minimum. Any subsequent Leak Testing will be compliant with PG2–93.

RATIONALE:

The maximum allowable leakage rate for the PFCS is 4E–04 standard cubic centimeters per second of helium. The chamber pressure is critical for vacuum drying of unit under test potential leak paths and to obtain the proper level of mass spectrometer sensitivity. Vacuum tests conducted on the chamber subsequent to the PFCS leak test showed a vacuum capability of 0.025 Torr (6.67 Pa) or less is sufficient for vacuum drying and it permits a sufficient level of mass spectrometer sensitivity for the leak rate specified for the PFCS.

The purpose of the four–hour test requirement is to ensure the unit under test has been positively pressurized for a sufficient duration to achieve steady state flow through all leak paths. The PFCS leak test procedure and test facility performance dictate that the PFCS is under high positive pressure (greater than 240 psia) for approximately 3–5 hours including 30–60 minutes at vacuum prior to the first leak rate measurement. The time is sufficient to achieve a steady state flow rate through all possible leaks.

Additional Helium sensitivity test performed by Hamilton Sunstrand using the same vacuum chamber and mass spectrometer demonstrated that the 10 minute–test was done with an error of approximately 7.85. Adjusted by this factor toward a more conservative value, the leakage rates of PFCS serial number 00007 (0.50E–04 and 0.53E–04 scc/sec He) are in the same data scatter range as all other PFCS assemblies (average leakage rate is 1.34E–04 scc/sec He).

The Acceptance Test Procedures are consistent with requirements and will not be changed.

PG2–95

ITEM:

Flow Control Valve (FCV) Part Number SV809907–2

SSP 41172 REQUIREMENT:

Paragraph 4.2.13, Life Test, Component Qualification.
Paragraph 4.2.13.3C, Test Level and Duration. The total operating time or number of operational cycles for a component life test shall be twice that predicted during the service life, including ground testing, to demonstrate adequate margin.

EXCEPTION:

The Flow Control Valve life shall not be subjected to the full duration of life testing specified in SSP 41172 as indicated.

Item	Life Test Requirement 2 Times Spec Life RC2814	Percent Completed during Testing
FCV Start/Stop Cycles	5,451,200	33%
FCV Directional Changes	2,550,400	31%
FCV Degrees of Rotation	119,536,000	36%

RATIONALE:

RC2814 3.7.11.2.1 FCV Degrees Traveled – The FCV shall accommodate a minimum of 59,768,000 degrees of rotation.

RC2814 3.7.11.2.2 FCV Directional Changes – The FCV shall accommodate a minimum of 1,275,200 directional changes.

RC2814 3.7.11.2.3 FCV Start/Stop Cycles – The FCV shall accommodate a minimum of 2,725,600 start/stop cycles.

Flow Control Valve test data and analysis data in EID–05051 supports FCV End of Life requirements for leakage at Thermal Control System level. Early External Active Thermal Control System Temperature Control Algorithm Analysis EID–03003 documents technical rationale that supports FCV end of life calculations. The Flow Control Valve specification life cycle requirements are 2,725,600 Stop/Starts, 1,275,200 Directional Changes, and 59,768,000 Degrees Traveled. These values envelop the worst case FCV life cycle requirements. The Flow Control Valve Life Test was terminated early because of program schedule issues caused by the late addition of a PFCS Type I qualification test, and subsequent Functional Configuration Audit. Completion of the FCV component life test did not support the Type I PFCS qualification test. Two Type I PFCS units are flown on ISS Flight 4A.

The method utilized to verify PFCS FCV life at End of Life was by test and analysis. The Hamilton Sundstrand qualification test report SVSHER 18514 revision A (Appendix A) titled PFCS Qualification Endurance Test Report Design Analysis concludes that the test as run cannot yield any reason to conclude that the FCV will not meet its life and cycle requirements. The following rationale is addressed in report.

Constant hysteresis since the FCV was assembled shows the gears are not wearing out.

Low seal leakage indicates the seals are operating as expected.

The increase in flow hysteresis from 2 percent to 2.9 percent at termination of life test is well below the 8 percent limit.

Test data on FCV hysterisis is limited to the beginning of test and end of test data points. The ability to record hysterisis data during test was limited by the test rig setup. Currently there is no plan to disassemble the FCV to inspect for wear. The PFCS qualification unit with subject FCV unit installed is planned for use in support of on–orbit anomaly resolution under ISS Sustaining Engineering contract.

Photovoltaic Module Thermal Control System PFCS FCV Application

The cycle requirements in PFCS specification are based on worst case scenarios and are therefore very conservative. FCV calculations are based on continuous (every orbit) anti–freeze protection. This results in maximum valve movements for low power/cold environments. This condition is the worst case operational scenario. Nominal Photovoltaic Module Thermal Control System operation (full Electrical Power System power) approximates the FCV valve movement at 50 percent of the PFCS FCV specification requirement.

Early External Active Thermal Control System PFCS FCV Application

A 10 percent margin exists in the PFCS FCV specification requirement for valve movements in the Early External Active Thermal Control System application.

The PFCS has only one FCV and excessive degradation would cause excessive hysteresis and/or seal leakage. This condition would cause some loss of temperature control capability under nominal conditions. Under worst case conditions, low power/cold environments, software modifications or operational workarounds can be performed to accommodate Thermal Control System temperature control stability if the FCV control limits exceed requirements. A PFCS removal and replacement option is also available after Flight 5A.

The PFCS will have spares on–orbit. During ISS Flights 4A through 12A, one PFCS will be spared on–orbit. After 12A the Thermal Control System will have the two Early External Active Thermal Control System PFCS ORUs as on–orbit spaces. The two Early External Active Thermal Control System PFCS used as on–orbit spares after 12A will have used up their FCV design life during use on Early External Active Thermal Control System. Their use as spares is over and above their FCV design life. Depot spares has accounted for this by procurement of two complete PFCS manifold assemblies with FCV installed, and two additional spare FCV units for use on PFCS ORU refurbishment.

PG2–96:

ITEM:

Fluid Quick Disconnect Coupling Part Number RE2800

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3B, Test Level and Duration. Method II. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be a minimum of one hour for the Fluid Quick Disconnect Coupling.

RATIONALE:

The purpose of the four–hour test requirement is to ensure the unit under test has been positively pressurized for a sufficient duration to achieve steady–state flow through all leak paths. Test procedure requires recording a leakage rate every 15 minutes. The last 3 consecutive readings must be within .0001 scc/sec Helium of each other; otherwise, the test shall continue in 15–minute intervals until the .0001 limit is met. Only then is the last reading recorded.

Stabilization of the mass spectrometer readings is a valid factor that proves that a steady–state flow rate through all possible leak paths is achieved.

PG2–97:

ITEM:

Fluid Quick Disconnect Coupling Part Number RE2800

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3B, Test Level and Duration. Method II. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be a minimum of one hour for the Fluid Quick Disconnect Coupling.

RATIONALE:

The purpose of the four–hour test requirement is to ensure the unit under test has been positively pressurized for a sufficient duration to achieve steady–state flow through all leak paths. Test procedure requires recording a leakage rate every 15 minutes. The last 3 consecutive readings must be within .0001 scc/sec Helium of each other; otherwise, the test shall continue in 15–minute intervals until the .0001 limit is met. Only then is the last reading recorded.

Stabilization of the mass spectrometer readings is a valid factor that proves that a steady–state flow rate through all possible leak paths is achieved.

PG2–98:

ITEM:

Fluid Quick Disconnect Coupling Part Number RE2800

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F during the cold portion of the cycle.

EXCEPTION:

The qualification thermal margin of 20 degrees F beyond the minimum and maximum operating temperature will not be achieved during the qualification thermal vacuum test of the Fluid Quick Disconnect Coupling. A zero degree F margin is acceptable.

RATIONALE:

Fluid Quick Disconnect Coupling Specification Requirements are as follows:
RC2800 3.2.1.47 Temperature – Nonoperating – The Fluid Quick Disconnect Coupling shall be capable of withstanding the temperature range of –107 degrees Fahrenheit at 180 psid to 120 degrees Fahrenheit at 300 psid.

RC2800 3.2.1.48 Temperature – Operating – The Fluid Quick Disconnect Coupling shall be capable of operating continuously over the temperature range of –107 degrees Fahrenheit at 180 psid to 85 degrees Fahrenheit at 280 psid.

NOTE: The Fluid Quick Disconnect Coupling is required to be actuated only over the temperature range of –80 to 85 degrees Fahrenheit.

The Fluid Quick Disconnect Coupling was Qualification Thermal Vacuum tested for three cycles per SSP 41172 requirements except that no margin over the minimum and maximum operating acceptance temperature levels was achieved. Functional testing (mate/demate) was performed at –80 degrees F and 85 degrees F after completion of low and upper temperature level testing. Non–functional (i.e., no mating/demating) testing was conducted at –117 degrees F to 140 degrees F. The predicted on–orbit temperatures are –107 degrees F to 118 degrees F for the operational (mate/demate) case and –102 degrees F to 79 degrees F for the non–operational (no mate/demate) case.

The purpose of the 20 degrees F margin during qualification thermal vacuum testing is to demonstrate hardware design margin for the worst–case service temperature conditions, including acceptance thermal vacuum testing, as well as to cover unit–to–unit variability between any flight article and the qualification article. This minimizes the risk of excessive acceptance thermal vacuum test failures due to a marginal design for the acceptance temperature environments. All Fluid Quick Disconnect Couplings have passed acceptance thermal vacuum testing under the worst–case predicted temperatures; therefore, the risk due to the lack of demonstrated qualification margin is somewhat mitigated.

The Baseline Assembly Sequence has Fluid Quick Disconnect Coupling mating only once on the ground and once more on–orbit for relocation of the Photovoltaic Module P6. The Fluid Quick Disconnect Coupling is Operational when the active half and passive half are mated and fluid lines are pressurized with gaseous Nitrogen. The Fluid Quick Disconnect Coupling is Non–Operational when the fluid lines are pressurized with gaseous Nitrogen during mate/demate functional operations.

PG2–99:

ITEM:

Fluid Quick Disconnect Coupling Part Number RE2800

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F during the cold portion of the cycle.

EXCEPTION:

The qualification thermal margin of 20 degrees F beyond the minimum and maximum operating temperature will not be achieved during the qualification thermal cycle test of the Fluid Quick Disconnect Coupling. A zero degree F margin is acceptable.

RATIONALE:

Fluid Quick Disconnect Coupling Specification Requirements are as follows:
RC2800 3.2.1.47 Temperature – Nonoperating – The Fluid Quick Disconnect Coupling shall be capable of withstanding the temperature range of –107 degrees Fahrenheit at 180 psid to 120 degrees Fahrenheit at 300 psid.

RC2800 3.2.1.48 Temperature – Operating – The Fluid Quick Disconnect Coupling shall be capable of operating continuously over the temperature range of –107 degrees Fahrenheit at 180 psid to 85 degrees Fahrenheit at 280 psid.

NOTE: The Fluid Quick Disconnect Coupling is required to be actuated only over the temperature range of –80 to 85 degrees Fahrenheit.

The Fluid Quick Disconnect Coupling was Qualification Thermal Cycle tested for twenty–four cycles per SSP 41172 requirements except that no margin over the minimum and maximum on–orbit operating temperature levels was achieved. Functional testing (mate/demate) was performed at –80 degrees F and 85 degrees F after completion of low and upper temperature level testing. Non–functional (i.e., no mating/demating) testing was conducted at –117 degrees F to 140 degrees F. The predicted on–orbit temperatures are –107 degrees F to 118 degrees F for the operational (mate/demate) case and –102 degrees F to 79 degrees F for the non–operational (no mate/demate) case.

The purpose of the 20 degrees F margin during qualification thermal cycle testing is to demonstrate hardware design margin for the worst–case service temperature conditions, including acceptance thermal cycle testing, as well as to cover unit–to–unit variability between any flight article and the qualification article. This minimizes the risk of excessive acceptance thermal cycle test failures due to a marginal design for the acceptance temperature environments. The Fluid Quick Disconnect Couplings do not undergo acceptance thermal cycle testing; therefore, the risk associated with not having the required qualification margin is minimal. Acceptance verification for worst–case predicted temperatures is conducted by acceptance thermal vacuum testing.

The Baseline Assembly Sequence has Fluid Quick Disconnect Coupling mating only once on the ground and once more on–orbit for relocation of the Photovoltaic Module P6. The Fluid Quick Disconnect Coupling is Operational when the active half and passive half are mated and fluid lines are pressurized with gaseous Nitrogen. The Fluid Quick Disconnect Coupling is Non–Operational when the fluid lines are pressurized with gaseous Nitrogen during mate/demate functional operations.

PG2–100:

ITEM:

Direct–Current to Direct–Current Converter Unit – External (DDCU–E) Part Number R076522–121, Serial Numbers X650373 and X650374

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:
B. Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The qualification random vibration test level for the DDCU–E shall not be required to envelope the acceptance test level in the X and Y axes for DDCU–E, Serial Number X650373 and Serial Number X650374 for acceptance testing conducted prior to August 10, 2000. Any future acceptance vibration testing of Serial Number X650373 and Serial Number X650374 shall be required to be conducted using the same test configuration and test control strategy as baselined in the acceptance test procedure as of August 10, 2000.

RATIONALE:

DDCU–E Serial Number X650373 and Serial Number X650374 were acceptance vibration tested prior to conducting the formal qualification vibration test. During vibration acceptance testing in the Y–axis for each of these two units, a dynamic coupling between the shaker and the ORU resulted in a slight high–side out–of tolerance condition at 580 Hertz. To successfully complete testing of these units, the allowable tolerance band was revised around this resonance. When formal qualification testing was later performed, it was decided to eliminate this dynamic coupling problem by incorporating an additional fixture between the slip plate and the fixture to which the ORU attaches. Incorporation of this additional fixture eliminated the coupling problem for Y–axis control. To minimize test time, this additional fixture was also used for the X–axis vibration although no control problems were experienced without it. Qualification was successfully completed utilizing this additional fixture in the X and Y–axes. All subsequent DDCU–E flight units were acceptance tested utilizing this fixture in the X and Y–axes.

In order to determine if the as–qualified configuration adequately enveloped the acceptance configuration of Serial Numbers X650373 and X650374, internal response accelerometer data from a development vibration test conducted in June, 1997 was compared to equivalent internal response accelerometer data from the formal qualification test. The development test was conducted on a high–fidelity engineering model (EM07) of the DDCU–E without the presence of the additional fixture in the X and Y–axes. In addition, the development test was conducted at slightly higher input vibration levels than the formal qualification level. Cross axis data for comparison was limited; however, where that cross–axis data existed, it tended to indicate that the internal response of the ORU is generally insensitive to the presence or absence of the additional fixture. The lack of sufficient internal cross–axis data at all critical internal locations in all axes prevents drawing firm conclusions between the two test configurations.

As a result of the above assessment, a decision as to whether additional qualification testing was required due to the different test fixture configuration was addressed. It was decided that additional qualification testing was neither necessary nor prudent based upon the following considerations: the total accumulated qualification test time already on the qualification DDCU–E; that it may be difficult, if not impossible, to establish whether a failure was due to the test fixture difference rather than a normal end–of–life failure; and review of the available internal response data. The possibility that the acceptance testing on units Serial Numbers X650373 and X650374 has excessively extracted fatigue life out of these units was deemed minimal, and thus a "use–as–is" disposition is warranted. Any future acceptance testing of these units will be carefully addressed so as not to perform unnecessary testing and will be performed with the additional fixture in the X and Y axes to ensure it is performed in the as–qualified configuration.

PG2–101:

ITEM:

Flex Hose Part Number RE4324–01 and RE4324–06

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3A, Test Level and Duration. The duration of immersion shall be 60 minutes at each pressure.

EXCEPTION:

The duration of immersion shall be 1 minute for the RE4324–01 Flex Hose.

The duration of immersion shall be 15 minutes for the RE4324–06 Flex Hose.

RATIONALE:

RE4324–01: Senior Flexonics Qualification Flex Hose (P/N 1812519–90, S/N 0001) was leak tested at 600+/–20 pounds per square inch gauge (psig) for 1 minute, passing the "No Visible Leakage" criteria in effect at the time (Test Report QTR 12–3401), per the requirements of RE4324, Revision A. Similarly, the Flight RE4324–01 Flex Hoses were also leak tested for 1 minute.

Flight RE4324–01 Flex Hoses were installed in the EAS Protoflight acoustic test article prior to acoustic testing. An integrated end item vacuum chamber leak test in accordance with SSP 41172, Method II, was conducted after the acoustic test to ensure the integrity of the fully assembled EAS prior to delivery. The measured leakage rate was 2.5E–04 scc/sec Helium, after stability of the Mass Spectrometer Leak Detector was reached over three consecutive readings in 5–minute intervals. This meets the assembly–level requirement to not exceed 6.8E–04 scc/sec Helium (RJ00342, paragraph 3.2.1.1.3).

RE4324–06: Qualification Test Procedure QTP 19055, Paragraph 4.2.4 states “FMH Corp. shall provide a completed copy of ATP 19055 Data Sheet 1 to satisfy the qualification test requirements for leak testing as defined in Rocketdyne Source Control Drawing RE4324.” RE4324, General Note 32, applicable to RE4234–01 and RE4234–06, as amended by EO R 436814, requires that: “The hose assembly shall be submerged in water and pressurized with gaseous Helium at 600 psig for a holding period of 15 minutes minimum without visible leakage. Apply pressure prior to submerging into deionized water.”

Technical rationale is that 10 minutes is sufficient to form one bubble at the average 0.06 cc size and therefore is enough to verify a leakage rate of 1E–04 standard cubic centimeters per second (scc/sec) Helium. The immersion test time of 15 minutes is sufficient to detect the specified leakage rate of 1E–04 scc/sec Helium. Fifteen minutes is 50 percent longer than the duration of 10 minutes required to form one bubble. If one 0.06 cc bubble was released in 10 minutes, then the equivalent of 1E–04 scc/sec Helium leak rate is present.

The bubble emission leak test is a valid method for leak detection down to 1E–04 scc/sec Helium in accordance with the Nondestructive Testing Handbook (Second Edition, 1985, Volume One, Leak Testing, American Society for Nondestructive Testing) and Leakage Testing Handbook (Revised Edition, July 1969, NASA–CR–106139). Thus, for the purposes of a gross leak test, the duration performed is acceptable.

PG2–102:

ITEM:

Flex Hose Part Number RE4324–01 and RE4324–06

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3A, Test Level and Duration. The duration of immersion shall be 60 minutes at each pressure.

EXCEPTION:

The duration of immersion shall be 1 minute for the RE4324–01 Flex Hose.

The duration of immersion shall be 15 minutes for the RE4324–06 Flex Hose.

RATIONALE:

RE4324–01: Senior Flexonics Flight Flex Hoses RE4324–01 were leak tested at 600+/–20 psig for 1 minute as a component.

Flight RE4324–01 Flex Hoses were installed in the EAS Protoflight acoustic test article prior to acoustic testing. An integrated end item vacuum chamber leak test in accordance with SSP 41172, Method II, was conducted after the acoustic test to ensure the integrity of the fully assembled EAS prior to delivery. The measured leakage rate was 2.5E–04 scc/sec Helium, after stability of the Mass Spectrometer Leak Detector was reached over three consecutive readings in 5–minute intervals. This meets the assembly–level requirement to not exceed 6.8E–04 scc/sec Helium (RJ00342, paragraph 3.2.1.1.3).

Future replacement Flex Hoses for RE4324–01 Flex Hoses will be developed under a new dash number. The Flex Hose Source Control Drawing RE4324 for these will include a leak test duration of 15 minutes.

RE4324–06: Acceptance Test Procedure ATP 19055, paragraph 4.2.2.1, which implements the leak test requirements of RE4324, General Note 32, as amended by EO R 436814, for the RE4324–06 assembly states: “At a temperature range of 70 degrees F +/– 10 degrees F, the Flex Hose Assembly shall be pressurized using gaseous Helium to 600 +/– 20 psig. After pressurization, submerge the entire assembly in deionized water and gently agitate by hand to dislodge any trapped gas bubbles from the exterior of the assembly. After a waiting period of 3 minutes to allow any trapped gas bubbles to escape, begin a timed observation period of 15 minutes duration. During this period, observation of one or more bubbles shall constitute failure of the paragraph 4.2.2 Leak Test. Should failure occur, repeat the entire leak test procedure. A second failure due to observed gas bubbles shall constitute failure of the Acceptance Test.”

Technical rationale is that 10 minutes is sufficient to form one bubble at the average 0.06 cubic centimeter (cc) size and therefore is enough to verify a leakage rate of 1E–04 scc/sec Helium. The immersion test time of 15 minutes is sufficient to detect the specified leakage rate of 1E–04 scc/sec Helium. Fifteen minutes is 50 percent longer than the duration of 10 minutes required to form one bubble. If one 0.06 cc bubble was released in 10 minutes, then the equivalent of 1E–04 scc/sec Helium leak rate is present.

The bubble emission leak test is a valid method for leak detection down to 1E–04 scc/sec Helium per the Nondestructive Testing Handbook (Second Edition, 1985, Volume One, Leak Testing, American Society for Nondestructive Testing) and Leakage Testing Handbook (Revised Edition, July 1969, NASA–CR–106139).

Thus, for the purposes of a gross leak test, the duration performed is acceptable.

PG2–103:

ITEM:

Vent Valve Part Number RE4301–01

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3B, Test Level and Duration. The external pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The duration of the SSP 41172, Revision T, Method II leak test of the Vent Valve shall be a minimum of 9 minutes.

RATIONALE:

The purpose of the four–hour test requirement is to ensure the Unit Under Test has been positively pressurized for a sufficient duration to achieve steady state flow through all leak paths, and that Helium which has leaked into the vacuum enclosure has equilibrated and a stable, accurate Mass Spectrometer Leak Detector reading is being obtained. Stabilization of the Mass Spectrometer Leak Detector readings is a valid factor that proves that steady state flow rate through all possible leaks is achieved.

The vacuum enclosure housing the Early Ammonia Servicer Vent Valve has a net evacuated volume of less than 2 cubic inches. The vacuum enclosure is evacuated to a stable vacuum of less than 1E–08 Torr, prior to initiation of the Mass Spectrometer Leak Detector. The leak rate data is monitored until a stable leak rate is established, at which point the leak rate data is taken for a period of 9 minutes to establish the leak results which is assessed against the requirement of less than 1E–05 scc/sec leak rates using Helium.

A Vent Valve leak test system sensitivity test was conducted as described below and showed that the test duration of 9 minutes is adequate to achieve a stable leak test using this leak test system architecture:

Testing Method: Assemble test unit into test fixture with 0 psig applied to the inlet of the test unit, and outlet of valve capped off, evacuate the inside of the test fixture external to the test unit, using a Mass Spectrometer Leak Detector. Verify background leak rate. Allow background leak rate to stabilize. Confirm the background leak rate to be stable within 10 percent over duration of 3 minutes. After background verification, begin leak test. Apply 550 psig to the inlet of the fixture and monitor the leak rate external to the valve over 15 minutes. Record leak rate each minute. Plot results. Verify stability of leak rate. Vent pressure to inlet. Vent Mass Spectrometer Leak Detector and remove test unit from fixture.

The smallest detectable leak rate for the Balzer HLT270 Mass Spectrometer Leak Detector is 4.93E–12 scc/sec. During the Vent Valve leak test using the methodology above, the documented result for the Vent Valve was 4.93E–12 scc/sec. As indicated, the test was conducted at 550 psig , which provided a 1.8 factor over the maximum operating pressure of 300 psig.

PG2–104:

ITEM:

Vent Valve Part Number RE4301–01

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The duration of the Method II leak test of the Vent Valve shall be a minimum of 9 minutes.

RATIONALE:

The purpose of the four–hour test requirement is to ensure the Unit Under Test has been positively pressurized for a sufficient duration to achieve steady state flow through all leak paths, and that Helium which has leaked into the vacuum enclosure has equilibrated and a stable, accurate Mass Spectrometer Leak Detector reading is being obtained. Stabilization of the Mass Spectrometer Leak Detector readings is a valid factor that proves that steady state flow rate through all possible leaks is achieved.

The vacuum enclosure housing the Early Ammonia Servicer Vent Valve has a net evacuated volume of less than 2 cubic inches. The vacuum enclosure is evacuated to a stable vacuum of less than 1E–08 Torr, prior to initiation of the Mass Spectrometer Leak Detector. The leak rate data is monitored until a stable leak rate is established, at which point the leak rate data is taken for a period of 9 minutes to establish the leak results which is assessed against the requirement of less than 1E–05 scc/sec Helium leak rate.

A Vent Valve leak test system sensitivity test was conducted as described below and showed that the test duration of 9 minutes is adequate to achieve a stable leak test using this leak test system architecture.

Testing Method: Assemble test unit into test fixture with 0 psig applied to the inlet of the test unit, and outlet of valve capped off, evacuate the inside of the test fixture external to the test unit, using a Mass Spectrometer Leak Detector. Verify background leak rate. Allow background leak rate to stabilize. Confirm the background leak rate to be stable within 10 percent over duration of 3 minutes. After background verification, begin leak test. Apply 550 psig to the inlet of the fixture and monitor leak rate external to the valve over 15 minutes. Record leak rate each minute. Plot results. Verify stability of leak rate. Vent pressure to inlet. Vent the Mass Spectrometer Leak Detector and remove test unit from fixture.

The smallest detectable leak rate for the Balzer HLT270 Mass Spectrometer Leak Detector is 4.93E–12 scc/sec. During the Vent Valve leak test using the methodology above, the documented result for the Vent Valve was 4.93E–12 scc/sec. As indicated, the test was conducted at 550 psig, which provided a 1.8 factor over the maximum operating pressure of 300 psig.

Future Vent Valves will be developed under a new dash number. The Vent Valve Acceptance Test Procedure HTP–7346–01 for these will include a leak test duration of 15 minutes.

PG2–105:

ITEM:

Vent Valve Part Number RE4301–01

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.2, Test Description. Valves shall be pressurized to operating pressure for this test and monitored for internal pressure decay if pressurized during ascent.

EXCEPTION:

The Vent Valve was not pressurized during Qualification Random Vibration Testing.

RATIONALE:

The RE4301–01 Vent Valves are pressurized with sufficient GN2 pad gas (less than 80 psig) to preclude in–leakage of atmosphere during ground processing. This degree of pressurization is not significant enough to overcome the valve seat spring pressure (lift pressure of 270 psig); thus, it is not significant enough to affect the random vibration testing of the Vent Valves, and was not included to simplify the valve fixture design and test architecture.

The Flight Early Ammonia Servicer, with Flight RE4301–01 Vent Valve, has completed Protoflight acoustic testing (141 dB) at the end item level. During this testing, the Vent Valve and its associated plumbing is pressurized with GN2 pad gas representing the Flight configuration (RL01543A), and thus met the intent of SSP 41172, Random Vibration Testing, under the Protoflight test requirements.

The Vent Valves are pressurized with GN2 pad gas (less than 80 psig) during Protoflight acoustic testing at the Early Ammonia Servicer level and are checked before and after test. Even in the event the valve loses pressure, venting to atmospheric pressure during test or launch is not of concern since the pressure is only applied to preclude in–leakage of atmosphere during ground processing.

Per the requirement of pre/post acoustic functional test procedure, the EAS NH3 plumbing, which contains the Vent Valve VV–02, is pressurized to 95 pounds per square inch absolute (psia) to properly simulate the launch conditions of these lines.

Results were as follows:

1) The launch configuration charged the spool from QD–M02 to QD–F4 with 95.1 psia of GN2 prior to acoustic test.

2) Post–Acoustic verification of these lines indicated 94.9 psia of GN2.

Vent Valve VV–02 withstood the acoustic testing environment without self–relieving internal pressure, thus maintained valve leak tightness, and was successful on this parameter for acoustic testing.

PG2–106:

ITEM:

Vent Valve Part Number RE4301–01

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.2, Test Description. Valves shall be pressurized to operating pressure for this test and monitored for internal pressure decay if pressurized during ascent.

EXCEPTION:

The Vent Valve was not pressurized during Acceptance Random Vibration testing.

RATIONALE:

The RE4301–01 Vent Valves are pressurized with sufficient GN2 pad gas (less than 80 psig) to preclude in–leakage of atmosphere during ground processing. This degree of pressurization is not significant enough to overcome the valve seat spring pressure (lift pressure of 270 psig). Thus, it is not significant enough to affect the random vibration testing of the Vent Valves, and was not included to simplify the valve fixture design and test architecture.

The Flight Early Ammonia Servicer, with Flight RE4301–01 Vent Valve, has completed Protoflight acoustic testing (141 dB) at the end item level. During this testing, the Vent Valve and its associated plumbing is pressurized with GN2 pad gas representing the Flight configuration (RL01543A), and thus met the intent of SSP 41172, Random Vibration Testing, under the Protoflight test requirements.

The Vent Valves are pressurized with GN2 pad gas (less than 80 psig) during Protoflight acoustic testing at the Early Ammonia Servicer level and are checked before and after test. Even in the event the valve loses pressure, venting to atmospheric pressure during test or launch is not of concern since the pressure is only applied to preclude in–leakage of atmosphere during ground processing.

Per the requirement of pre/post acoustic functional test procedure, the EAS NH3 plumbing, which contains the Vent Valve VV–02, is pressurized to 95 psia to properly simulate the launch conditions of these lines.

Results were as follows:

1) The launch configuration charged the spool from QD–M02 to QD–F4 with 95.1 psia of GN2 prior to acoustic test.

2) Post–acoustic verification of these lines indicated 94.9 psia of GN2.

Vent Valve VV–02 withstood the acoustic testing environment without self–relieving internal pressure, thus maintained valve leak tightness, and was successful on this parameter for acoustic testing.

PG2–107:

ITEM:

Nitrogen Tank Part Number RE4302–01

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.

EXCEPTION:

The Qualification leak test on the Nitrogen Tank was not performed with sufficient sensitivity and accuracy to verify the maximum allowable leakage rate.

RATIONALE:

The Nitrogen Tank underwent a Qualification leak test for one minute. This duration was not sufficient to verify a leakage rate of 1E–07 scc/sec Helium.

Flight Nitrogen Tanks were installed in the EAS Protoflight acoustic test article prior to acoustic testing. An integrated end item vacuum chamber leak test in accordance with SSP 41172, Method II, was conducted after the acoustic test to ensure the integrity of the fully assembled EAS prior to delivery. The measured leakage rate was 2.5E–04 scc/sec Helium, after stability of the Mass Spectrometer Leak Detector was reached over three consecutive readings in 5–minute intervals. This meets the assembly–level requirement to not exceed 6.8E–04 scc/sec Helium (RJ00342, paragraph 3.2.1.1.3).

PG2–108:

ITEM:

Nitrogen Tank Part Number RE4302–01 Serial Numbers 8833906 and 8833907

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the components specified maximum allowable leak rate.

EXCEPTION:

The Acceptance Leak Test on the Nitrogen Tank indicated was not performed with sufficient sensitivity and accuracy to verify the maximum allowable leakage rate.

RATIONALE:

The Nitrogen Tank underwent an acceptance leak test for one minute. This duration was not sufficient to verify a leakage rate of 1E–07 scc/sec Helium.

Flight Nitrogen Tanks were installed in the EAS Protoflight acoustic test article prior to acoustic testing. An integrated end item vacuum chamber leak test in accordance with SSP 41172, Method II, was conducted after the acoustic test to ensure the integrity of the fully assembled EAS prior to delivery. The measured leakage rate was 2.5E–04 scc/sec Helium, after stability of the Mass Spectrometer Leak Detector was reached over three consecutive readings in 5–minute intervals. This meets the assembly–level requirement to not exceed 6.8E–04 scc/sec Helium (RJ00342, paragraph 3.2.1.1.3).

PG2–109:

ITEM:

Capture Assembly Mechanism Part Number RH000232

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3A, Test Level and Duration. The pressure shall be reduced from atmospheric to below 0.001 Torr (0.133 Pa).
Paragraph 4.2.2.3C, Test Level and Duration. The time required to reach thermal equilibrium shall be determined by pre–qualification analysis or test or by measuring the component's internal thermal response during an extended dwell period of not less than 12 hours at each temperature extreme of the first qualification thermal vacuum cycle.

EXCEPTION:

The pressure of the Capture Assembly Mechanism shall be at ambient and the duration of the first–cycle dwell period shall be one hour during the Qualification Thermal Vacuum test.

RATIONALE:

The Capture Assembly Mechanism (Part Number RH000232 for the Early Ammonia Servicer) is the mechanical interface of the Early Ammonia Servicer with the P6 longeron trunnion. The Capture Assembly Mechanism does not have electronics or close tolerances requiring precise adjustment and can be inspected effectively.

Thermal testing of the Capture Assembly Mechanism used SSP 41172 Qualification Thermal Vacuum test requirements, except to conduct the test at atmospheric pressure and to perform the first–cycle dwell period for a duration of one hour each. The test has been performed with three cycles. This mechanism has no power and, due to its small size and open construction, thermally stabilized in 20 to 30 minutes in the chamber. Thermal testing without vacuum provided adequate fit and clearance testing with minimal risk. All thermal stabilization times were one hour.

Vacuum does not affect the materials used. The Capture Assembly Mechanism uses dry film lubricant per MIL–L–46010, and for the thermal test, the EVA contingency screws (RE112–1016–0023, 4 places) were lubricated with Braycote 601EF. Running torque measurements were made and compared to Flight configuration (i.e., dry–film lubed screws) and found to be acceptable.

PG2–110

ITEM:

Capture Assembly Mechanism Part Number RH000232

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3A, Test Level and Duration. The pressure shall be reduced from atmospheric to below 0.001 Torr (0.133 Pa).

EXCEPTION:

The pressure of the Capture Assembly Mechanism shall be at ambient during the Acceptance Thermal Vacuum test.

RATIONALE:

The Capture Assembly Mechanism (Part Number RH000232 for the Early Ammonia Servicer) is the mechanical interface of the Early Ammonia Servicer with the P6 longeron trunnion. The Capture Assembly Mechanism does not have electronics or close tolerances requiring precise adjustment and can be inspected effectively.

Thermal testing of the Capture Assembly Mechanism used SSP 41172 Acceptance Thermal Vacuum test requirements, except to conduct the test at atmospheric pressure. The test was performed with one cycle. The thermal stabilization time for the single cycle of the acceptance test was determined by the equilibrium established during the Qualification thermal vacuum testing. Thermal testing without vacuum provided adequate fit and clearance testing with minimal risk. All thermal stabilization times were one hour.

Vacuum does not affect the materials used. The Capture Assembly Mechanism uses dry film lubricant per MIL–L–46010, and for the thermal test, the EVA contingency screws (RE112–1016–0023, 4 places) were lubricated with Braycote 601EF. Running torque measurements were made and compared to Flight configuration (i.e., dry–film lubed screws) and found to be acceptable.

PG2–111

ITEM:

Capture Assembly Mechanism Part Number RH000232

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Capture Assembly Mechanism will not experience a component–level Qualification Random Vibration Test.

RATIONALE:

The Capture Assembly Mechanism (Part Number RH000232 for the Early Ammonia Servicer) is the mechanical interface of the Early Ammonia Servicer with the P6 longeron trunnion. The Capture Assembly Mechanism does not have electronics or close tolerances requiring precise adjustment and can be inspected effectively. The Protoflight Acoustic Test performed at the Early Ammonia Servicer Orbital Support Equipment level will provide adequate testing with minimal risk.

The Capture Assembly Mechanism is part of the Protoflight Acoustic Test and is verified at the Early Ammonia Servicer level. Post–acoustic test functional testing has been successfully conducted on the Capture Assembly Mechanism to verify the following:

(1) Drive bolt running torque: 25 in–lb. or less;

(2) Drive bolt seating torque: 306 in–lb. or less;

(3) Soft–Dock latch actuation force: 10 lb. or less in each direction; and

(4) Pip pin removal/replacement force: less than 5 lb. to depress button, less than 5 lb. to transfer pin to/from hole.

PG2–112

ITEM:

Capture Assembly Mechanism Part Number RH000232

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Capture Assembly Mechanism will not experience a component–level Acceptance Random Vibration Test.

RATIONALE:

The Capture Assembly Mechanism (Part Number RH000232 for the Early Ammonia Servicer) is the mechanical interface of the Early Ammonia Servicer with the P6 longeron trunnion. The Capture Assembly Mechanism does not have electronics or close tolerances requiring precise adjustment and can be inspected effectively. The Protoflight Acoustic Test performed at the Early Ammonia Servicer Orbital Support Equipment level will provide adequate testing with minimal risk.

The Capture Assembly Mechanism is part of the Protoflight Acoustic Test and is verified at the Early Ammonia Servicer level. Post–acoustic test functional testing has been successfully conducted on the Capture Assembly Mechanism to verify the following:

(1) Drive bolt running torque: 25 in–lb. or less;

(2) Drive bolt seating torque: 306 in–lb. or less;

(3) Soft–dock latch actuation force: 10 lb. or less in each direction; and

(4) Pip pin removal/replacement force: less than 5 lb. to depress button, less than 5 lb. to transfer pin to/from hole.

PG2–113

ITEM:

Gaseous Nitrogen Pressure Gauge Part Number RE4323–01 Serial Numbers 9E001 and 9E002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.

When the assembly/components qualification tests are conducted on an assembly intended for subsequent Flight, the test shall be the same (as defined in paragraph 4.2 for component qualification) with the following exceptions:

Paragraph 6.1.1A. For the thermal vacuum tests, the temperature extremes shall be 10 degrees F (5.6 degrees C) beyond the minimum and maximum predicted temperatures. The minimum number of cycles shall be one.

Paragraph 6.1.1B. For the thermal cycling test, the temperature cycles shall be conducted at 10 degrees F (5.6 degrees C) beyond the maximum and minimum predicted temperatures. The minimum number of cycles shall be eight.

EXCEPTION:

Thermal testing of the GN2 Pressure Gauges indicated was performed from – 126 degrees F to 167 degrees F.

RATIONALE:

Original gauges (per SLG 39128229) were designed for a thermal environment of 11 degrees F to 120 degrees F. The design and construction of GN2 Pressure Gauge was sufficiently robust due to all metal–brazed and welded construction to permit retesting to the worst–case, Non–Operating thermal requirement of –105 degrees F to 140 degrees F as indicated in Source Control Drawing RE4323, General Note 10.

During the delta testing, two units (Serial Numbers 9E001 and 9E002) were selected and Protoflight Thermal Vacuum tested for 3 cycles over the range of –126 degrees F to 167 degrees F while pressurized to 600 psia. This is beyond the thermal margin defined in the Protoflight test requirement in SSP 41172, 6.1.1A and 6.1.1B. However, a post–test examination, a gauge accuracy test, and leaking testing did not indicated any overstress conditions due to this thermal testing.

Future GN2 Pressure Gauges will be developed under a new dash number. The test procedure for these will include a thermal test from –115 degrees F to 150 degrees F.

PG2–114

ITEM:

In–Line Filter Assembly Part Number RH000474

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.

EXCEPTION:

The Qualification Leak Test on the In–Line Filter Assembly was not performed with sufficient sensitivity and accuracy to verify the maximum allowable leakage rate.

RATIONALE:

Qualification Leak Testing on the In–Line Filter Assembly was performed in accordance with RA0115–105 General Specification for Rockwell International Corporation, Leak Testing with a Helium Mass Spectrometer Leak Detector, as follows:

"Note 16 Helium leak test assembly per RA0115–105 using procedure IV with interior surfaces pressurized with Helium to 550 +/–10 PSIG. System background is to be constant for 15 minutes prior to test and not exceed $9x10^{-7}$ STD cc/sec. The MSLD reading must be constant or decreasing (No rise above background) for a minimum of 10 minutes after the introduction of Helium."

However, the Mass Spectrometer Leak Detector when used with a "sniffer" does not have sufficient sensitivity and accuracy to verify the maximum allowable leakage.

Non–destructive testing (radiographic inspection, dye penetrant, and ultrasonic inspection) was performed on the In–Line Filter Assembly.

Additionally, the In–Line Filter Assembly was installed in the EAS Protoflight Acoustic Test article prior to acoustic testing. An integrated end item vacuum chamber leak test in accordance with SSP 41172, Method II, was conducted after the acoustic test to ensure the integrity of the fully assembled EAS. The measured leakage rate was 2.5E–04 scc/sec Helium, after stability of the Mass Spectrometer Leak Detector was reached over three consecutive readings in 5–minute intervals. This meets the assembly–level requirement to not exceed 6.8E–04 scc/sec Helium (RJ00342, paragraph 3.2.1.1.3).

PG2–115

ITEM:

In–Line Filter Assembly Part Number RH000474

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the components specified maximum allowable leak rate.

EXCEPTION:

The Acceptance Leak Test on the In–Line Filter Assembly was not performed with sufficient sensitivity and accuracy to verify the maximum allowable leakage rate.

RATIONALE:

Acceptance Leak Testing on the In–Line Filter Assembly was performed in accordance with RA0115–105 General Specification for Rockwell International Corporation, Leak Testing with a Helium Mass Spectrometer Leak Detector, as follows:

"Note 16 Helium leak test assembly per RA0115–105 using procedure IV with interior surfaces pressurized with Helium to 550+/–10 PSIG. System background is to be constant for 15 minutes prior to test and not exceed $9x10^{-7}$ STD cc/sec. The MSLD reading must be constant or decreasing (No rise above background) for a minimum of 10 minutes after the introduction of Helium."

However, the Mass Spectrometer Leak Detector when used with a "sniffer" does not have sufficient sensitivity and accuracy to verify the maximum allowable leakage.

Non–destructive testing (radiographic inspection, dye penetrant, and ultrasonic inspection) was performed on the In–Line Filter Assembly.

Additionally, the In–Line Filter Assembly was installed in the EAS Protoflight Acoustic Test article prior to acoustic testing. An integrated end item vacuum chamber leak test in accordance with SSP 41172, Method II, was conducted after the acoustic test to ensure the integrity of the fully assembled EAS. The measured leakage rate was 2.5E–04 scc/sec Helium, after stability of the Mass Spectrometer Leak Detector was reached over three consecutive readings in 5–minute intervals. This meets the assembly–level requirement to not exceed 6.8E–04 scc/sec Helium (RJ00342, paragraph 3.2.1.1.3).

Future Flight In–Line Filter Assemblies will be developed under a new part number. The test procedure for these will include an Acceptance Leakage Test in compliance with SSP 41172.

PG2–116

ITEM:

Early Ammonia Servicer Configuration Item FSE0108A

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.

EXCEPTION:

The Qualification Leak Test on the Gamah Fittings internal to the Early Ammonia Servicer were not performed with sufficient sensitivity and accuracy to verify the maximum allowable leakage rate.

RATIONALE:

Qualification Leak Testing on the Gamah Fitting Assembly was performed in accordance with RA0115–105 General Specification for Rockwell International Corporation, Leak Testing with a Helium Mass Spectrometer Leak Detector, as follows:

“Note 16 Helium leak test assembly per RA0115–105 using procedure IV with interior surfaces pressurized with Helium to 550 +/–10 PSIG. System background is to be constant for 15 minutes prior to test and not exceed $9x10^{-7}$ STD cc/sec. The MSLD reading must be constant or decreasing (No rise above background) for a minimum of 10 minutes after the introduction of Helium.”

However, the Mass Spectrometer Leak Detector when used with a “sniffer” does not have sufficient sensitivity and accuracy to verify the maximum allowable leakage.

Qualification Test Report Gamah JS14402–2 was supplemented with Hamilton Standard SVHSER 19908 June 1999 “External Leakage Requirement for the Ammonia Servicer Gamah Fitting“ to qualify external leakage requirement of 1E–06 scc/sec Helium at a pressure of 550 psid. The measured leakage rate was 2E–08 scc/sec Helium over a period of 5 minutes.

The Gamah Fitting Assembly was installed in the EAS Protoflight Acoustic Test article prior to acoustic testing. An integrated end item vacuum chamber leak test in accordance with SSP 41172, Method II, was conducted after the acoustic test to ensure the integrity of the fully assembled EAS. The measured leakage rate was 2.5E–04 scc/sec Helium, after stability of the Mass Spectrometer Leak Detector was reached over three consecutive readings in 5–minute intervals. This meets the assembly–level requirement to not exceed 6.8E–04 scc/sec Helium (RJ00342, paragraph 3.2.1.1.3).

PG2–117

ITEM:

Early Ammonia Servicer Configuration Item FSE0108A

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Descriptions and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the components specified maximum allowable leak rate.

EXCEPTION:

The Acceptance Leak Test on the Gamah Fittings internal to the Early Ammonia Servicer were not performed with sufficient sensitivity and accuracy to verify the maximum allowable leakage rate.

RATIONALE:

Acceptance Leak Testing on the Gamah Fitting Assembly was performed in accordance with RA0115–105 General Specification for Rockwell International Corporation, Leak Testing with a Helium Mass Spectrometer Leak Detector, as follows:

“Note 16 Helium leak test assembly per RA0115–105 using procedure IV with interior surfaces pressurized with Helium to 550 +/–10 PSIG. System background is to be constant for 15 minutes prior to test and not exceed $9x10^{-7}$ STD cc/sec. The MSLD reading must be constant or decreasing (No rise above background) for a minimum of 10 minutes after the introduction of Helium.”

Acceptance leakage test was performed in accordance with the RA0115–105 Procedure IV requirement of no greater than 9E–07 scc/sec Helium at a pressure of 200 psig for a minimum of 10 minutes. However, the Mass Spectrometer Leak Detector when used with a "sniffer" does not have sufficient sensitivity and accuracy to verify the maximum allowable leakage.

The Gamah Fitting Assembly was installed in the EAS Protoflight Acoustic Test article prior to acoustic testing. An integrated end item vacuum chamber leak test in accordance with SSP 41172, Method II, was conducted after the acoustic test to ensure the integrity of the fully assembled EAS. The measured leakage rate was 2.5E–04 scc/sec Helium, after stability of the Mass Spectrometer Leak Detector was reached over three consecutive readings in 5–minute intervals. This meets the assembly–level requirement to not exceed 6.8E–04 scc/sec Helium (RJ00342, paragraph 3.2.1.1.3).

PG2–118:

ITEM:

NH3 Accumulator Part Number SV809903

SSP 41172 REQUIREMENT:

Paragraph 4.2.1l, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Description and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate.

EXCEPTION:

The NH3 accumulator gas fill tube shall not be verified by test to the Helium leak rate requirement of 6.67E–08 scc/sec.

RATIONALE:

Designed to SSP 30559 requirements for fracture critical hardware.

The accumulator gas fill tube is a robust design utilizing a crimp and welded fill tube with a welded cap (double weld). This design has a very low probability of leaking at a rate greater than specified.

All welds are inspected to MSFC–STD–1249 non–destructive test methods. The personnel and equipment utilized for this special weld flaw detection method are certified and demonstrate a capability of a 90 percent probability of detection at a 95 percent confidence level.

An accumulator gas fill tube leak test per SSP 41172 Method II was performed on a single PIO 40 spare unit (Serial Number 044) and a leak rate better than specified was achieved.

PG2–119:

ITEM:

NH3 Accumulator Part Number SV809903–4 Serial Numbers 17, 30, 31, 33, 36, 37, 38, 39, 40, and 41
NH3 Accumulator Part Number SV809903–5 Serial Numbers 12, 13, 14, 18, 21, 22, 24, 25, 28, 29, 34, and 35

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Description and Alternatives. The test method employed shall have sensitivity and accuracy consistent with the components specified maximum allowable leak rate.

EXCEPTION:

The NH3 accumulator gas fill tube shall not be verified by test to the Helium leak rate requirement of 6.67E–08 scc/sec. Any future acceptance leak testing of NH3 accumulator gas fill tubes shall be in compliance with SSP 41172.

RATIONALE:

Designed to SSP 30559 requirements for fracture critical hardware.

The accumulator gas fill tube is a robust design utilizing a crimp and welded fill tube with a welded cap (double weld). This design has a very low probability of leaking at a rate greater than specified.

All welds are inspected to MSFC–STD–1249 non–destructive test methods. The personnel and equipment utilized for this special weld flaw detection method are certified and demonstrate a capability of a 90 percent probability of detection at a 95 percent confidence level.

An accumulator gas fill tube leak test per SSP 41172 Method II was performed on a single PIO 40 spare unit (Serial Number 044) and a leak rate better than specified was achieved.

PG2–120:

ITEM:

Motor/Motor Controller Part Number 83–36884–101

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The Photovoltaic Radiator Motor and Motor Controller shall have 0 degrees F margin between qualification and acceptance thermal testing for hot operating and nonoperating conditions and cold nonoperating conditions. For cold operating conditions, the Photovoltaic Radiator Motor and Motor Controller shall have a negative 36 degrees F margin (–67 degrees F during acceptance thermal testing and –31 degrees F during qualification thermal testing).

RATIONALE:

The Motor and Motor Controller EEE parts are not designed to be operated below –67 degrees F (–55 degrees C).

Redundancy Backup:
The Motor and Motor Controller has EVA redundancy for deployment. The EVA drive was successfully tested during the Plum Brook Thermal Vacuum testing of PVR–1. It was used to deploy and retract the radiator when the environmental temperature was ambient, –100 degrees F, and 120 degrees F. When the environment was –110 degrees F, the EVA drive temperature was approximately 0 degrees F. During Acceptance testing of the four Photovoltaic Radiators, the EVA drives were successfully tested at ambient temperature.

EEE parts Life and Detection of Electrical Defects:
The life of the EEE parts will not be affected by the acceptance testing at –67 degrees F since they are specified to be operable at –67 degrees F. The Motor and Motor Controller component functional testing at –67 degrees F should have uncovered electrical defects such as cracked solder joints that were either existing or were introduced by the component Thermal or Vibration testing.

Margin of Approximately 67 degrees F Demonstrated Relative to the Minimum On–Orbit Operating Temperature:

At the component level the Motor and Motor Controllers were functioned at –67 degrees F during Acceptance Thermal Vacuum and Thermal Cycle testing. The Motor and Motor Controller is only at cold temperatures (less than 0 degrees F) when it is nonoperational, which is during P6 or Photovoltaic Radiator relocation activities, or Launch to Activation activities. Following these activities, power will be restored to the heaters and the Motor and Motor Controller will be approximately 0 degrees F when operated to deploy/retract a Photovoltaic Radiator.

Margin of Approximately 20 degrees F Demonstrated Relative to the Minimum On–Orbit nonoperating Temperature:

The minimum Motor and Motor Controller nonoperating on–orbit temperature of –47 degrees F occurs only when the heaters are not functional which is during PVM P4 or PVM P6 Launch to Activation, or during the relocation of PVM P6 or an Early Photovoltaic Radiator. During the Launch to Activation or relocation activities the Motor and Motor Controllers are not operated since the Photovoltaic Radiators are not being deployed or retracted.

Margin of 18 degrees F Relative to Minimum EEE Parts nonoperating temperature:

The EEE Parts are specified to be good to –85 degrees F (–65 degrees C) nonoperating. This is 18 degrees F below the minimum Acceptance Thermal Vacuum and Thermal Cycle Test temperatures of –67 degrees F.

PG2–121:

ITEM:

Assembly Level: Motor/Motor Controller Part Number 83–36884–101
Subassembly: Motor Tecstar Part Number 2961062–001
Motor Controller Tecstar Part Number 2961062–501

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The Photovoltaic Radiator Motor and Motor Controller shall not be required to be electrically energized and monitored during qualification random vibration testing. Future testing of the Motor Controllers (Tecstar Part Number 2961062–501) will be performed in compliance with SSP 41172 requirements. Motor Controllers not already installed in Photovoltaic Radiators 1, 2, 3 and 4, any rework of the Motor Controllers already installed, and any spares shall require the Motor Controllers to be electrically energized and monitored during qualification (if required due to hardware changes) and acceptance random vibration testing.

RATIONALE:

The component–level Qualification Vibration testing was performed prior to the component–level Qualification Thermal Vacuum and Thermal Cycle testing. The Motor and Motor Controllers were energized during the Qualification Thermal Vacuum and Thermal Cycle testing which followed.

Thermal testing should have uncovered electrical defects (such as cracked solder joints) which were either existing or were introduced during Vibration testing. This is true as the Motor and Motor Controllers were thermal tested over a large temperature ranges. The ”nonoperating” thermal temperature range was from –67 degrees F to 257 degrees F during Qualification and Acceptance thermal testing. The ”operating” temperature range was from –31 degrees F to 129 degrees F during Qualification and Acceptance thermal testing.

The Motor/Motor Controller has EVA redundancy for deployment.

The Photovoltaic Radiator motor shall not require powering and monitoring during any vibration testing since false indications of failure may result. Operating the motor during vibration testing may result in a questionable continuity strip chart. The brake will chatter during vibration. Since the motor cannot “over power” the brake, the running current value on the strip chart would ”spike” to the stall current value. Thus, during vibration, it would be impossible to differentiate between current spikes cause by partial engagement of the brake and a “failure”.

PG2–122:

ITEM:

Assembly Level: Motor/Motor Controller Part Number 83–36884–101
Subassembly: Motor Tecstar Part Number 2961062–001
Motor Controller Tecstar Part Number 2961062–501

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Qualification.
Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The Photovoltaic Radiator Motor and Motor Controller shall not be required to be electrically energized during acceptance random vibration testing. Future testing of the Motor Controllers (Tecstar Part Number 2961062–501) will be performed in compliance with SSP 41172 requirements. Motor Controllers not already installed in Photovoltaic Radiators 1, 2, 3 and 4, any rework of the Motor Controllers already installed, and any spares shall require the Motor Controllers to be electrically energized and monitored during qualification (if required due to hardware changes) and acceptance random vibration testing.

RATIONALE:

The component–level Acceptance Vibration testing was performed prior to the component–level Acceptance Thermal Vacuum and Thermal Cycle testing. The Motor and Motor Controllers were energized during the Acceptance Thermal Vacuum and Thermal Cycle testing which followed.

Thermal testing should have uncovered electrical defects (such as cracked solder joints) which were either existing or were introduced during Vibration testing. This is true as the Motor and Motor Controllers were thermal tested over a large temperature range. The ”nonoperating” thermal temperature range was from –67 degrees F to 257 degrees F during Qualification and Acceptance thermal testing. The flight units were operated at –67 degrees F. The ”operating” temperature range was from –31 degrees F to 129 degrees F during Qualification and Acceptance thermal testing.

The Motor/Motor Controller has EVA redundancy for deployment.

The Photovoltaic Radiator motor shall not require powering and monitoring during any vibration testing since false indications of failure may result. Operating the motor during vibration testing may result in a questionable continuity strip chart. The brake will chatter during vibration. Since the motor cannot “over power” the brake, the running current value on the strip chart would ”spike” to the stall current value. Thus, during vibration, it would be impossible to differentiate between current spikes cause by partial engagement of the brake and a “failure”.

PG2–123:

ITEM:

Motor/Motor Controller Part Number 83–36884–101 Serial Numbers 002, 003, 004, and 005

SSP 41172 REQUIREMENT:

Paragraph 4.2.5 Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3 Test Levels and Duration The test duration in each of the three orthogonal axes shall be three times the expected flight exposure time to the maximum predicted level and spectrum or three times the component random vibration acceptance test time if that is greater, but not less than three minutes per axis.

EXCEPTION:

The duration of the Photovoltaic Radiator Motor and Motor Controller Qualification Random Vibration test shall be the same as the Acceptance Random Vibration test duration of 3 minutes per axis.

RATIONALE:

The Photovoltaic Radiator Motor and Motor Controllers indicated underwent Acceptance Vibration tests for 3 minutes per axis; however, the Qualification Vibration test was also 3 minutes per axis. Fatigue analyses (EID–05680) shows that the electronics were not damaged and have adequate life. The analysis indicated:

Motor: the minimum remaining life is 165 minutes (at flight levels); and

Motor Controller: the remaining life is 141 minutes (at flight levels).

For the motor, one flight is equivalent to 7 seconds of acceptance test time (worst–case z–axis). The minimum remaining life of 165 minutes equates to 330 flights or 38 acceptance tests remaining in the hardware. For the motor controller, one flight is equivalent to 8 seconds of acceptance test time (worst case z–axis). The minimum remaining life of 141 minutes equates to 282 flights or 37 acceptance tests remaining in the hardware.

In addition, a development test (on a development article) was performed at Qualification levels for 3 minutes per axis. Additional testing was performed at 1 dB above flight for 1 minute, 2 dB above flight for 1 minute, and 3 dB above flight for 0.38 minute (test abort was not due to failure of unit under test). No broken standoffs, cracked epoxy, or other failure of unit under test was found.

The Motor and Motor Controller component–level post–vibration functional testing should have uncovered electrical defects such as cracked solder joints that were either existing or were introduced by the component–level vibration testing.

The Motor and Motor Controller Acceptance Test Procedures have been updated to reduce the duration of the acceptance vibration test to one minute per axis.

PG2–124:

ITEM:

Motor/Motor Controller Part Number 83–36884–101

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.4, Supplemental Requirements. Functional tests shall be conducted, as a minimum, at the maximum operating temperature plus 20 degrees F (11.1 degrees C) and at the minimum operating temperature minus 20 degrees F (11.1 degrees C) during the first and last operating cycle after the dwell and after return of the component to ambient temperature.

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.4, Supplemental Requirements. Functional tests shall be conducted, as a minimum, at the maximum predicted temperature plus 20 degrees F (11.1 degrees C) and at the minimum predicted temperature minus 20 degrees F (11.1 degrees C) during the first and last thermal cycles and after return of the component to ambient.

EXCEPTION:

During Qualification Thermal Vacuum and Thermal Cycle tests, the EVA Drive of the Motor was not functioned after the dwell at the maximum and minimum predicted operating temperatures with 20 degrees F margin during the first and last operating cycles. The EVA drive was only functioned at ambient temperature during qualification thermal vacuum and thermal cycle testing.

RATIONALE:

The Motor and Motor Controller has EVA redundancy for deployment. The EVA drive was successfully tested during the Plum Brook Thermal Vacuum testing of PVR–1. It was used to deploy and retract the radiator when the environmental temperature was ambient, –100 degrees F, and 120 degrees F. When the environment was –100 degrees F, the EVA drive temperature was controlled with heaters to approximately 0 degrees F. During Acceptance testing of the four Photovoltaic Radiator ORUs, the EVA drives were successfully tested at ambient temperature.

Motor and Motor Controller Run–in testing (referred to as 5th Stage testing) of the worm gear set is performed at Tecstar in a flight–like assembly (parts fabricated to flight requirements but designated “run–in” fixtures). Parts not flight equivalent are the first 4 stages of the 5th stage gear–head which is replaced with STE. Four planet gears are replaced with the WT61770 STE, which interfaces to the 5th stage ring gear and transmits torque to the output shaft. The planets are replaced in the run–in test so that all the gear torque data and calculated efficiencies are for the worm shaft to gear alone (do not include the planetary to ring gear mesh).

During the run–in test setup, alignment of the worm shaft and the worm/ring gear is established with shims. Visual inspection of the no–load run–in wear pattern in the dry film lube applied to the worm wheel teeth is performed. The shimming information is recorded and the shims remain with the match–set worm shaft and worm wheel. During final assembly, the flight worm housing interface dimensions are compared to the run–in housing dimensions, and shim adjustments are made if necessary. No shim adjustments have been made to date.

The worm gear is run–in (5th Stage level) and tested at ambient temperature and at temperature extremes. Once the Motor and Motor Controller is assembled, the worm gear efficiency is tested at the ambient temperature in the Acceptance Test Procedure. The efficiency calculation includes an engineering estimate for the efficiency of the now functional 5th stage planet gear pass. If the efficiencies are essentially the same, the temperature extreme efficiency data is transferred to the Motor and Motor Controller Acceptance Test Procedure data sheets. During Photovoltaic Radiator level testing at Plumbrook, the Motor and Motor Controller operational temperatures were from 0 degrees F to 120 degrees F. Relatively benign temperatures would have relatively little thermal affect on EVA functionality. Although the 5th Stage testing does not function the planet gears at temperature extremes, the full–up motor testing at temperature extremes does. The 5th Stage thermal testing primarily concentrates on the worm shaft to worm gear interface. Therefore, the entire EVA drive string is functioned at temperature with both the full–up Motor testing and the 5th Stage testing at temperature extremes.

PG2–125:

ITEM:

Motor/Motor Controller Part Number 83–36884–101

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.4, Supplemental Requirements. Functional tests shall be conducted at the maximum and minimum predicted temperature levels during the first and last operating cycles after the dwell and after return of the component to ambient temperature.

Paragraph 5.1.3, Thermal Cycle Test, Component Qualification.
Paragraph 5.1.3.4, Supplemental Requirements. Functional tests shall be conducted during the first and last operating thermal cycles after the dwell at the maximum and minimum predicted operating temperatures and after return of the component to ambient.

EXCEPTION:

During Acceptance Thermal Vacuum and Thermal Cycle tests, the EVA Drive of the Motor was not functioned after the dwell at the maximum and minimum predicted operating temperatures during the first and last operating cycles. The EVA drive was only functioned at ambient temperature during acceptance thermal vacuum and thermal cycle testing.

RATIONALE:

The Motor and Motor Controller has EVA redundancy for deployment. The EVA drive was successfully tested during the Plum Brook Thermal Vacuum testing of PVR–1. It was used to deploy and retract the radiator when the environmental temperature was ambient, –100 degrees F, and 120 degrees F. When the environment was –100 degrees F, the EVA drive temperature was controlled with heaters to approximately 0 degrees F. During Acceptance testing of the four Photovoltaic Radiator ORUs, the EVA drives were successfully tested at ambient temperature.

Motor and Motor Controller Run–in testing (referred to as 5th Stage testing) of the worm gear set is performed at Tecstar in a flight–like assembly (parts fabricated to flight requirements but designated “run–in” fixtures). Parts not flight equivalent are the first 4 stages of the 5th stage gear–head which is replaced with STE. Four planet gears are replaced with the WT61770 STE, which interfaces to the 5th stage ring gear and transmits torque to the output shaft. The planets are replaced in the run–in test so that all the gear torque data and calculated efficiencies are for the worm shaft to gear alone (do not include the planetary to ring gear mesh).

During the run–in test setup, alignment of the worm shaft and the worm/ring gear is established with shims. Visual inspection of the no–load run–in wear pattern in the dry film lube applied to the worm wheel teeth is performed. The shimming information is recorded and the shims remain with the match–set worm shaft and worm wheel. During final assembly, the flight worm housing interface dimensions are compared to the run–in housing dimensions, and shim adjustments are made if necessary. No shim adjustments have been made to date.

The worm gear is run–in (5th Stage level) and tested at ambient temperature and at temperature extremes. Once the Motor and Motor Controller is assembled, the worm gear efficiency is tested at the ambient temperature in the Acceptance Test Procedure. The efficiency calculation includes an engineering estimate for the efficiency of the now functional 5th stage planet gear pass. If the efficiencies are essentially the same, the temperature extreme efficiency data is transferred to the Motor and Motor Controller Acceptance Test Procedure data sheets. During Photovoltaic Radiator level testing at Plumbrook, the Motor and Motor Controller operational temperatures were from 0 degrees F to 120 degrees F. Relatively benign temperatures would have relatively little thermal affect on EVA functionality. Although the 5th Stage testing does not function the planet gears at temperature extremes, the full–up motor testing at temperature extremes does. The 5th Stage thermal testing primarily concentrates on the worm shaft to worm gear interface. Therefore, the entire EVA drive string is functioned at temperature with both the full–up Motor testing and the 5th Stage testing at temperature extremes.

PG2–126:

ITEM:

Heater Controller Assembly Part Number 83–48368–101

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Heater Controller Assembly will not undergo a Qualification Thermal Vacuum Test.

RATIONALE:

Thermal Vacuum testing in accordance with SSP 41172 Qualification Thermal Vacuum Test requirements was performed on the Engineering Model Heater Controller Assembly.

The Engineering Model Heater Controller Assembly is the same as a flight fidelity Heater Controller Assembly except for the addition of electrical splices for reading internal voltages and a hole in the cover to accommodate routing of test instrumentation wires. The added wiring and associated monitoring circuitry have an insignificant effect on the electrical/electronic performance of the Heater Controller Assembly. The opening for the additional wires should not affect the thermal response of the Heater Controller Assembly other than to slightly change the thermal response rate of the electronics. The opening would tend to increase the response (ramp) rate when the vacuum temperature is changing, which is more stressful. At the hot and cold soak temperatures, the opening would have no significant effect since it is small and basically plugged by the wiring passing through it.

Addition of instrumentation for testing purposes is typical of Space Station hardware testing and has not been an issue.

PG2–127:

ITEM:

Heater Controller Assembly Part Number 83–48368–101

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Photovoltaic Radiator Heater Controller Assembly shall not undergo an Acceptance Thermal Vacuum Test. Future testing will be performed in compliance with SSP 41172 requirements. Heater Controller Assemblies not already installed in Photovoltaic Radiators 1, 2, 3 and 4, any rework of the Heater Controller Assemblies already installed, and any spares shall be procured with a different part number. The new part number shall require the Heater Controller Assemblies to be Acceptance Thermal Vacuum tested.

RATIONALE:

Based on an engineering review of the Heater Controller Assembly, the only feature which requires Thermal Vacuum testing for screening is the heater bonding. The following rationale supports the requested Acceptance Thermal Vacuum Testing exception.

”Process Specification for Bonding with Thermally Conductive Epoxy Adhesive” (Number 508–8–44) specifies surface preparation and bonding of heaters and RTDs onto flat and curved surfaces using epoxy adhesive primer (Hysol EA9205) and epoxy adhesive (TRA–BOND 2151). Epoxy primer is not used for bonding RTDs. The specification requires visual inspection to verify that no bubbles or voids exist in the applied adhesive prior to joining bonding surfaces.

Lockheed Martin Missiles and Fire Control analysis documented in 3–47300/2000DIR–008 shows that an Heater Controller Assembly heater operating in the (conservative) Phase 3 test environment documented in 3–47300/9R–005, "Final Test Report for Photovoltaic Radiator Heater System Thermal–Vacuum Qualification Test", dated March 1999, could experience a debond of 52 percent of the heater surface area and still meet the heating requirements with a 20 percent margin in heater power at the minimum supply voltage.

Also, a heater similar to the PVR Heater Controller Assembly heater was tested by Boeing – Huntington Beach in a vacuum with 40 percent of the heater unbonded without failure. The sum of the differences between a Heater Controller Assembly heater and Boeing – Huntington Beach heater reflect that an unbonded area of the Heater Controller Assembly heater would not get as hot as the Boeing – Huntington Beach heater tested. The biggest contributing factor in the comparison of the heaters is watt density, and the Heater Controller Assembly heater has less than half of the watt density that the Boeing – Huntington Beach heater as–tested. Therefore, an unbonded area of the Heater Controller Assembly heater should not be as hot as the Boeing – Huntington Beach heater tested and the Heater Controller Assembly heater should be able to survive an unbonding case in which at least 40 percent of the heater was unbonded, based on Boeing – Huntington Beach heater testing.

Additionally, PVR heat rejection capability is not compromised following a complete Heater Controller Assembly heater failure, since it does not cause the manifolds in either PVR ammonia loop to freeze. If one PVR ammonia loop is not flowing (failed), the heat rejection capability is still intact. Heat rejection capability is not lost unless both ammonia loops are non–flowing (two failures).

PVR deploy/retraction capability with the Motor/Motor Controller may be jeopardized since loss of the HCA heater could result in loss of the HCA, which is used to control the Motor/Motor Controller heaters. Very conservative analysis shows the Motor/Motor Controller electronics could reach temperatures below –100 degrees F following complete loss of the Motor/Motor Controller heaters. The EVA drive may function at the cold temperatures and, if not, should regain functionality when the environment warms.

PG2–128:

ITEMS:

Short Spacer P5 Part Number R081200
Short Spacer S5 Part Number R081200

SSP 41172 REQUIREMENT:

Paragraph 4.4, Flight Element Qualification Test. "The flight element qualification test baseline consists of all the required tests specified in Table 4–2." Table 4–2 identifies a Static Structural Load test as one of the required tests.

EXCEPTION:

A Qualification Static Structural Load Test will not be performed on the Short Spacer P5 and the Short Spacer S5.

RATIONALE:

The Short Spacers P5 and S5 are truss structures similar in design concept to the P6 Long Spacer structure. It has no attached ORUs, only a 230–pound Grapple Fixture. It must solely carry its weight. It has a robust structural capability since it was designed with a factor of safety of 2.0 on ultimate. The loads used in the design are conservative as indicated by using a misalignment load of 8010 pounds, or about 1/3 of the total trunnion load. Recent load analysis shows that coupling a Short Spacer with the Orbiter model reduces the misalignment loads to 4800 pounds. In addition, margins quoted against peak bending stresses where a modulus of rupture approach (not used in current analysis) would show additional strength capability.

The lessons learned from P6 Long Spacer static test correlation were applied to the Short Spacers. Eccentricities in the truss joints were eliminated or reduced in the design to improve load paths. High fidelity local models of the truss joints were used to better define the stiffness of the joints, which was a weak area in the P6 Long Spacer correlation. An alternative review was done using a nominal, 1.4–factor of safety and the worst P6 Long Spacer correlation factors. This review resulted in positive margins, providing a comparable check to P6.

A P5 Modal test will be conducted. The finite element model is set up to allow easy validation of joint stiffness values. The modal test will provide an assurance of the overall health of the finite element model. Sensitivity studies show that the on–orbit loads are not sensitive to P5 stiffness. The stiffness of the P5 Short Spacer segment was doubled and halved with no significant impact to the on–orbit frequency response. The frequency response was within the tolerance specified for on–orbit model validation (+/– 10 percent) with appendage (i.e., on–orbit critical) frequency changes less than 1 percent and truss frequency changes less than 4 percent.

PG2–129:

ITEMS:

Short Spacer P5 Part Number R081200
Short Spacer S5 Part Number R081200

SSP 41172 REQUIREMENT:

Paragraph 4.4, Flight Element Qualification Test. “The flight element qualification test baseline consists of all the required tests specified in Table 4–2.” Table 4–2 identifies an Acoustic Vibration test as one of the required tests.

EXCEPTION:

A Qualification Acoustic Vibration Test will not be performed on the Short Spacer P5 and the Short Spacer S5.

RATIONALE:

The Short Spacers P5 and S5 are truss structures similar in design concept to the P6 Long Spacer structure. It has no attached ORUs, only a 230–pound Grapple Fixture. The P6 Long Spacer was acoustically tested as part of the P6 Cargo Element. Whereas the P6 Long Spacer Structure includes two Photovoltaic Radiators and two Pump Flow Control System units, the Short Spacers do not. These items on the P6 Long Spacer add considerable surface area to interact with the acoustic field. Any significant levels measured on the P6 Long Spacer were attributed to the mechanical energy transfer for the attached radiators. Otherwise, the levels were very low. Since the Short Spacers P5 and S5 do not have this hardware, the Short Spacers’ surface area to volume ratio is considerably less than the P6 Long Spacer’s surface area to volume ratio. This essentially renders the Short Spacers acoustically transparent.

Structural analysis is performed on sets of loads combined in accordance with SSP 30559. The required methodology has the acoustic loads Root Sum Squared with the remainder of the loads. This makes the effect of the small acoustic loads negligible on the total loads.

Therefore, the conduct of an acoustic test is unnecessary.

PG2–130:

ITEM:

Solar Array Wing Mast Canister Tip/Pivot Assembly Part Number 5838844

SSP 41172 REQUIREMENT:

Paragraph 6.1.1B, Assembly/Components Protoflight Tests. For the thermal cycling test, the temperature extremes shall be 10 degrees F (5.6 degrees C) beyond the maximum and minimum predicted temperatures.

EXCEPTION:

The maximum temperature during protoflight thermal cycle testing under non–operating conditions for the Solar Array Wing Mast Canister Tip/Pivot Assembly shall be 158 degrees F.

RATIONALE:

The maximum temperature predicted under non–operating conditions for the Solar Array Wing Mast Canister Tip/Pivot Assembly is 151 degrees F. Thus, during the protoflight thermal cycle testing under non–operating conditions, the Tip/Pivot Assembly was exposed to 7 degrees F beyond the maximum predicted temperature.

The maximum non–operating condition is not a hardware performance condition. Therefore, the only issue is the design integrity of the hardware to survive exposure to non–operating temperatures. In all interfaces where dissimilar materials were used, the clearances are sufficient to provide for material thermal expansion and contraction. The maximum predicted operating condition of the hardware was surpassed during testing by 55 degrees F and the hardware successfully functioned. Thus, missing the required maximum non–operating temperature during protoflight thermal cycle testing by 3 degrees F is low risk.

PG2–131:

ITEM:

Solar Array Wing Pivot Fitting Part Number 5835853

SSP 41172 REQUIREMENT:

Paragraph 6.1.1B, Assembly/Components Protoflight Tests. For the thermal cycling test, the temperature extremes shall be 10 degrees F (5.6 degrees C) beyond the maximum and minimum predicted temperatures.

EXCEPTION:

The maximum temperature during protoflight thermal cycle testing under non–operating conditions for the Solar Array Wing Mast Canister Pivot Fitting shall be 166 degrees F.

RATIONALE:

The maximum temperature predicted under non–operating conditions for the Solar Array Wing Pivot Fitting is 163 degrees F. Thus, during the protoflight thermal cycle testing under non–operating conditions, the Pivot Fitting was exposed to 3 degrees F beyond the maximum predicted temperature.

The maximum non–operating condition is not a hardware performance condition. Therefore, the only issue is the design integrity of the hardware to survive exposure to non–operating temperatures. In all interfaces where dissimilar materials were used, the clearances are sufficient to provide for material thermal expansion and contraction. The maximum predicted operating condition of the hardware was surpassed during testing by 15 degrees F and the hardware successfully functioned. Thus, missing the required maximum non–operating temperature during protoflight thermal cycle testing by 7 degrees F is low risk.

PG2–132:

ITEM:

Solar Array Wing Locking Strut Part Number 5851331

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The maximum temperature during qualification thermal cycle testing under non–operating conditions for the Solar Array Wing Mast Canister Locking Strut shall be 212 degrees F.

RATIONALE:

The maximum worst–case on–orbit temperature under non–operating conditions predicted for the Solar Array Wing Locking Strut is 218 degrees F. Thus, during the qualification thermal cycle testing under non–operating conditions, the Locking Strut was exposed to 6 degrees F below the maximum predicted temperature.

The hot non–operating condition is not a hardware performance condition. Therefore, the only issue is the design integrity of the hardware to survive exposure to non–operating temperatures. In all interfaces where dissimilar materials were used, the clearances are sufficient to provide for material thermal expansion and contraction. The maximum predicted operating condition of the hardware was surpassed during testing by 29 degrees F and the hardware successfully functioned. Thus, missing the required maximum worst–case on–orbit temperature under non–operating conditions during qualification thermal cycle testing by 26 degrees F is low risk.

PG2–133:

ITEM:

Solar Array Wing Tension Mechanism Part Number 5836966

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The maximum temperature during qualification thermal cycle testing under non–operating conditions for the Solar Array Wing Mast Canister Tension Mechanism shall be 240 degrees F.

The maximum temperature during qualification thermal cycle testing under operating conditions for the Solar Array Wing Mast Canister Tension Mechanism shall be 176 degrees F.

The minimum temperature during qualification thermal cycle testing under non–operating conditions for the Solar Array Wing Mast Canister Tension Mechanism shall be – 118 degrees F.

RATIONALE:

The maximum worst–case on–orbit temperature under non–operating conditions predicted for the Solar Array Wing Tension Mechanism is 229 degrees F. The maximum worst–case on–orbit temperature under operating conditions predicted for the Solar Array Wing Tension Mechanism is 194 degrees F. Thus, during the qualification thermal cycle testing under non–operating conditions, the Tension Mechanism was exposed to 11 degrees F above the maximum predicted temperature and during the qualification thermal cycle testing under operating conditions, the Tension Mechanism was exposed to 16 degrees F below the maximum predicted operating temperature (36 degrees F below the required maximum qualification thermal cycling operating temperature).

Additionally, the minimum worst–case on–orbit temperature under non–operating conditions predicted for the Solar Array Wing Tension Mechanism is – 102 degrees F. Thus, during the qualification thermal cycle testing under non–operating conditions, the Tension Mechanism was exposed to 16 degrees F below the minimum predicted non–operating temperature.

The cold and hot non–operating conditions are not hardware performance conditions. Therefore, the only issue is the design integrity of the hardware to survive exposure to non–operating temperatures. In all interfaces where dissimilar materials were used, the clearances are sufficient to provide for material thermal expansion and contraction. The cold operating condition of the hardware was surpassed during testing by 6 degrees F and the hardware successfully functioned. Thus, missing the required minimum–operating worst–case on–orbit temperature during qualification thermal cycle testing by 4 degrees F is low risk.

A tolerance stackup analysis EID–05787 was performed for every interface of the Solar Array Wing Latch Mechanism, which includes the tension mechanism, for the effects of maximum predicted delta T from ambient to cold. After a review of the design, NASA/Boeing/Lockheed Martin determined that in all interfaces where dissimilar materials were used, the clearances increased as temperatures increased. Thus, missing the required maximum worst–case on–orbit temperature during non–operating conditions by 9 degrees F and the required maximum worst–case on–orbit temperature under operating conditions by 38 degrees F during qualification thermal cycle testing is low risk.

PG2–134:

ITEM:

Solar Array Wing Guidewire Mechanism Part Number 5851246

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.

Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The maximum temperature during qualification thermal cycle testing under non–operating conditions for the Solar Array Wing Mast Canister Guidewire Mechanism shall be 256 degrees F.

The minimum temperature during qualification thermal cycle testing under non–operating conditions for the Solar Array Wing Mast Canister Guidewire Mechanism shall be – 130 degrees F.

RATIONALE:

The maximum worst–case on–orbit temperature under non–operating conditions predicted for the Solar Array Wing Guidewire Mechanism is 260 degrees F. Thus, during the qualification thermal cycle testing under non–operating conditions, the Guidewire Mechanism was exposed to 4 degrees F below the maximum predicted non–operating temperature (24 degrees F below the required maximum non–operating qualification thermal cycling temperature).

Additionally, the minimum worst–case on–orbit temperature under non–operating conditions predicted for the Solar Array Wing Guidewire Mechanism is – 119 degrees F. Thus, during the qualification thermal cycle testing under non–operating conditions, the Guidewire Mechanism was exposed to 11 degrees F below the minimum predicted non–operating temperature.

The cold and hot non–operating conditions are not hardware performance conditions. Therefore, the only issue is the design integrity of the hardware to survive exposure to non–operating temperatures. In all interfaces where dissimilar materials were used, the clearances are sufficient to provide for material thermal expansion and contraction. The minimum operating condition of the hardware was surpassed during testing by 3 degrees F (including the 20–degree F qualification margin) and the maximum operating condition was surpassed by 16 degrees F (including the 20–degree F qualification margin). In each case, the hardware successfully functioned. Thus, missing the required minimum worst–case temperature under non–operating conditions by 9 degrees F and the required maximum worst–case temperature under non–operating conditions by 24 degrees F is low risk.

PG2–135:

ITEM:

Solar Array Wing Latch Hook Assembly Part Number 5851286

SSP 41172 REQUIREMENT:

Paragraph 6.1.1B, Assembly/Components Protoflight Tests. For the thermal cycling test, the temperature extremes shall be 10 degrees F (5.6 degrees C) beyond the maximum and minimum predicted temperatures.

EXCEPTION:

The minimum temperature during protoflight thermal cycle testing under non–operating conditions for the Solar Array Wing Latch Hook Assembly shall be – 113 degrees F.

The minimum temperature during protoflight thermal cycle testing under operating conditions for the Solar Array Wing Latch Hook Assembly shall be – 70 degrees F.

RATIONALE:

The minimum worst–case on–orbit temperature under non–operating conditions predicted for the Solar Array Wing Latch Hook Assembly is – 106 degrees F. Thus, during the protoflight thermal cycle testing under non–operating conditions, the Guidewire Mechanism was exposed to 7 degrees F below the minimum predicted non–operating temperature.

Additionally, the minimum worst–case on–orbit temperature under operating conditions predicted for the Solar Array Wing Latch Hook Assembly is – 91 degrees F. Thus, during the protoflight thermal cycle testing under operating conditions, the Guidewire Mechanism was exposed to 21 degrees F above the minimum predicted operating temperature (31 degrees F above the required minimum operating protoflight thermal cycle temperature).

The cold non–operating condition is not a hardware performance condition. The only issue is the design integrity of the hardware to survive exposure to non–operating temperatures. As 7 degrees F margin was obtained during the protoflight thermal cycle testing under non–operating conditions, cognizant NASA, Boeing, and Lockheed Martin engineers accepted as little risk to the program.

The required minimum protoflight thermal cycle temperature under operating conditions was not achieved by 31 degrees F. A tolerance stackup analysis EID–05787 was performed for every interface of the Solar Array Wing Latch Mechanism, which includes the latch hook assembly. The effects of maximum predicted delta T (170 degrees F from ambient to cold) were calculated for every interface. A review of this information by cognizant NASA, Boeing and Lockheed Martin Structures and Mechanisms engineers concluded there was sufficient design margin in the Solar Array Wing Latch Hook Assembly.

Thus, missing the required minimum worst–case temperature under non–operating conditions by 3 degrees F and the required maximum worst–case temperature under operating conditions by 31 degrees F during protoflight thermal cycle testing is deemed an acceptable risk.

PG2–136:

ITEM:

Solar Array Wing Latch Mechanism Part Number 5851286

SSP 41172 REQUIREMENT:

Paragraph 6.1.1B, Assembly/Components Protoflight Tests. For the thermal cycling test, the temperature extremes shall be 10 degrees F (5.6 degrees C) beyond the maximum and minimum predicted temperatures.

EXCEPTION:

The minimum temperature during protoflight thermal cycle testing under operating conditions for the Solar Array Wing Latch Mechanism shall be – 77 degrees F.

RATIONALE:

The minimum worst–case on–orbit temperature under operating conditions predicted for the Solar Array Wing Latch Mechanism is – 73 degrees F. Thus, during the protoflight thermal cycle testing under operating conditions, the Latch Mechanism was exposed to 4 degrees F below the minimum predicted operating temperature.

The required minimum protoflight thermal cycle temperature under operating conditions was not achieved by 6 degrees F. A tolerance stackup analysis EID–05787 was performed for every interface of the Solar Array Wing Latch Mechanism. The effects of maximum predicted delta T (170 degrees F from ambient to cold) were calculated for every interface. A review of this information by cognizant NASA, Boeing and Lockheed Martin Structures and Mechanisms engineers concluded there was sufficient design margin in the Solar Array Wing Latch Mechanism.

PG2–137:

ITEM:

Solar Array Wing Motor Drive Assembly Part Number 5843318

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3B, Test Levels and Duration. The component shall be at the maximum predicted temperature during the hot portion of the cycle and at the minimum predicted temperature during the cold portion of the cycle.

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

The maximum temperature during acceptance thermal vacuum and thermal cycle testing under operating conditions for the Solar Array Wing Motor Drive Assembly shall be 68 degrees F.

RATIONALE:

The maximum predicted temperature under operating conditions for the Solar Array Wing Motor Drive Assembly is 118 degrees F. However, the Motor Drive Assembly has been exposed to non–operating temperatures of 185 degrees F without degradation of performance in follow–on testing. Cognizant NASA, Boeing and Lockheed Martin engineers have concluded that operation at 118 degrees F is within the capability of the existing hardware design.

PG2–138:

ITEM:

Photovoltaic Radiator Grapple Fixture Part Number RH000043

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A qualification thermal vacuum test will not be performed on the Photovoltaic Radiator Grapple Fixture.

RATIONALE:

Fit checks of the Photovoltaic Radiator Grapple Fixture to PVRs 002, 003, and 004 were completed. Fit checks were performed with PVR under preload to simulate worst–case on–orbit thermal environment. The mating of the Photovoltaic Radiator Grapple Fixture to the radiators were evaluated, using actual dimensions, with the Radiator and Grapple Fixture each varying in temperature between –107 degrees F and 150 degrees F. Digital Pre–Assembly measurements for PVRs 001, 002, 003, and 004 were used to develop on–orbit operating envelope for temperature and preload via Thermal/Tolerance Analysis. (EID–05781 Revision B)

Thermal tolerance analysis showed no close fits and/or tight tolerances that would impede operability of the four bearings which articulate over a limited range to facilitate mating and demating of the Photovoltaic Radiator Grapple Fixture. Bearings have been lot–tested for breakaway torque at thermal extremes between –180 degrees F and 235 degrees F. (Reference RE3328, Note 6)

Also, per Source Control Drawing RE3095, the PIP Pins are tested at the component level to the following acceptance requirements and their Acceptance Data Packages on file:

> Each pin met the actuating force requirements when the test article is at the acceptance test temperature of –120 degrees F and 260 degrees F. A functional test verified actuating force requirements at ambient temperature, before and after the test at temperature extremes. For thermal testing, temperature stabilization has been achieved when the rate of change of the test article is no more than 5.4 degrees F per hour. Dwell period, prior to functional testing, was a minimum of 1 hour.

Therefore, thermal vacuum testing is not required for the Photovoltaic Radiator Grapple Fixture.

PG2–139:

ITEM:

Photovoltaic Radiator Grapple Fixture Part Number RH000043

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Acceptance.
Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A qualification random vibration test will not be performed on the Photovoltaic Radiator Grapple Fixture.

RATIONALE:

The P6 Long Spacer was acoustically tested as part of the P6 Cargo Element. Whereas the P6 Long Spacer Structure includes numerous ORUs, including three PVRs, the Short Spacer does not. These items on the P6 Long Spacer add considerable surface area to interact with the acoustic field. Any significant levels measured on the P6 Long Spacer were attributed to the mechanical energy transfer from the attached items. Despite these effects, the energy transfer levels were very low. Therefore, the P6 Cargo Element is not a source of random vibration loads.

Based on the above and that the P5 and S5 Short Spacers are similar in design to the P6 Long Spacer structure (with no ORUs and little surface area to interact with the acoustic field), the Short Spacer would not be a significant source of random vibration loads.

The Photovoltaic Radiator Grapple Fixture Assembly, similar in design to the Short Spacer, with relatively small surface area compared to the PVR and the P5 and S5 Short Spacers, will see insignificant random vibration loads. Also, during launch the Photovoltaic Radiator Grapple Fixture is restrained by EVA Bolts at the four attached points on P5 and S5.

In addition, per note 7 in Table 5–1, "Only components with close tolerances requiring precise adjustment, or that cannot be inspected effectively, require random tests." Precise adjustment is not required for this hardware and all components are accessible for inspection to screen out any workmanship defects. In addition, inspection will provide the verification that the mechanisms are assembled and properly designed.

Based on the above, no random vibration test for the Photovoltaic Radiator Grapple Fixture is required.

PG2–140:

ITEM:

Photovoltaic Radiator Grapple Fixture Part Number RH000043

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An acceptance random vibration test will not be performed on the Photovoltaic Radiator Grapple Fixture.

RATIONALE:

The P6 Long Spacer was acoustically tested as part of the P6 Cargo Element. Whereas the P6 Long Spacer Structure includes numerous ORUs, including three PVRs, the Short Spacer does not. These items on the P6 Long Spacer add considerable surface area to interact with the acoustic field. Any significant levels measured on the P6 Long Spacer were attributed to the mechanical energy transfer from the attached items. Despite these effects, the energy transfer levels were very low. Therefore, the P6 Cargo Element is not a source of random vibration loads.

Based on the above and that the P5 and S5 Short Spacers are similar in design to the P6 Long Spacer structure (with no ORUs and little surface area to interact with the acoustic field), the Short Spacer would not be a significant source of random vibration loads.

The Photovoltaic Radiator Grapple Fixture Assembly, similar in design to the Short Spacer, with relatively small surface area compared to the PVR and the P5 and S5 Short Spacers, will see insignificant random vibration loads. Also, during launch the Photovoltaic Radiator Grapple Fixture is restrained by EVA Bolts at the four attached points on P5 and S5.

In addition, per note 7 in Table 5–1, “Only components with close tolerances requiring precise adjustment, or that cannot be inspected effectively, require random tests.” Precise adjustment is not required for this hardware and all components are accessible for inspection to screen out any workmanship defects. In addition, inspection will provide the verification that the mechanisms are assembled and properly designed.

Based on the above, no random vibration test for the Photovoltaic Radiator Grapple Fixture is required.

PG2–141:

ITEM:

Photovoltaic Radiator Grapple Fixture Part Number RH000043

SSP 41172 REQUIREMENT:

Paragraph 4.2.7, Pyrotechnic Shock Test, Component Qualification.
Paragraph 4.2.7.3, Test Levels and Exposure. ALL REQUIREMENTS.
Paragraph 4.2.7.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A qualification pyrotechnic shock test will not be performed on the Photovoltaic Radiator Grapple Fixture.

RATIONALE:

A pyrotechnic shock qualification test in accordance with SSP 41172, Table 4–1 is not required for this component. The accepted exception criteria, that a minimum of 10 feet of structure and three structural joints separates the shock origin and the assembly in question, has been verified for the structural components located on the P5/S5. There are more than seven major structural joints spanning more than 46 feet between the radiator cinch ordinance (NASA Standard initiator), installed on ITS S1 and P1 per 1F80001 and the S3/S4 Alpha Joint Interface Structure and the P3/P4 Alpha Joint Interface Structure. The same rationale holds for the ITS P5 and ITS S5, since they are further outboard of the Alpha Joint Interface Structure.

PG2–142:

ITEM:

Photovoltaic Radiator Grapple Fixture Part Number RH000043

SSP 41172 REQUIREMENT:

Paragraph 4.3.1, Static Structural Test, Component Qualification.
Paragraph 4.3.1.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.3.1.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A qualification static structural load test will not be performed on the Photovoltaic Radiator Grapple Fixture.

RATIONALE:

A modal run performed for each of the two lift point attachment configurations on the Photovoltaic Radiator Grapple Fixture /Short Spacer configuration showed that the required minimum frequency of 0.2 Hz was met as indicated for the first five modal frequencies:

Lift Point 1	Lift Point 2
0.44 Hz	0.43 Hz
1.72 Hz	1.69 Hz
2.50 Hz	2.44 Hz
7.24 Hz	7.43 Hz
7.66 Hz	7.48 Hz

Structural analysis utilizing the following Ultimate Factor of Safety (2.0) and Yield Factor of Safety (1.25) yields the following safety margins:

Margin of Safety for Launch Loads Table:

Part Number	Part Name	Yield Margin of Safety	Ultimate Margin of Safety
RH000045	Box Brace Beam	0.76	0.46
RH000047	Cross–Beam Brace	NA	0.09
RH000048	Support Arm	0.28	0.08
RH000049	Center Support	0.28	0.08
RH000051	Adapter Plate	100	83
RE3328	Spherical Bearing	NA	1.36

Margin of Safety for SEE and Kick* Loads Table:

Part Number	Part Name	Yield Margin of Safety	Ultimate Margin of Safety
RH000045	Box Brace Beam	5.00	3.97

RH000047	Cross–Beam Brace	NA	6.19
RH000048	Support Arm	0.49	0.26
RH000049	Center Support	0.49	0.26
RH000051	Adapter Plate	0.73	0.46
RH000145*	Bracket, Storage	0.40	0.06
RE3095*	Pin, Locking	NA	16.97
RE3328	Spherical Bearing	NA	6.16

Therefore, Static Structural tests are not required for the Photovoltaic Radiator Grapple Fixture.

PG2–143:

ITEM:

Mast Canister Assembly Part Number 5818235

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3A, Test Levels and Duration. The pressure shall be reduced from atmospheric to below 0.0001 Torr (0.0133 Pa).

EXCEPTION:

The Solar Array Wing Mast Canister Assembly experienced ambient pressure thermal testing in lieu of vacuum during the Qualification Thermal Vacuum program.

RATIONALE:

The only vacuum sensitive material is the Mast Roller Grease (Braycote 601) which was tested during the Mast Roller Thermal Vacuum Test (AEC95518R904). The only electronic components are EEE parts in the Motor Drive Assembly. The Motor Drive Assembly was qualified separately through a component–level thermal vacuum test. In addition, the Flight Wing 1 and 2 mast canisters were tested under thermal vacuum conditions during wing–level thermal vacuum tests. Thus, the risk associated with not performing thermal vacuum testing at the Solar Array Wing Mast Canister Assembly is low.

PG2–144:

ITEM:

Mast Canister Assembly Part Number 5818235

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3A, Test Levels and Duration. The pressure shall be reduced from atmospheric to below 0.0001 Torr (0.0133 Pa).

EXCEPTION:

The Solar Array Wing Mast Canister Assembly experienced ambient pressure thermal testing in lieu of vacuum during the Acceptance Thermal Vacuum program.

RATIONALE:

The only vacuum sensitive material is the Mast Roller Grease (Braycote 601) which was tested during the Mast Roller Thermal Vacuum Test (AEC95518R904). The only electronic components are EEE parts in the Motor Drive Assembly. The Motor Drive Assembly was screened for workmanship separately through a component–level thermal vacuum test. In addition, the Flight Wing 1 and 2 mast canisters were tested under thermal vacuum conditions during wing–level thermal vacuum tests. Thus, the risk associated with not performing thermal vacuum testing at the Solar Array Wing Mast Canister Assembly is low.

PG2–145:

ITEM:

Mast Canister Assembly Part Number 5818235

SSP 41172 REQUIREMENT:

Paragraph 6.1.1 Assembly/Components Protoflight Tests.
Paragraph 6.1.1B. For the thermal cycling test, the temperature cycles shall be conducted at 10 degrees F (5.6 degrees C) beyond the maximum and minimum predicted temperatures.

EXCEPTION:

During Protoflight Flight Wing testing on Solar Arrays 1 and 2, the Solar Array Wing Mast Canister Assembly experienced a minimum temperature of – 113 degrees F and a maximum temperature of 104 degrees F during deploy operations and 195 degrees F during retract operations.

RATIONALE:

As discussed in PG2–43, delivered mast canisters were scheduled to be subjected to protoflight temperature extremes (maximum and minimum predicted +/– 10 degrees F margin) during the wing level protoflight tests on Solar Arrays 1 and 2. This is to ensure adequate workmanship screening to anticipated Space Station Revision D+ predicted temperatures of 193 +/–5 degrees F (non–op hot) and –127 +/–5 degrees F (non–op cold). As indicated, the temperatures experienced by the Mast Canister Assembly during this protoflight flight wing testing were in violation of the protoflight thermal cycle test requirement.

The non–operational conditions are not hardware performance conditions. Therefore, the only issue is the design integrity of the hardware to survive exposure to non–operational temperatures. Cognizant Boeing and NASA engineers agreed that the test temperatures experienced during the wing–level testing did not unduly impact the quality of the design integrity screening of the Mast Canister Assembly. Additionally, the Revision D+ predicted operating conditions of 94 degrees F (hot) and – 95 degrees F (cold) was reached during thermal testing and the hardware successfully functioned.

PG2–146:

ITEM:

Motor Drive Assembly Part Number 5843318

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.2, Test Description. With the component operating (power on) and while parameters are being monitored, the chamber temperature shall be reduced to bring the component to the specified low qualification temperature level as measured at a representative location on the component such as the mounting point on the baseplate for conduction–dominated internal designs or at a representative location(s) on the case for radiation–controlled designs.

EXCEPTION:

MDA was not directly instrumented during the qualification thermal cycling tests. Instrumentation was located on the test fixture.

RATIONALE:

The MDA is a fairly compact assembly (6.375 inches x 10.5 inches x 7.25 inches) weighing less than 16 pounds. The thermal conduction path from the mechanical assembly baseplate (which was instrumented during the test) to the base of the electrical assembly is an all–aluminum path of 4 inches. Therefore, any lag between the mechanical assembly baseplate and the electrical assembly baseplate will be minimal and does not invalidate the thermal cycle testing performed.

PG2–147:

ITEM:

Motor Drive Assembly Part Number 5843318

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.2, Test Description. With the component operating (power on) and while parameters are being monitored, the chamber temperature shall be reduced to bring the component to the specified low acceptance temperature level as measured at a representative location on the component such as the mounting point on the baseplate for conduction–dominated internal designs or at a representative location(s) on the case for radiation–controlled designs.

EXCEPTION:

MDA was not directly instrumented during the acceptance thermal cycling tests. Instrumentation was located on the test fixture.

RATIONALE:

The MDA is a fairly compact assembly (6.375 inches x 10.5 inches x 7.25 inches) weighing less than 16 pounds. The thermal conduction path from the mechanical assembly baseplate (which was instrumented during the test) to the base of the electrical assembly is an all–aluminum path of 4 inches. Therefore, any lag between the mechanical assembly baseplate and the electrical assembly baseplate will be minimal and does not invalidate the thermal cycle testing performed.

PG2–148:

ITEM:

Motor Drive Assembly Part Number 5843318

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3C, Test Levels and Duration. The dwell period shall be long enough for the component to reach internal thermal equilibrium for not less than 1 hour.

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3C, Test Levels and Duration. Each cycle shall have a 1 hour minimum dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on.

EXCEPTION:

The minimum dwell periods during qualification thermal vacuum and thermal cycle testing prior to functional testing at the minimum and maximum operating conditions for the Solar Array Wing Motor Drive Assembly were 30 minutes.

RATIONALE:

In all cases, prior to performing hot and cold functional tests, the MDA was coming from a more severe non–operational temperature condition (i.e., prior to performing functional tests at a maximum temperature of +86 degrees F during qualification, the MDA is approaching from a maximum nonoperational temperature of +185 degrees F; for functional tests at a minimum temperature of – 49 degrees F during qualification, the MDA is approaching from a minimum nonoperational condition of – 67 degrees F). Since the dwell period is intended to insure that the unit has reached a state of internal thermal equilibrium at the required temperature condition, the MDA would have been at a more extreme temperature condition than required (i.e., either hotter or colder than required) if 30 minutes is insufficient for it to reach internal thermal equilibrium. Therefore, the measured 30–minute minimum dwell periods during MDA thermal testing are of minimal risk.

PG2–149:

ITEM:

Motor Drive Assembly Part Number 5843318

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3C, Test Levels and Duration. The component shall undergo a dwell period of at least one hour or a time sufficient for the component to reach internal thermal equilibrium as established by qualification testing, whichever is greater, at both the high and low temperature extremes with power off and then turned on.

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.3C, Test Levels and Duration. Each operating cycle shall have a 1 hour minimum dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on.

EXCEPTION:

The minimum dwell periods during acceptance thermal vacuum and thermal cycle testing prior to functional testing at the minimum and maximum operating conditions for the Solar Array Wing Motor Drive Assembly were 30 minutes.

RATIONALE:

In all cases, prior to performing hot and cold functional tests, the MDA was coming from a more severe non–operational temperature condition (i.e., prior to performing functional tests at a maximum temperature of +68 degrees F during acceptance, the MDA is approaching from a maximum nonoperational temperature of +167 degrees F; for functional tests at a minimum temperature of – 49 degrees F during acceptance, the MDA is approaching from a minimum nonoperational condition of – 67 degrees F). Since the dwell period is intended to insure that the unit has reached a state of internal thermal equilibrium at the required temperature condition, the MDA would have been at a more extreme temperature condition than required (i.e., either hotter or colder than required) if 30 minutes is insufficient for it to reach internal thermal equilibrium. Therefore, the measured 30–minute minimum dwell periods during MDA thermal testing are of minimal risk.

PG2–150:

ITEM:

Integrated Truss Segment Z1 Truss Integrated Assembly Configuration Item R074480
Photovoltaic Module P4 Integrated Equipment Assembly Configuration Item R083499
Photovoltaic Module S4 Integrated Equipment Assembly Configuration Item R082499
Photovoltaic Module P6 Integrated Equipment Assembly Configuration Item R075300
Photovoltaic Module S6 Integrated Equipment Assembly Configuration Item R083300
Early Ammonia Servicer Configuration Item RH000186

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 4.4.4, Pressure/Leak Test, Flight Element Qualification.
Paragraph 4.4.4.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Tube welds performed on the Configuration Items indicated and spare Photovoltaic Radiators and its components manifested as replacements on the indicated Configuration Items will not be required to show compliance to a SSP 41172 Qualification Leak Test methodology.

RATIONALE:

The tubing welds completed on ITS Z1, PVM P4, PVM S4, PVM P6, PVM S6 and the Early Ammonia Servicer, and the welds on the Photovoltaic Radiators and Photovoltaic Radiator spare parts tubing are ”good” welds since the weld process prevents the acceptance of poor quality welds. The process, which was developed and successfully implemented over many years includes:

Designing the Tubing to be inspectable;
Procuring the tubing from certified vendors;
Inspecting the raw tubing stock (via X–Ray (Boeing–Canoga Park), Eddy current (Lockheed Martin Missiles and Fire Control), and Ultrasound Testing (Hamilton Sunstrand));
Establishing the weld process pedigree and documenting it through weld schedules;
Training welders and inspectors;
Inspecting welds with the etch & dye penetrant method;
X–raying the welds (multiple views, and real–time x–rays performed on non–vendor Boeing–Canoga Park Thermal & Environmental Control Systems tube welds); and
Proof testing and burst testing for qualification hardware.

In addition to the above process, welds on Boeing–Canoga Park Thermal & Environmental Control Systems hardware are exposed to pressurized ammonia at KSC during testing and final fill. Area ammonia detectors will detect leaks which are sufficient to cause an ammonia concentration of approximately 1 to 2 parts per million. Evidence of "good" welds on Boeing–Canoga Park Thermal & Environmental Control Systems tubing is the successful operation of PVM P6 Integrated Equipment Assembly and Long Spacer ammonia coolant systems for approximately 1 year with no detectable loss of ammonia. This hardware includes over 1000 tube welds.

Conservative calculations show that the leak rate per weld could be as high as 1E–04 scc per second Helium without resulting in a violation of life requirements due to a low ammonia inventory at End of Life. Also, this assumes all of the welds are leaking at a rate of 1E–04 scc per second Helium which is extremely improbable since there are over 300 welds on each Integrated Equipment Assembly element on PVM P4, PVM S4, and PVM S6 with the radiators installed and "good" welds are as leak tight as tubing. Although the probe detection method used was not calibrated to prove it could detect point leakages of 1E–05 sccs Helium, it should have that capability.

PG2–151:

ITEM:

Integrated Truss Segment Z1 Truss Integrated Assembly Configuration Item R074480
Photovoltaic Module P4 Integrated Equipment Assembly Configuration Item R083499
Photovoltaic Module S4 Integrated Equipment Assembly Configuration Item R082499
Photovoltaic Module P6 Integrated Equipment Assembly Configuration Item R075300
Photovoltaic Module S6 Integrated Equipment Assembly Configuration Item R083300
Early Ammonia Servicer Configuration Item RH000186

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 5.2.3, Pressure/Leak Test, Flight Element Acceptance.
Paragraph 5.2.3.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Tube welds performed on the Configuration Items indicated and spare Photovoltaic Radiator components manifested as replacements on the indicated Configuration Items will not be required to show compliance to a SSP 41172 Acceptance Leak Test methodology. Any reworked welds shall be required to be in compliance with a SSP 41172 Acceptance Leak Test methodology.

RATIONALE:

The tubing welds completed on ITS Z1, PVM P4, PVM S4, PVM P6, PVM S6 and the Early Ammonia Servicer, and the welds on the Photovoltaic Radiators and Photovoltaic Radiator spare parts tubing are "good" welds since the weld process prevents the acceptance of poor quality welds. The process, which was developed and successfully implemented over many years includes:

Designing the Tubing to be inspectable;
Procuring the tubing from certified vendors;
Inspecting the raw tubing stock (via X–Ray (Boeing–Canoga Park), Eddy current (Lockheed Martin Missiles and Fire Control), and Ultrasound Testing (Hamilton Sunstrand));
Establishing the weld process pedigree and documenting it through weld schedules;
Training welders and inspectors;
Inspecting welds with the etch & dye penetrant method; and
X–raying the welds (multiple views, and real–time x–rays performed on non–vendor Boeing–Canoga Park Thermal & Environmental Control Systems tube welds).

In addition to the above process, welds on Boeing–Canoga Park Thermal & Environmental Control Systems hardware are exposed to pressurized ammonia at KSC during testing and final fill. Area ammonia detectors will detect leaks which are sufficient to cause an ammonia concentration of approximately 1 to 2 parts per million. Evidence of ”good” welds on Boeing–Canoga Park Thermal & Environmental Control Systems tubing is successful operation of PVM P6 Integrated Equipment Assembly and Long Spacer ammonia coolant systems for approximately 1 year with no detectable loss of ammonia. This hardware includes over 1000 tube welds.

Conservative calculations show the leak rate per weld could be as high as 1E–04 scc per second Helium without resulting in a violation of life requirements due to a low ammonia inventory at End of Life. Also, this assumes all of the welds are leaking at a rate of 1E–04 scc per second Helium which is extremely improbable since there are over 300 welds on each Integrated Equipment Assembly element on PVM P4, PVM S4, and PVM S6 with the radiators installed and ”good” welds are as leak tight as tubing. Although the probe detection method used was not calibrated to prove it could detect point leakages of 1E–05 sccs Helium, it should have that capability.

APPENDIX C PG–3 APPROVED EXCEPTIONS

The following is a list of exceptions to this document taken by Product Group 3. The exceptions to this document in no way eliminates the Contractor's responsibility for showing compliance to the sections 3.2 through 3.7 of the applicable specification.

PG3–01:

Nitrogen Interface Assembly (NIA) Thermal Vacuum and Thermal Cycling

SSP 41172 REQUIREMENT:

Thermal Vacuum and Thermal Cycling Tests of components. See Tables 4–1 and 5–1.

EXCEPTION:

Thermal Vacuum and Thermal Cycling tests will not be performed on the NIA as an assembly. Both tests will be performed on the electronic components that go into the NIA, and Thermal Cycling (but not Thermal Vacuum) testing will be performed on the active mechanical components.

RATIONALE:

A pair of solenoid valves contain the only electronic parts in the NIA. Because these valves are used individually as ORUs elsewhere on the station, they are already being run through a complete test program. The only other parts in the NIA are structure, a wire harness, a mechanical relief valve, and a mechanical pressure regulator. Because the valve and regulator are sealed units, a vacuum environment will have no effect on them other than pressure stresses (which are accounted for during proof and ultimate pressure tests). Therefore, only Thermal Cycling will be performed on them. The structure and wire harness are not susceptible to failures from Thermal Vacuum or Thermal Cycling tests.

PG3–3A:

Thermal Vacuum and Cycling

SSP 41172 REQUIREMENT:

Paragraph 5.1.2.1. This test detects material and workmanship defects prior to installation into a flight element by subjecting the article to a thermal vacuum environment.

EXCEPTION:

The Avionics Air Assembly will not require thermal vacuum acceptance testing to any flight units.

RATIONALE:

The subject test will be performed on the qualification unit. This is because the air bearings are not required to operate below 9 psia. Insufficient air mass will not support bearing lubrication. Therefore, damage to the air bearing would occur. This would be ”destructive testing” at acceptance.

PG3–10:

Node Structure

SSP 41172 REQUIREMENT:

Acoustic Vibration Qualification Test of Flight Elements. See Table 4–2.

EXCEPTION:

Acoustic Vibration Qualification Testing will not be performed on the Node Structural Assembly.

RATIONALE:

(1) United States Laboratory (USL) Acoustic Vibration test was cut (via Design Decision Package (DDP)) to save money and the Node verification was depending on the USL test.

(2) Flight critical, externally mounted equipment has been conservatively designed.

(3) Acoustic Vibration testing on the Meteoroids/Orbital Debris shield fasteners will verify their integrity.

(4) Analytical verification will be performed with attenuation data.

PG3–13:

Secondary Power Distribution Assembly (SPDA), Thermal Vacuum Test, Qualification and Acceptance

SSP 41172 REQUIREMENT:

Thermal Vacuum Qualification and Acceptance Test of Electronic or Electrical Equipment. See Tables 4–1 and 5–1.

EXCEPTION:

Thermal Vacuum Testing will not be run on the whole SPDA subassembly.

RATIONALE:

The SPDA structure and wire harness will be qualified by analysis. The main components (the RPCMs and the electrical connectors) will be qualified at the component level for this requirement. Acceptance testing on the RPCMs and electrical connectors will be performed to meet this requirement as well.

PG3–14:

SPDA, Thermal Cycling Test, Qualification and Acceptance

SSP 41172 REQUIREMENT:

Thermal Cycling Qualification and Acceptance Tests of Electrical/Electronic Equipment. See Tables 4–1 and 5–1.

EXCEPTION:

No Thermal Cycling Test will be run on the SPDA at the subassembly level.

RATIONALE:

The SPDA structure and electrical wire harness will be qualified by analysis for this requirement. The main components (RPCMs and electrical connectors) will already be qualified at the component level for this requirement. Acceptance testing on the RPCMs and electrical connectors will be performed to meet this requirement as well.

PG3–15:

EMI/EMC Test (Qualification)

SSP 41172 REQUIREMENT:

EMI/EMC Qualification Testing for Electrical/Electronic Equipment. See Table 4–1.

EXCEPTION:

No EMI/EMC Test will be performed on the SPDA at the subassembly level.

RATIONALE:

The SPDA consists of RPCMs that are Government–Furnished Equipment or Contractor–Furnished Equipment. Per the RPCM standard, the RPCMs delivered by Rocketdyne will meet all emission and susceptibility requirements of SSP 30237 validated by the methods in SSP 30238. It would be redundant to repeat the EMI tests at the SPDA level. An analysis of the test data collected from the Government–Furnished Equipment RPCM component EMI tests will be performed to verify that the specified EMI requirements of SSP 30237 have been satisfied.

PG3–17:

Emergency Egress Lighting System

SSP 41172 REQUIREMENT:

Paragraphs 4.2.2.3, 4.2.3.3, 5.1.2.3, and 5.1.3.3, Thermal Vacuum, including Depress/Repress, and Thermal Cycling Qualification and Acceptance.

EXCEPTION:

Test Temperatures will be limited to a range of +16 to +140 degrees F, depress rate of 25 psi per minute, repress rate of 3.1 psi per minute, and a pressure range from 15.2 to 1.9E–07 psia.

PG3–19:

Utility Outlet Panel (UOP)

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Acceptance Test, Table 5–1.

EXCEPTION:

Test will be performed to the design limits only, 30 degrees F to 140 degrees F operating, 15 degrees F to 150 degrees F nonoperating, with no vacuum operation.

RATIONALE:

The UOP is designed to operate in the specified temperature range; no vacuum operation is required. The UOP only has to tolerate depress/repress.

PG3–20:

Utility Outlet Panel

SSP 41172 REQUIREMENT:

Paragraph 4.2, Thermal Vacuum Qualification Test, Table 4–1.

EXCEPTION:

Test will be performed to the design limits only, 30 degrees F to 142 degrees F operating, 15 degrees F to 150 degrees F nonoperating, with no vacuum operation.

RATIONALE:

The UOP is designed to operate in the specified temperature range; no vacuum operation is required. The UOP only has to tolerate depress/repress.

PG3–23:

Refrigerator/Freezer Protoflight Tests

SSP 41172 REQUIREMENT:

Section 6.1. Subsequent assemblies to the first assembly subjected to protoflight tests shall be subjected to identical test.

EXCEPTION:

Refrigerator/Freezer Rack Assemblies subsequent to the first assembly will not require identical tests to the first assembly. Tests that validate the Refrigerator/Freezer Rack Assembly design, specifically EMI, acoustic (self noise), maximum Vibration (self generated), and Depress/Repress will not be repeated on subsequent assemblies to the first assembly.

RATIONALE:

The subject tests will be performed on the initial unit to validate the Refrigerator/Freezer Rack Assembly design. Repeating these tests on subsequent units does not provide additional benefit. Tests that verify workmanship, specifically Acoustic Vibration, Thermal Cycling, and Functional will be repeated on all units.

PG3–26:

General Light Assembly

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Qualification/Thermal Cycling.

EXCEPTION:

Verification of the nonoperating and operating thermal conditions for the general light assembly shall be as stated in the section 4's of envelope drawing 683–10044, Revision J, in lieu of what is in SSP 41172.

RATIONALE:

The Lights are only required to survive the nonoperating temperatures, they will be turned on and performance will be measured only while within specified operating temperatures.

PG3–29:

Vacuum system components

SSP 41172 REQUIREMENT:

Paragraphs 4.2.5 and 6.1.1, Thermal Cycling Test.

EXCEPTION:

No thermal cycling test will be performed on the manifolds, ducts, bellows, flex hoses, flanges, or couplings.

RATIONALE:

Active components of the vacuum system will be thermal tested as specified in ED 683–18005. The components listed above have no failure modes that may be detected by the thermal testing.

PG3–34:

Carbon Dioxide Removal Assembly (CDRA)

EXCEPTION:

The thermal cycling test will consist of 8 thermal cycles with the equipment operating within the specified operational range with the +/– 10 degrees F margins, and 24 cycles conducted in the nonoperating range.

RATIONALE:

SSP 41172 does not distinguish between operating and nonoperating temperatures. These components have a significant difference between the two ranges. The required margins will be applied to both (operating and nonoperating) cycling tests. Exception PG3–37 has been combined with this exception.

PG3–38:

USL Structure

SSP 41172 REQUIREMENT:

Table 4–2 (Acoustic Vibration).

EXCEPTION:

Acoustic Vibraton Qualification testing will not be performed on the element USL/Habitation (Hab) structural assemblies.

RATIONALE:

USL Acoustic Vibration Structural element test was taken out of the scope of structural testing for the USL/HAB (See DDP 149R1: as a program cost savings.). Flight critical, externally mounted equipment has been conservatively designed. External component acoustic vibration testing on the Meteoroid/Debris shield fasteners will assure their environmental integrity. Analytical verification will be performed with attenuation data.

PG3–39:

Rack Composite Structure

SSP 41172 REQUIREMENT:

Paragraph 4.4.3, Acoustic Vibration Test, Flight Element Qualification.

EXCEPTION:

Delete vibro–acoustic test for rack structures.

RATIONALE:

USL vibro–acoustic test was out (via DDP 149) as a cost savings and the rack was to be installed and verified during this activity. An analysis of rack responses to input vibro–acoustic loads will be performed using System Engineering and Analysis methods. The results of the analysis will be correlated with test verified responses. This will verify model accuracy. Resulting dynamic loads will then be used to compute equivalent static loads and stresses. Static loads will be applied to the detailed FEM model to obtain local stresses. The resulting loads will be added (or rssed) to the liftoff transient analysis loads to obtain the total load occurring at shuttle liftoff. The resulting stresses will then be used to verify static strength margins and fatigue life margins and validity of the static load test.

PG3–40:

Rack Composite Structure

SSP 41172 REQUIREMENT:

Table 5–2, (1) Toxic Off–Gassing, (2) Acoustic noise, (3) EMC, and (4) Mass properties.

EXCEPTION:

Delete:

(1) Toxic Off–Gassing;

(2) Acoustic Noise;

(3) EMC; and

(4) Only measure mass, not 2 axis center of gravity (paragraph 5.2.6.12). An analytic center of gravity will be provided for each rack structure.

RATIONALE:

(1) Rack material has already met or exceeded all off–gassing requirements;

(2) The rack does not produce any noise;

(3) The rack does not generate any electromagnetic waves; and

(4) The center of gravity for the rack structure will change after acceptance and before flight due to equipment/payload installed by the users. Integrated racks center of gravity will be determined.

PG3–41:

ITEM:

Thermal Radiator Rotary Joint Part Number 5839193–501

SSP 41172 REQUIREMENT:

Section 6.0. Requires that protoflight tests consist of the same number of thermal vacuum cycles, i.e. three, as are required for qualification testing in accordance with paragraph 4.2.2.

EXCEPTION:

The TRRJ #1 and #2 will be Protoflight Thermal Vacuum tested to one thermal vacuum cycle, rather than three cycles required by SSP 41172.

RATIONALE:

During qualification testing, the test article is subjected to three times the number of thermal cycles as the flight article will experience during acceptance testing. This is to ensure that the flight hardware can survive multiple acceptance tests in the event that the tests have to be repeated. Three Qualification Thermal Vacuum test cycles are performed to provide the needed margin over the one required for Acceptance Thermal Vacuum testing. This will also allow for Acceptance retest. This philosophy assures the design is adequate and provides confidence that the hardware will successfully pass Acceptance testing. Since the TRRJ undergoes a Protoflight rather than a traditional Qualification/Acceptance thermal vacuum test program, it is therefore not necessary to allow for retest. The hardware will undergo whatever retesting is necessary as part of the normal course of events.

PG3–42:

ITEMS:

Intermodule Ventilation (IMV) Valve Boeing Part Numbers:

235–3024–2–1 (Serial Numbers D0022)
235–3024–4–1 (Serial Numbers D0016, D0017, D0019, D0021, D0025, and D0027)
235–3024–5–1 (Serial Numbers D0012, D0023, and D0024)
235–3024–6–1 (Serial Numbers D0020)
235–3024–7–1 (Serial Numbers D0018 and D0026)
235–3024–8–1 (Serial Numbers D0029)

SSP 41172B REQUIREMENT:

Paragraph 5.1.4.3B. Component random vibration test levels and spectrums shall be the envelope of the following:

B. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime Contractor

EXCEPTION:

Applicability of the Random Vibration Workmanship Screening spectrum (Figure 5–2) is limited to the frequency range between 50 and 500 Hz.

RATIONALE:

The subject IMV Valves have been screened via the following tests: (1) a Reliability Acceptance Test performed at the controller level which combined three thermal cycles at a 10 degrees F change rate with a 6 grms random vibration excitation during the third thermal cycle in the axis normal to the board; (2) a controller level burn–in consisting of seven thermal cycles with a change rate of 10 degrees F per minute transition; and (3) acceptance testing as required by SSP 41172 except for vibration which deviated below 50 Hz and above 500 Hz. Based on sine sweep data in each axis, the following frequencies produced the greatest displacements: 216.95, 503.96, and 501.34 Hz. These levels were approximately 10.2 g peak. Many other resonances are also present, but most of these fall between 200 and 500 Hz and the response levels are at least a factor of two below the maximum resonance responses. Based on these results, the IMV Valve test response strain, where the correct vibration level was used, induces sufficient workmanship stress levels in the valve.

IMV Valve incorporated design and operational features that mitigate the impact of failure are: (1) the IMV Valve can be operated by an Intravehicular Activity (IVA) crewmember, through a flexible mechanical drive operable with a gloved hand, via the Remote Manual Operator (RMO); (2) redundancy is provided by two valves, one on each module – except the Airlock has only one valve; (3) the valve retains the last position selected; (4) removal and replacement can be accomplished on–orbit; and (5) the butterfly valve has dual seals to protect against vacuum.

PG3–43:

ITEM:

Common Berthing Mechanism (CBM) Actuation System Boeing Configuration End Item Number 683D27A

SSP 41172B REQUIREMENT:

Paragraph 4.2.5.3B. Component random vibration test levels and spectrums shall be the envelope of the following:

B. Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The CBM Actuation System will be qualification vibration tested to maximum flight levels. These levels do not envelope the acceptance test levels plus test tolerances defined in Figure 5–2 in all frequencies in any axis.

RATIONALE:

x– and z–axis: Maximum flight levels only fails to envelope acceptance vibration test levels plus test tolerances at greater than 900 Hertz. This is beyond the range of critical resonance frequencies (less than 500 Hertz) generally exhibited by electronic hardware. Since displacements at the higher frequencies are generally small, induced strains are likewise small. The higher frequencies do not significantly contribute to fatigue accumulation. Therefore, acceptance testing to minimum workmanship screening levels in the frequency range not enveloped by the qualification test is of minimum risk.

y–axis: The component was tested to the maximum flight environment. Although the maximum flight environment did not encompass the acceptance environment, the current plans for reflying these components do not include additional acceptance vibration testing. The risk of fatigue damage due to the launch environment is approximately 1.0 percent. Since the y–axis is parallel to the plane of the circuit boards, it is not a critical axis for workmanship screening. If an additional acceptance vibration test in the y–axis is ever required and the hardware passes, the risk of failure during flight is minimal for the reason stated above.

PG3–44:

ITEMS:

0.125 inch Feedthrough–mounted, Mated Pair Coupling Boeing Part Numbers:

683–19363–1
683–19364–1
683–19485–3
683–19485–4
683–19485–6

SSP 41172B REQUIREMENT:

Paragraph 4.2.3.3C.

(1) The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total.

(2) Each cycle shall have a one hour minimum dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on.

EXCEPTION:

The 0.125 inch Feedthrough–mounted Mated Pair Coupling Thermal Cycling Test is limited to a Thermal Extreme test of two cycles.

RATIONALE:

In lieu of a Thermal Cycle test, a Thermal Extreme test of two cycles was performed during qualification testing. The test was performed to simulate actual thermal exposure of the feedthrough components. During the test, the Coupling was operationally tested by mating and demating the Coupling at the temperature extremes of 35 degrees F and 129 degrees F with the unit pressurized at 15.2 psia. The test also consisted of a nonoperating test that exposed the coupling to –50 degrees F for 30 minutes and 170 degrees F for 30 minutes. In each case, a functional test was performed after completion of the temperature extreme test and the Coupling passed.

The 0.125 inch Feedthrough–mounted, Mated Pair Coupling is simple, robust, and rugged.

The predicted Mean Time Between Failure (MTBF) for a similarly designed Quick Disconnect (Boeing Part Number 683–16348) is 48,710 hours.

The 0.125 inch Feedthrough–mounted, Mated Pair Coupling has a criticality rating of 3.

PG3–45:

ITEMS:

0.125 inch Feedthrough–mounted, Mated Pair Coupling Boeing Part Numbers:

683–19363–1
683–19364–1
683–19485–3
683–19485–4
683–19485–6

SSP 41172B REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

(1) Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Thermal Cycling Test for the 0.125 inch Feedthrough–mounted, Mated Pair Coupling is not required

RATIONALE:

The 0.125 inch Feedthrough–mounted, Mated Pair Coupling is simple, robust, and rugged.

The predicted MTBF for a similarly designed Quick Disconnect (Boeing Part Number 683–16348) is 48,710 hours.

The 0.125 inch Feedthrough–mounted, Mated Pair Coupling has a criticality rating of 3.

PG3–46:

ITEM:

System On/Off Remote Control Boeing Part Number 219006

SSP 41172B REQUIREMENT:

Paragraph 4.2.2.2. The components that are required to operate during ascent, descent, and depressurization/repressurization shall be operating and monitored for arcing and corona during the initial reduction of pressure to the specified lowest levels.

EXCEPTION:

The System On/Off Remote Control is exempt from monitoring for corona.

RATIONALE:

Supply voltage for the System On/Off Remote Control is 12 V dc. Documents AFAPL–TR–65–122 and 50M05189 indicate that corona will not occur at this voltage level.

The System On/Off Remote Control has a criticality rating of 3.

PG3–47:

ITEMS:

Cold Plate Part Number 683–10041–01 AES Number 235 1410–1
Cold Plate Part Number 683–10041–04 AES Number 235 1440–1
Cold Plate Part Number 683–10041–10 AES Number 235 1500–1
Payload Regen Heat Exchanger Part Number 683–10042–01 AES Number 215 1340–1
SPCU Heat Exchanger Part Number 683–10042–02 AES Number 235 1350–1

SSP 41172 REQUIREMENT:

Random Vibration/Vibroacoustic Qualification and Acceptance Testing of Fluid or Propulsion Equipment. See Tables 4–1 and 5–1.

EXCEPTION:

Random Vibration and vibroacoustic tests will not be conducted on three Allied Signal Cold Plates and two Heat Exchangers.

RATIONALE:

(1) As part of the Qualification and Acceptance Requirements for these items, a proof pressure test is performed. Relative to the internal structure of these items, the pressure test will induce stresses that will be greater than what would be achieved during a random vibration test. Thus, the pressure test will serve as the best “screen” for manufacturing flaws.

(2) The items are designed to safety factors of 1.25 on yield and 2.0 for ultimate with the design loads derived from combined low and high frequency accelerations. This provides for additional conservatism appropriate for untested hardware.

(3) There are no components in these items that are sensitive to vibration environments. There are no electronics or high precision mechanisms that might break or misalign due to random vibration.

PG3–48:

ITEM:

0.125 inch Manual Valve Boeing Part Number 683–19393–1

SSP 41172B REQUIREMENT:

Paragraph 4.2.3.3C.

(1) The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total.

(2) Each cycle shall have a one hour minimum dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on.

EXCEPTION:

The 0.125 inch Manual Valve Qualification Thermal Cycling Test is limited to one Thermal Extreme test between ambient, operating extremes, and nonoperating extremes with soaking to achieve temperature stabilization plus one hour at each and functional test at ambient and operational extremes with a margin of +20 degrees F on the maximum operating and nonoperating extremes and –20 degrees F on the minimum operating and nonoperating extremes.

RATIONALE:

Retesting the 0.125 inch Manual Valve to 24 Thermal Cycles in Qualification and 8 Thermal Cycles in Acceptance will not enhance the serviceability or reliability of the valve.

The 1/8th inch valve, with load bearing parts made of stainless steel in a housing of 6061–T6 Aluminum, is simple, robust, rugged, and manually operated.

Leak and functional checks at minimum extreme operating temperature (55 degrees F) minus 20 degrees F and at extreme positive operating temperature (109 degrees F) plus 20 degrees F revealed the valve to operate within specification.

The thermal extreme test included the minimum nonoperating temperature of –30 degrees F minus 20 degrees F and the maximum nonoperating temperature 150 degrees F plus 20 degrees F.

The Qualification of the valve included a Burst Pressure (Factor of 4) Test after which the valve was functionally and leak tested and found to have no deformation and no defects in operability. Leakage was well within specification. Measured external and internal leakage was 3.77E–09 and 1.6E–08 sccs GHe against requirements of 1E–06 and 1E–03 sccs GHe, respectively, after 10,000 Open/Close cycles and 10,000 operating pressure cycles (pressure increase from 1.0 psia to 15.2 psia (maximum operating pressure)).

No deformation of EPR seals occurred at extreme pressures and temperatures or after pressure cycles and all measured leakage was well within specified values.

This valve design is similar in function and physical design to the Oxygen Supply Valve, Moog Part Number 1–4–00–51–27, developed in a similar manner by Carlton and used on Space Shuttle (31 valves on each Orbiter). It has accomplished 87 (flights) x 31 (valves) = 2697 missions without a single failure. Consequently, this similar valve currently enjoys an in–service reliability of one. A similar design was also used on the Apollo Saturn vehicle without experiencing any failures.

PG3–49:

ITEM:

0.125 inch Manual Valve Boeing Part Number 683–19393–1

SSP 41172B REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

(1) Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Thermal Cycling Test for the 0.125 inch Manual Valve is not required.

RATIONALE:

Retesting the 0.125 inch Manual Valve to 24 Thermal Cycles in Qualification and 8 Thermal Cycles in Acceptance will not enhance the serviceability or reliability of the valve.

The 1/8th inch valve, with load bearing parts made of stainless steel in a housing of 6061–T6 Aluminum, is simple, robust, rugged, and manually operated.

Leak and functional checks at minimum extreme operating temperature (55 degrees F) minus 20 degrees F and at extreme positive operating temperature (109 degrees F) plus 20 degrees F revealed the valve to operate within specification.

The thermal extreme test included the minimum nonoperating temperature of –30 degrees F minus 20 degrees F and the maximum nonoperating temperature 150 degrees F plus 20 degrees F.

The Qualification of the valve included a Burst Pressure (Factor of 4) Test after which the valve was functionally and leak tested and found to have no deformation and no defects in operability. Leakage was well within specification. Measured external and internal leakage was 3.77E–09 and 1.6E–08 sccs GHe against requirements of 1E–06 and 1E–03 sccs GHe, respectively, after 10,000 Open/Close cycles and 10,000 operating pressure cycles (pressure increase from 1.0 psia to 15.2 psia (maximum operating pressure)).

No deformation of EPR seals occurred at extreme pressures and temperatures or after pressure cycles and all measured leakage was well within specified values.

This valve design is similar in function and physical design to the Oxygen Supply Valve, Moog Part Number 1–4–00–51–27, developed in a similar manner by Carlton and used on Space Shuttle (31 valves on each Orbiter). It has accomplished 87 (flights) x 31 (valves) = 2697 missions without a single failure. Consequently, this similar valve currently enjoys an in–service reliability of one. A similar design was also used on the Apollo Saturn vehicle without experiencing any failures.

PG3–50:

ITEM:

Negative Pressure Relief Valve (NPRV) Boeing Part Number 683–16322–3

SSP 41172B REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.

(1) Paragraph 4.2.2.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 4.2.2.4, Supplementary Requirements. ALL REQUIREMENTS.

(3) Paragraph 4.2.2.5, Depress/Repress Vacuum Requirements. ALL REQUIREMENTS.

EXCEPTION:

Qualification Thermal Vacuum Test for the NPRV is replaced by a Functional Test at the maximum delta pressure at temperature extremes.

RATIONALE:

During Qualification, the exterior portion of the NPRV is held at 1E–05 Torr by the mass spectrometer in order to measure leakage. The interior portion of the NPRV is held at 15.3 +/– 0.01 psia (corresponding to an interior high pressure). As the test progresses through its cycles, the temperature is raised and lowered to qualification levels. The external and internal pressures remain essentially steady in order to detect any leakage which would be a cause of failure.

Since the NPRV is a pressure control device, the function of the NPRV cannot be tested with vacuum conditions maintained on both sides as required. Testing to the thermal vacuum requirements on both sides of the valve would not allow adequate functional testing of the device and would not expose NPRV to the maximum stress scenario.

Functional testing exposes the NPRV to the maximum stress scenario (i.e. maximum delta pressure at temperature extremes). The Functional test does not expose the device to vacuum conditions on both sides of the device as seen on–orbit during a depress/repress event. However, vacuum conditions on both sides of the device would represent a less stressful nonoperating scenario. The ability to survive this Thermal Vacuum scenario has been shown by the Material Usage Agreements (MUA) for component materials indicating that the integrity of materials is not effected by exposure to vacuum.

PG3–51:

ITEM:

Negative Pressure Relief Valve Boeing Part Number 683–16322–3

SSP 41172B REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.

(1) Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Thermal Vacuum Test for the NPRV is replaced by a Functional Test at the maximum delta pressure at temperature extremes.

RATIONALE:

During Acceptance, the exterior portion of the NPRV is held at 1E–05 Torr by the mass spectrometer in order to measure leakage. The interior portion of the NPRV is held at 15.3 +/– 0.01 psia (corresponding to an interior high pressure). As the test progresses through its cycles, the temperature is raised and lowered to acceptance levels. The external and internal pressures remain essentially steady in order to detect any leakage which would be a cause of failure.

Since the NPRV is a pressure control device, the function of the NPRV cannot be tested with vacuum conditions maintained on both sides as required. Testing to the thermal vacuum requirements on both sides of the valve would not allow adequate functional testing of the device and would not expose NPRV to the maximum stress scenario.

Functional testing exposes the NPRV to the maximum stress scenario (i.e. maximum delta pressure at temperature extremes). The Functional test does not expose the device to vacuum conditions on both sides of the device as seen on–orbit during a depress/repress event. However, vacuum conditions on both sides of the device would represent a less stressful nonoperating scenario. The ability to survive this Thermal Vacuum scenario has been shown by the MUAs for component materials indicating that the integrity of materials is not effected by exposure to vacuum.

PG3–52:

ITEMS:

CBM Control Panel Assembly Part Numbers 2355260–1–1, 2355260–2–1, and 2355260–3–1
CBM Bolt Actuator Part Number 2357650–2–1
CBM Latch Actuator Part Number 2357660–2–1

SSP 41172B REQUIREMENT:

Paragraph 4.2.5.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Qualification Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) Acceptance Test Procedure (ATP)/Qualification Test Procedure (QTP) related failures. QTP inductor failed during random vibration test due to incorrect level. All units now have the inductor epoxied to the circuit board and the test was re–run to the correct levels.

(2) Two exceptions to SSP 41172 have been approved:

CR1030 for random vibration test conducted with the spectrum below requirement at the high and low frequencies.

CR1085 authorizing the collection of performance data during the last thermal cycle only versus the first and last cycles.

(3) CBM Components have a calculated MTBF as follows:

Control Panel Assembly 57,000 hours

Latch Actuator 195,000 hours

Bolt Actuator 162,000 hours

(4) The manufacturing process incorporates significant screens for defects at each level of assembly (board level, box level, and assembly level).

(5) All items are classified as criticality 2R.

(6) The items' ability to function on–orbit after consequences of a failure include:

Controller Panel Assembly – The Controller panel design utilizes a master slave relationship. Each of the four Controller panels have master capability and a separate 1553 feed; likewise, the design utilizes a redundant 485 bus structure which provides for the transfer of commands from the master to all twenty slave controllers.

Capture Latch Actuator – Capture can be accomplished with any three of the four capture latches on each active CBM.

Bolt Actuator – Mate and rigidization can be accomplished with any fifteen of the sixteen power bolt assemblies on each active CBM.

(7) There exist the ability to restore function for these items on–orbit. Subsequent to mate and rigidization, the actuators and controller panel assemblies can be removed via IVA; thus, any defective parts on mated CBMs can be replaced on–orbit.

PG3–53:

ITEMS:

CBM Control Panel Assembly Part Numbers 2355260–1–1, 2355260–2–1, and 2355260–3–1
CBM Bolt Actuator Part Number 2357650–2–1
CBM Latch Actuator Part Number 2357660–2–1

SSP 41172B REQUIREMENT:

Paragraph 5.1.4.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Acceptance Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) ATP/QTP related failures. QTP inductor failed during random vibration test due to incorrect level. All units now have the inductor epoxied to the circuit board and the test was re–run to the correct levels.

(2) Two exceptions to SSP 41172 have been approved:

CR1030 for random vibration test conducted with the spectrum below requirement at the high and low frequencies.

CR1085 authorizing the collection of performance data during the last thermal cycle only versus the first and last cycles

(3) CBM Components have a calculated MTBFs as follows:

Control Panel Assembly 57,000 hours

Latch Actuator 195,000 hours

Bolt Actuator 162,000 hours

(4) The manufacturing process incorporates significant screens for defects at each level of assembly (board level, box level, and assembly level).

(5) All items are classified as criticality 2R.

(6) The items' ability to function on–orbit after consequences of a failure include:

Controller Panel Assembly – The Controller panel design utilizes a master slave relationship. Each of the four Controller panels have master capability and a separate 1553 feed; likewise, the design utilizes a redundant 485 bus structure which provides for the transfer of commands from the master to all twenty slave controllers.

Capture Latch Actuator – Capture can be accomplished with any three of the four capture latches on each active CBM.

Bolt Actuator – Mate and rigidization can be accomplished with any fifteen of the sixteen power bolt assemblies on each active CBM.

(7) There exist the ability to restore function for these items on–orbit. Subsequent to mate and rigidization, the actuators and controller panel assemblies can be removed via IVA; thus, the defective parts on mated CBMs can be replaced on–orbit.

PG3–54:

ITEMS:

Avionics Air Assembly (AAA) Part Number SV809992–5
IMV Fan Part Number SV809111–6

SSP 41172B REQUIREMENT:

Paragraph 4.2.3.3. Test Levels and Duration.

(1) Paragraph 4.2.3.3C, Duration. The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total.

EXCEPTION:

The duration of the Qualification Thermal Cycle Test for the nonoperating temperature range of the AAA and the IMV Fan shall be three thermal cycles.

RATIONALE:

Nonoperating thermal cycle testing is used to verify that the items will survive the nonoperating temperature range. Survivability can be demonstrated through fewer thermal cycles. Workmanship will continue to be verified via the thermal cycle testing for the operating temperature range of 24 cycles for qualification.

PG3–55:

ITEMS:

Avionics Air Assembly Part Number SV809992–5
IMV Fan Part Number SV809111–6

SSP 41172B REQUIREMENT:

Paragraph 5.1.3.3. Test Levels and Duration.

(1) Paragraph 5.1.3.3C, Duration. The minimum number of temperature cycles shall be eight.

EXCEPTION:

The duration of the Acceptance Thermal Cycle Test for the nonoperating temperature range of the AAA and the IMV Fan shall be one thermal cycle.

RATIONALE:

Nonoperating thermal cycle testing is used to verify that the items will survive the nonoperating temperature range. Survivability can be demonstrated through fewer thermal cycles. Workmanship will continue to be verified via the thermal cycle testing for the operating temperature range of eight cycles for acceptance.

PG3–56:

ITEMS:

Common Cabin Air Assembly (CCAA) Part Number SV806610–3
Inlet ORU Assembly Part Numbers SV811840–3 and SV811840–4

SSP 41172B REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)..." This requires protoflight hardware thermal cycling duration to be 24 thermal cycles as indicated in paragraph 4.2.3.3C.

EXCEPTION:

The Protoflight Thermal Cycle Test for the CCAA and the Inlet ORU Assembly shall be eight operating thermal cycles and one nonoperating thermal cycle.

RATIONALE:

Nonoperating thermal cycle testing is used to verify that the items will survive the nonoperating temperature range. Survivability can be demonstrated through fewer thermal cycles than indicated in the Qualification Thermal Cycling requirements. Workmanship will continue to be verified via the thermal cycle testing for the operating temperature range of eight cycles.

PG3–57:

ITEM:

Internal Thermal Control System Coldplates Boeing Part Numbers 683–10041–1, 683–10041–4, 683–10041–5, 683–10041–6, 683–10041–7, 683–10041–8, 683–10041–9, and 683–10041–10

SSP 41172B REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.

Paragraph 4.2.3.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Qualification Thermal Cycling Test for the Internal Thermal Control System Coldplates is replaced with a Thermal Shock Test. The Thermal Shock Test consists of the following:

With the Coldplate thermal chamber temperature at ambient, conduct Coldplate functional tests. Decrease thermal chamber temperature to –45 degrees F and hold the chamber at this temperature for one hour until the Coldplate temperature achieves –45 degrees F +/–5 degrees F with transient less than 4.5 degrees F per hour. Inject coolant between 74 to 80 degrees F at a flowrate between 300 to 306 lb per hour and a pressure less than 100 psia into the Coldplate inlet port. Following dwell, increase temperature to ambient and conduct operability tests in conjunction with the collapse pressure test.

RATIONALE:

Characteristics of the Internal Thermal Control System Coldplate design and identified application are such that the Thermal Cycle Tests would not enhance the serviceability or reliability of the Coldplates.

The Coldplate design is not susceptible to thermal stress in its normal operational environment. The worst case nonoperational environment is Node 1 Dry Conditions.

Maximum Operating Pressure, Proof Pressure, Internal Partial Vacuum, and Low Pressure Leak Rate Tests were performed for Qualification. These tests stressed the Brazed Joints of the Coldplates in excess of which the Qualification Thermal Cycling Tests would have stressed. There was no deformation or failure.

Coldplate 683–10041–6 Thermal Shock Test resulted in the Coldplate and fins experiencing a thermal transient of approximately 23 degrees F per second (1380 degrees F per minute) over a period of five seconds. The thermal transient had no deleterious effects (no deformation or failure) on the Coldplate mounting surface, brazed joints, or fins.

The test objective of the Coldplate Thermal Shock Test is to thermally stress the Coldplate brazed joints to ensure the suitability of the Coldplate to survive extreme low temperature without deformation or failure. The resulting heat transfer forces the Coldplate to experience thermal transients as follows:

Top Plate: 101 degrees F in approximately 20 seconds

Bottom Plate: 115 degrees F in approximately 20 seconds.

The pretense to do only the low temperature level is the brazing process is accomplished at high temperature with the pieces to be brazed at equilibrium (or relaxed), i.e., not under load. After the high temperature process of brazing is accomplished, there is little stress in the joint until the brazed item temperature begins to decline. When the temperature drops, the brazed joint experiences a stress increase which continues until the normal operating temperature is reached. Normal stress fluctuations about this norm will occur until some change in environmental conditions such as pressure or temperature occurs. In an extreme temperature change, the stress levels decreases with increased temperatures and stress levels increases with reduced temperatures. Therefore, the low temperature thermal extreme of –45 degrees F would be the worst case stress for the Coldplate brazed joints.

PG3–58:

ITEM:

Internal Thermal Control System Coldplates Boeing Part Numbers 683–10041–1, 683–10041–4, 683–10041–5, 683–10041–6, 683–10041–7, 683–10041–8, 683–10041–9, and 683–10041–10

SSP 41172B REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

(1) Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Thermal Cycling Test for the Internal Thermal Control System Coldplates is not required.

RATIONALE:

Characteristics of the Thermal Control System Coldplate design and identified application are such that the Thermal Cycle Tests would not enhance the serviceability or reliability of the Coldplates.

The Coldplate design is not susceptible to thermal stress in its normal operational environment. The worst case nonoperational environment is Node 1 Dry Conditions.

Maximum Operating Pressure, Proof Pressure, Internal Partial Vacuum, and Low Pressure Leak Rate Tests were performed for Acceptance. These tests stressed the Brazed Joints of the Coldplates in excess of which the Acceptance Thermal Cycling Tests would have stressed. There was no deformation or failure. In additional, each Coldplate was checked for flatness following Proof Pressure to ensure no deformation.

The Brazing process is a certified process verified during qualification test and ensured for repeatability. Application and inspection of the Brazing Process is defined in Allied Signal Specification WBS49, “Process Specification – Welding, Brazing and soldering – Brazing Space Station Freedom”. Non Destructive Inspection methods and requirements are included.

PG3–59:

ITEM:

Videotape Recorder (VTR) Part Number 683–51020–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.

(1) Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Random Vibration test on the VTR will not be performed.

RATIONALE:

VTR is launched stowed, encased in foam, in a benign vibration environment. Therefore, an acceptance vibration test is a workmanship screen only, and the qualification test is conducted to demonstrate margin over the acceptance test environment. Structural analysis of the VTR shows that due to required design modifications to the COTS tape recorder subassembly, the unit will not withstand the minimum acceptance screening environment. The circuit cards will be subjected to card–level random vibration screening. The VTR assembly will be subjected to inspections, functional tests, and thermal cycling tests to verify workmanship. Additionally, this hardware is criticality 3 and redundant units are present on–orbit. Thus, with the exception to not perform an acceptance random vibration test on the VTR, the need to demonstrate margin in a qualification random vibration test on the VTR is not necessary.

PG3–60:

ITEM:

Videotape Recorder Part Number 683–51020–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.

(1) Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Random Vibration test on the VTR will not be performed.

RATIONALE:

VTR is launched stowed, encased in foam, in a benign vibration environment. Therefore, an acceptance vibration test is a workmanship screen only. Structural analysis of the VTR shows that due to required design modifications to the COTS tape recorder subassembly, the unit will not withstand the minimum acceptance screening environment. The circuit cards will be subjected to card–level random vibration screening. The VTR assembly will be subjected to inspections, functional tests, and thermal cycling tests to verify workmanship. Additionally, this hardware is criticality 3 and redundant units are present on–orbit. Therefore, the limited value of the screen permits an exception to the Acceptance Random Vibration test for the VTR.

PG3–61:

ITEMS:

Bolt Actuator Part Number 2357650–2–1
Latch Actuator Part Number 2357660–2–1

SSP 41172B REQUIREMENT:

Paragraph 5.1.2.4. Functional test shall be conducted at the maximum and minimum predicted temperature levels during the first and last cycle and after return of the component to ambient temperature. During the remainder of the test, electrical and electronic components, including redundant circuits and paths, shall be monitored for failures and intermittences. Components with rotating equipment that uses air as a lubricant should not be spinning when the atmosphere is removed.

EXCEPTION:

Collection of performance data is not required during the first thermal vacuum cycle. The Bolt Actuator and Capture Latch Actuator will be operated for 8 thermal vacuum cycles with electronic components being monitored for failures and intermittences. A functional test which includes the collection of performance data will be conducted at maximum and minimum predicted temperatures and at ambient temperatures at the conclusion of the 8 thermal vacuum cycles.

RATIONALE:

Actuators are thermal cycle insensitive. Life cycle qualification shows that there is no change in performance between first and last cycles. Qualification actuators were life cycle tested in vacuum under thermal cycles for approximately 270 operational cycles with no change in performance. An additional 200 life cycles were conducted on the original qualification actuator after rework to verify that the rework procedure did not affect life cycle performance. The qualification actuator did not have any failures and there were no change in performance after the additional life cycles.

PG3–62:

ITEMS:

Bolt Actuator Part Number 2357650–2–1
Latch Actuator Part Number 2357660–2–1

SSP 41172B REQUIREMENT:

Paragraph 5.1.3.4. Functional tests shall be conducted during the first and last thermal cycles at the maximum and minimum predicted temperatures and after return of the component to ambient. During the remainder of the test, electrical components shall be cycled through various operational modes and parameters monitored for failures and intermittence.

EXCEPTION:

Collection of performance data is not required during the first thermal cycle. The Bolt Actuator and Capture Latch Actuator will be operated for 8 thermal vacuum cycles with electronic components being monitored for failures and intermittences. A functional test which includes the collection of performance data will be conducted at maximum and minimum predicted temperatures and at ambient temperatures at the conclusion of the 8 thermal vacuum cycles.

RATIONALE:

Actuators are thermal cycle insensitive. Life cycle qualification shows that there is no change in performance between first and last cycles. Qualification actuators were life cycle tested in vacuum under thermal cycles for approximately 270 operational cycles with no change in performance. An additional 200 life cycles were conducted on the original qualification actuator after rework to verify that the rework procedure did not affect life cycle performance. The qualification actuator did not have any failures and there were no change in performance after the additional life cycles.

PG3–63:

ITEM:

CBM Capture Latch Part Number 683–13434–6

SSP 41172B REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

(1) Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Thermal Cycling Test for the CBM Capture Latch is not required.

RATIONALE:

The CBM Capture Latch is predominately a moving mechanical assembly with one 5 V dc switch. A single temperature cycle conducted during the Acceptance Thermal Vacuum test is adequate to verify manufacturing workmanship of the Capture Latch Switch Installation (two hand solder joints). This test will screen any ”binding” that occurs as a result of materials expansion or contraction.

PG3–64:

ITEMS:

IMV Valve Part Numbers 2353024–2–1, 2353024–4–1, 2353024–5–1, 2353024–6–1, 2353024–7–1, and 2353024–8–1

SSP 41172B REQUIREMENT:

Paragraph 4.2.5.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Qualification Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) Failures detected and Corrective actions implemented during the QTP/ATP process. Six Failure Reporting and Corrective Actions (FRACA) were initiated during Qualification Testing. Four involved leakage above the requirement during the Thermal Cycling test. The flange seal, seal body, and butterfly seal were replaced and the unit successfully passed retest. One FRACA was issued when the valve would not actuate at 20 degrees F. Failure analysis concluded test setup problem and testing continued without further failures. The final FRACA was initiated when the visual indication did not match the electrical position indicator during the vibration test (z–axis). Failure analysis concluded the RMO and cable had been twisted 180 degrees relative to the valve when the test setup was changed between y– and z–axes. The valve was re–rigged and testing completed successfully.

(2) Other SSP 41172 exception:

CR 1029: The first 14 valves were acceptance random vibration tested to a vibration spectrum that was below the screening spectrum in the high and low frequencies. Remaining valves were tested to the full vibration spectrum and levels.

(3) The IMV Valve has a calculated MTBFs of 109,000 hours.

(4) A Manual Remote Operator, in the valve design, provides redundancy for valve operation.

(5) The valve design is based on proven technology and manufacturing processes. The valve controller is used on other valves in the station. The controller is used in the CDRA Pump Fan Motor Controller, Vent and Relief Valve, CBM actuators, and Internal Thermal Control System (ITCS) valves. The controller assembly contains small low part density, rigidly mounted boards; the EMI board is potted in a pliable potting compound.

(6) Screening tests in addition to the required SSP 41172 test included:

(a) Additional lower–level qualification vibration testing was performed prior to the full SSP 41172 spectrum and level testing. The cumulative stress introduced by the lower level qual test provided additional opportunities to precipitate latent defects.

(b) A Reliability Acceptance Test was performed at the IMV Controller assembly level for three cycles from 0 to 140 degrees F at 6.0 grms with a 90 minute dwell at the maximum and minimum vibration levels. The Reliability Acceptance Test was performed at the IMV Controller assembly level with random vibration on the third cycle.

(c) 100 electrically actuated open–to–close cycles were performed on the valve.

(7) The IMV valve function may be restored on–orbit for mated modules via valve replacement.

PG3–65:

ITEM:

0.125 inch Three–Way Valve Part Number 683–19446–1

SSP 41172B REQUIREMENT:

Paragraph 4.2.5.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Qualification Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) Failures detected and Corrective actions implemented during the QTP/ATP process. A single failure was detected via the ATP process. During ATP of the fifth flight valve, a pinched wire was detected during the functional test following random vibration. A design change was implemented and all valves were modified to preclude a recurrence of this anomaly.

(2) The 0.125 inch Three–Way Valve has a calculated MTBFs of 311,000 hours.

(3) The valve design is based on proven technology and manufacturing processes. Electrical connections are hard wired to the solenoid and latch actuator, and there are no circuit boards in the assembly.

(4) Function on–orbit including consequences of a failure. The Sample Distribution System (SDS) system is 1 failure tolerant. If one Node 1 0.125 inch Three–Way Valve fails stuck in either position A or B, the Atmosphere Revitalization (AR) SDS system can still take samples from all connected modules.

(5) The 0.125 inch Three–Way valve function may be restored on–orbit. The valve is a Configuration Maintenance Item and can be replaced on orbit.

PG3–66:

ITEM:

IMV Valve Part Numbers 2353024–2–1, 2353024–4–1, 2353024–5–1, 2353024–6–1, 2353024–7–1, and 2353024–8–1

SSP 41172B REQUIREMENT:

Paragraph 5.1.4.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Acceptance Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) Failures detected and Corrective actions implemented during the QTP/ATP process. Six FRACAs were initiated during Qualification Testing. Four involved leakage above the requirement during the Thermal Cycling test. The flange seal, seal body, and butterfly seal were replaced and the unit successfully passed retest. One FRACA was issued when the valve would not actuate at 20 degrees F. Failure analysis concluded test setup problem and testing continued without further failures. The final FRACA was initiated when the visual indication did not match the electrical position indicator during the vibration test (z–axis). Failure analysis concluded the RMO and cable had been twisted 180 degrees relative to the valve when the test setup was changed between y– and z–axes. The valve was re–rigged and testing completed successfully.

(2) Other SSP 41172 exception:

CR 1029: The first 14 valves were acceptance random vibration tested to a vibration spectrum that was below the screening spectrum in the high and low frequencies. Remaining valves were tested to the full vibration spectrum and levels.

(3) The IMV Valve has a calculated MTBFs of 109,000 hours.

(4) A Manual Remote Operator, in the valve design, provides redundancy for valve operation.

(5) The valve design is based on proven technology and manufacturing processes. The valve controller is used on other valves in the station. The controller is used in the CDRA Pump Fan Motor Controller, Vent and Relief Valve, CBM actuators, and ITCS valves. The controller assembly contains small low part density, rigidly mounted boards; the EMI board is potted in a pliable potting compound.

(6) Screening tests in addition to the required SSP 41172 test included:

(a) Additional lower–level qualification vibration testing was performed prior to the full SSP 41172 spectrum and level testing. The cumulative stress introduced by the lower level qual test provided additional opportunities to precipitate latent defects.

(b) Reliability Acceptance Test was performed at the IMV Controller assembly level for three cycles from 0 to 140 degrees F at 6.0 grms with a 90 minute dwell at the maximum and minimum vibration levels. Reliability Acceptance Test was performed at the IMV Controller assembly level with random vibration on the third cycle.

(c) 100 electrically actuated open–to–close cycles were performed on the valve.

(7) The IMV valve function may be restored on–orbit for mated modules via valve replacement.

PG3–67:

ITEM:

0.125 inch Three–Way Valve Part Number 683–19446–1

SSP 41172B REQUIREMENT:

Paragraph 5.1.4.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Acceptance Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) Failures detected and Corrective actions implemented during the QTP/ATP process. A single failure was detected via the ATP process. During ATP of the fifth flight valve, a pinched wire was detected during the functional test following random vibration. A design change was implemented and all valves were modified to preclude a recurrence of this anomaly.

(2) The 0.125 inch Three–Way Valve has a calculated MTBFs of 311,000 hours.

(3) The valve design is based on proven technology and manufacturing processes. Electrical connections are hard wired to the solenoid and latch actuator, and there are no circuit boards in the assembly.

(4) Function on–orbit including consequences of a failure. The SDS system is 1 failure tolerant. If one Node 1 0.125 inch Three–Way Valve fails stuck in either position A or B, the AR SDS system can still take samples from all connected modules.

(5) The 0.125 inch Three–Way valve function may be restored on–orbit. The valve is a Configuration Maintenance Item and can be replaced on orbit.

PG3–68:

ITEM:

Internal Thermal Control System Coldplates Boeing Part Numbers 683–10041–1, 683–10041–4, 683–10041–5, 683–10041–6, 683–10041–7, 683–10041–8, 683–10041–9, and 683–10041–10

SSP 41172B REQUIREMENT:

Paragraph 4.2.2.5.1. ALL REQUIREMENTS.

EXCEPTION:

Qualification Thermal Vacuum Depress/Repress Test for the Internal Thermal Control System Coldplates is not required.

RATIONALE:

Since the Coldplates are passive, the Thermal Vacuum Depress/Repress is limited to providing assurance of the Coldplates ability to withstand the design delta pressure.

The Coldplates are constructed of metal (nickel alloy, stainless steel, and brazing).

The Coldplates have no imbedded electronics which can fail during Thermal Vacuum Depress/Repress.

The Coldplates are designed and tested to operate at 100 psi (approximately 7 atmospheres) in their normal operational limits. They are also burst tested to 200 psi (approximately 14 atmospheres). Thus, the delta pressure across the surface of the Coldplates was more severe during Pressure/Leak testing (paragraphs 4.2.10 and 4.2.11) than it would be during Depress/Repress testing.

Since the difference between Pressure/Leak tests and Depress/Repress Test is the Depress/Repress test is conducted in a vacuum and the Pressure/Leak test has a greater delta pressure, the intent of the Depress/Repress test is achieved through analyses using the pressure test data to determine the Coldplates ability to withstand 180 days at hard vacuum. The analyses show the Coldplates design is sufficient to withstand 180 days at hard vacuum followed by a return to ambient conditions and operation (See 97–68985–7, 3.3.19). The analysis further shows a safety margin of 3.25 for the collapse pressure condition of –30 psid (coldplate internal pressure) (See 97–68985–8, 3.3.20).

Stress Analysis (based on pressure tests) also show that, since the maximum stress encountered for all operating conditions (including depressurization) is less than the yield strength, the Coldplates can experience 30 depressurization events without suffering any permanent deformation. The analysis revealed a safety margin of 1.96 for the depressurization event (See 97–68985–1, 3.3.13 and 97–68985–4, 3.3.14).

PG3–69:

ITEM:

Variable Air Volume Damper Assembly Part Number 683–15144–003

SSP 41172B REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification

Paragraph 4.2.2.5.2, Depress/Repress Vacuum Automated Power Down Requirements. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Depress/Repress Test will not be performed on the Variable Air Volume Damper Assembly. The Variable Air Volume Damper Assembly will be qualified by analysis.

RATIONALE:

The Variable Air Volume Damper Assembly consists of the Variable Air Volume Damper and an actuator. The actuator portion of the assembly was qualified and accepted in accordance with the requirements of SSP 41172.

The Variable Air Volume Damper is composed of a rectangular section of aluminum ducting which houses a flapper blade that blocks the air flow when closed. The flapper blade is attached to the housing by a shaft. The blade is a one piece machined aluminum part with a hole down the center for the shaft to be inserted. The shaft is secured by a spring pin. The spring pin is a “C” shaped part that is compressed and inserted into the hole. The pin exerts a force on the inside diameter of the hole preventing it from backing out. Proper installation of the spring can be verified by visual inspection and once installed properly, the spring rests in a recessed area which will preclude it from coming out under vibration loading.

The three load conditions used during analysis were (1) launch/landing, 23.9 g maximum; (2) internal duct pressure, 0.4 psig; and (3) crew induced loads, T = 11 ft–lbs. and 50 lbs. push or pull. The factors of safety used during analysis were 2.0 for ultimate loads, 1.25 for yield, and 1.15 fitting factor for structures verified by analysis only.

The unit is contained in the Cabin Air Ducts where the worse case temperatures would be approximately 45 to 85 degrees F. Thermal analysis and tolerance stack ups show the unit can withstand temperatures well below zero without experiencing interferences which would preclude proper valve operation.

The 0.4 psig is based upon the dead head pressure. The air flow stops during a depress event so the problem turns into a venting analysis. The equivalent hole size in the ducts are such that the maximum delta P (due to the venting) occurs when the cabin is approximately 1.0E–03 psia and the ducts are at approximately 1.0E–02 psia. This is far below the 0.4 psig level.

PG3–70:

ITEM:

Variable Air Volume Damper Assembly Part Number 683–15144–003

SSP 41172B REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Random Vibration Test will not be performed on the Variable Air Volume Damper Assembly. The Variable Air Volume Damper Assembly will be qualified by analysis.

RATIONALE:

The Variable Air Volume Damper Assembly consists of the Variable Air Volume Damper and an actuator. The actuator portion of the assembly was qualified and accepted in accordance with the requirements of SSP 41172.

The Variable Air Volume Damper is composed of a rectangular section of aluminum ducting which houses a flapper blade that blocks the air flow when closed. The flapper blade is attached to the housing by a shaft. The blade is a one piece machined aluminum part with a hole down the center for the shaft to be inserted. The shaft is secured by a spring pin. The spring pin is a “C” shaped part that is compressed and inserted into the hole. The pin exerts a force on the inside diameter of the hole preventing it from backing out. Proper installation of the spring can be verified by visual inspection and once installed properly, the spring rests in a recessed area which will preclude it from coming out under vibration loading.

The three load conditions used during analysis were (1) launch/landing, 23.9 g maximum; (2) internal duct pressure, 0.4 psig; and (3) crew induced loads, T = 11 ft–lbs. and 50 lbs. push or pull. The factors of safety used during analysis were 2.0 for ultimate loads, 1.25 for yield, and 1.15 fitting factor for structures verified by analysis only.

The Factors of Safety applied during the analysis of untested structure gives a very high degree of confidence that the Variable Air Volume Damper will operate as designed after being subjected to the launch environment.

The unit is contained in the Cabin Air Ducts where the worse case temperatures would be approximately 45 to 85 degrees F. Thermal analysis and tolerance stack ups show the unit can withstand temperatures well below zero without experiencing interferences which would preclude proper valve operation.

PG3–71:

ITEMS:

Emergency Egress Lights – with 57–Inch Light Strip Part Number 683–26007–2
Emergency Egress Lights – with 36–Inch Light Strip Part Number 683–26007–3

SSP 41172B REQUIREMENT:

Paragraph 4.2.5.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Qualification Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

The Emergency Egress Lights contain a Power Supply and Light Strip. The Power Supply is currently being redesigned and will be random vibration tested with Power On and Monitoring. This exception is for the light strips portion of the Emergency Egress Lights only.

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) Failures detected and Corrective actions implemented during the QTP/ATP process. During the Qualification Humidity Verification test, it was noted that a single Light Emitting Diode (LED) was not lit. An examination of photographs showed that the light strip was bent in one direction during the bend radius test, and bent in the opposite direction when placed on the humidity qualification test fixture. Both light strips were replaced with flight hardware which had already passed acceptance testing. All qualification testing was done with the new light strips.

(2) The Emergency Egress Lights have a calculated MTBFs of 704,000 hours for those containing the 57–inch light strip and 1,111,600 hours for those containing the 36–inch light strip.

(3) The light strips are a series of LEDs soldered to a bus. This design is simple, rugged, and easily inspected.

(4) The Emergency Egress Lights are classified as criticality 1R.

(5) The ability to function on–orbit through a failure. Each exit has light strips containing from 130 to 214 LEDs wired in parallel. A failure of one or more LEDs would not prevent the light strips from illuminating the exit. Additionally, each light strip is being illuminated to 150 percent of the design requirement.

(6) The Emergency Egress Lights function may be restored on–orbit. The light strips are designed for replacement on–orbit. Spares have been purchased. A special tool and detailed instructions for replacement are currently available.

PG3–72:

ITEM:

General Luminaire Light Assembly Part Number 219003

SSP 41172B REQUIREMENT:

Paragraph 4.2.5.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Qualification Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) Failures detected and Corrective actions implemented during the QTP/ATP process. During qualification and acceptance test of the General Luminaire Light Assembly Lamp Housing Assemblies (LHA) and Baseplate Ballast Assemblies (BBA), the supplier experienced a total of six failures/anomalies during vibration and post–vibration testing. The failure and findings are as follows:

(a) Fasteners were discovered to be broken on qualification unit after vibration test because the vibration table surface was not counter–bored before securing BBA fixture to vibration table;

(b) BBA qualification unit failed to turn–on as a result of a broken wire to Q6 transistor wire. The cause was deemed as workmanship. This wire was rerouted and secured in place with epoxy. This change was incorporated into all flight units;

(c) LHA qualification unit failed to turn–on due to a broken wire to lamp filament as a result of supplier’s outside test lab failure to properly ramp–up to required vibration ‘g’ level. LHA unit was subjected to a shock level test instead of a random vibration test;

(d) LHA and BBA flight units was exposed to higher vibration levels than required for acceptance test at 450 Hertz as a result of operator error. However, this did not exceed the qualification test levels for the LHA and BBA;

(e) LHA flight unit nut and washer came off inside LHA during vibration test. Proper torque was not applied on hardware and the cause was deemed as workmanship. No other failure occurred during vibration test; and

(f) BBA flight unit failed to turn–on after vibration test as a result of a transistor (Q1) failure. Q1 was removed and replaced and acceptance test was repeated with no additional failures.

(2) General Luminaire Light Assembly components have calculated MTBFs of 3,000,000 hours for the LHA and 1,929,000 hours for the BBA.

(3) The Design is rugged, durable, and easily inspected.

(4) The General Luminaire Light Assemblies are classified as criticality 2R.

(5) Function on–orbit including consequences of a failure. The General Luminaire Light Assemblies are redundant on–orbit (8 each in Node 1 and Italian Node 2, 12 in USL and Italian Node 3, and 4 in Airlock). A failure to any one General Luminaire Light Assembly will not cause a catastrophic failure to the remaining General Luminaire Light Assemblies. A simultaneous loss of all can only occur with the loss of station provided power (Power buses A and B) that supply the required 120 V dc input voltage.

(6) The General Luminaire Light Assembly function may be restored on–orbit. The General Luminaire Light Assembly is made of a LHA and a BBA. Both the LHA and BBA are identified as ORUs and are required to be removable and replaceable in order to restore function on–orbit in the event of a failure.

PG3–73:

ITEM:

System On/Off Remote Control Part Number 219006–1

SSP 41172B REQUIREMENT:

Paragraph 4.2.5.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Qualification Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) Failures detected and Corrective actions implemented during the QTP/ATP process. No failures were experienced during qualification and acceptance vibration test of the System Remote On/Off Control Assemblies (SRCA).

(2) Qualification and Acceptance testing has been performed as described in SSP 41172 with the following approved exception:

CR 1052: The SRCAs was exempt from monitoring for Corona due to its low supply voltage (12 V dc). Documents AFAPL–TR–65–122 and 50M05189 indicate that corona will not occur at this voltage level.

(3) The SRCAs have a calculated MTBFs of 48,840,000 hours.

(4) The SRCAs are classified as criticality 3.

(5) Function on–orbit including consequences of a failure. The SRCAs provides a remote means for turning element lights on and off. In the event of a failure, this convenience would be lost and the light would be controlled at the General Luminaire Light Assemblies.

(6) The SRCA function may be restored on–orbit. The SRCAs are ORUs and are required to be removable and replaceable in order to restore function on–orbit in the event of a failure.

PG3–74:

ITEMS:

Emergency Egress Lights – with 57–Inch Light Strip Part Number 683–26007–2
Emergency Egress Lights – with 36–Inch Light Strip Part Number 683–26007–3

SSP 41172B REQUIREMENT:

Paragraph 5.1.4.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Acceptance Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

The Emergency Egress Lights contain a Power Supply and Light Strip. The Power Supply is currently being redesigned and will be random vibration tested with Power On and Monitoring. This exception is for the light strips portion of the Emergency Egress Lights only.

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) Failures detected and Corrective actions implemented during the QTP/ATP process. During the Qualification Humidity Verification test, it was noted that a single LED was not lit. An examination of photographs showed that the light strip was bent in one direction during the bend radius test, and bent in the opposite direction when placed on the humidity qualification test fixture. Both light strips were replaced with flight hardware which had already passed acceptance testing. All qualification testing was done with the new light strips.

(2) The Emergency Egress Lights have a calculated MTBFs of 704,000 hours for those containing the 57–inch light strip and 1,111,600 hours for those containing the 36–inch light strip.

(3) The light strips are a series of LEDs soldered to a bus. This design is simple, rugged, and easily inspected.

(4) The Emergency Egress Lights are classified as criticality 1R.

(5) The ability to function on–orbit through a failure. Each exit has light strips containing from 130 to 214 LEDs wired in parallel. A failure of one or more LEDs would not prevent the light strips from illuminating the exit. Additionally, each light strip is being illuminated to 150 percent of the design requirement.

(6) The Emergency Egress Lights function may be restored on–orbit. The light strips are designed for replacement on–orbit. Spares have been purchased. A special tool and detailed instructions for replacement are currently available.

PG3–75:

ITEM:

General Luminaire Light Assembly Part Number 219003

SSP 41172B REQUIREMENT:

Paragraph 5.1.4.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Acceptance Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) Failures detected and Corrective actions implemented during the QTP/ATP process. During qualification and acceptance test of the General Luminaire Light Assembly LHAs and BBAs, the supplier experienced a total of six failures/anomalies during vibration and post–vibration testing. The failure and findings are as follows:

(a) Fasteners were discovered to be broken on qualification unit after vibration test because the vibration table surface was not counter–bored before securing BBA fixture to vibration table;

(b) BBA qualification unit failed to turn–on as a result of a broken wire to Q6 transistor wire. The cause was deemed as workmanship. This wire was rerouted and secured in place with epoxy. This change was incorporated into all flight units;

(c) LHA qualification unit failed to turn–on due to a broken wire to lamp filament as a result of supplier's outside test lab failure to properly ramp–up to required vibration 'g' level. LHA unit was subjected to a shock level test instead of a random vibration test;

(d) LHA and BBA flight units was exposed to higher vibration levels than required for acceptance test at 450 Hertz as a result of operator error. However, this did not exceed the qualification test levels for the LHA and BBA;

(e) LHA flight unit nut and washer came off inside LHA during vibration test. Proper torque was not applied on hardware and the cause was deemed as workmanship. No other failure occurred during vibration test; and

(f) BBA flight unit failed to turn–on after vibration test as a result of a transistor (Q1) failure. Q1 was removed and replaced and acceptance test was repeated with no additional failures.

(2) General Luminaire Light Assembly components have calculated MTBFs of 3,000,000 hours for the LHA and 1,929,000 hours for the BBA.

(3) The Design is rugged, durable, and easily inspected.

(4) The General Luminaire Light Assemblies are classified as criticality 2R.

(5) Function on–orbit including consequences of a failure. The General Luminaire Light Assemblies are redundant on–orbit (8 each in Node 1 and Italian Node 2, 12 in USL and Italian Node 3, and 4 in Airlock). A failure to any one General Luminaire Light Assembly will not cause a catastrophic failure to the remaining General Luminaire Light Assemblies. A simultaneous loss of all can only occur with the loss of station provided power (Power buses A and B) that supply the required 120 V dc input voltage.

(6) The General Luminaire Light Assembly function may be restored on–orbit. The General Luminaire Light Assembly is made of a LHA and a BBA. Both the LHA and BBA are identified as ORUs and are required to be removed and replaced in order to restore function on–orbit in the event of a failure.

PG3–76:

ITEM:

System On/Off Remote Control Part Number 219006–1

SSP 41172B REQUIREMENT:

Paragraph 5.1.4.4. A functional test shall be conducted before and after the completion of the random vibration test. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

EXCEPTION:

Acceptance Random Vibration Testing without Power On and monitoring is permitted.

RATIONALE:

A review of the manufacturing and screening processes for each hardware item was conducted based on performance within the following categories: (1) Additional screening, (2) Manufacturing, (3) Reliability, (4) Design/Function, and (5) Criticality. Results of this evaluation (as described below) indicate that the risk of not detecting a latent defect due to performing the random vibration without power and monitoring is low.

(1) Failures detected and Corrective actions implemented during the QT /ATP process. No failures were experienced during qualification and acceptance vibration test of the SRCAs.

(2) Qualification and Acceptance testing has been performed as described in SSP 41172 with the following approved exception:

CR 1052: The SRCAs was exempt from monitoring for Corona due to its low supply voltage (12 V dc). Documents AFAPL–TR–65–122 and 50M05189 indicate that corona will not occur at this voltage level.

(3) The SRCAs have a calculated MTBFs of 48,840,000 hours.

(4) The SRCAs are classified as criticality 3.

(5) Function on–orbit including consequences of a failure. The SRCAs provides a remote means for turning element lights on and off. In the event of a failure, this convenience would be lost and the light would be controlled at the General Luminaire Light Assemblies.

(6) The SRCA function may be restored on–orbit. The SRCAs are ORUs and are required to be removable and replaceable in order to restore function on–orbit in the event of a failure.

PG3–77:

ITEM:

Thermal Standoff Part Number 683–13580

SSP 41172 REQUIREMENT:

Paragraph 4.2: The component qualification tests that are a Program requirement are designated in Table 4–1 according to test category and component category.

Table 4–1: Component Qualification test matrix requires that Moving Mechanical Assemblies undergo thermal vacuum testing.

EXCEPTION:

Thermal Standoff Qualification will be performed by analysis of dimensions, tolerances, and materials supported by demonstration at thermal vacuum extremes.

RATIONALE:

The Thermal Standoff Assembly is a simple moving mechanical assembly of metal construction consisting of a housing and a spring plunger device. An analysis of the clearances and thermal coefficients demonstrate there will be no binding or interferences at temperature extremes. A material evaluation and tolerance analysis also demonstrates material and spring performance will not be affected by temperature extremes.

PG3–78:

ITEM:

Ready to Latch Assembly Part Number 683–13729

SSP 41172 REQUIREMENT:

Paragraph 4.2.5.3: Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 4.2.5.3B: Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The same Power Spectral Density was used for Qualification and Acceptance. Therefore, Qualification levels did not envelope Acceptance levels plus test tolerances.

RATIONALE:

During Qualification testing, the Ready to Latch Assembly (Serial Number 0000001) was vibration tested for a total of 9 minutes in each axis (6 minutes Qualification and 3 minutes Acceptance). Post vibration functional test noted no discrepancies. An inspection of the test article showed no signs of chatter or clearance interference. The flight articles are acceptance tested to the same level but have only seen three minutes in each axis. The Qualification tests exposed the test article to 13.64 grms in the Z–axis and 9.5 grms in the X– and Y–axis. The predicted flight environment is 11.9 grms and 3 grms respectively. Since Acceptance units were vibration tested to the same level as qualification, no amplitude margin exists. The duration of applied stress, though, effectively screens the Ready to Latch Assembly for failures due to applied fatigue with acceptable qualification margin. Twenty Ready To Latch flight units have been acceptance tested to date with no evidence of chatter or clearance problems noted.

PG3–79:

ITEM:

Powered Bolt Nut Assembly Part Number 683–13503

SSP 41172 REQUIREMENT:

Paragraph 4.2.5.3: Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 4.2.5.3B: Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The same Power Spectral Density was used for Qualification and Acceptance. Therefore, Qualification levels did not envelope Acceptance levels plus test tolerances.

RATIONALE:

The Qualification Powered Bolt Nut Assembly was random vibration tested for 1 minute during acceptance and 13.5 minutes for qualification (total 14.5 minutes).

Since Qualification and Acceptance Tests included vibration testing to the same level and no amplitude margin exists, the duration of applied stress effectively screens the Powered Bolt Nut Assemblies for failures due to fatigue with acceptable qualification margin. In addition, any anomaly that had occurred during random vibration would have been detectable via post test visual inspection or post test functional testing.

The Powered Bolt Nut Assembly is a simple mechanical assembly that is easily inspected. In the case of Power Bolt Nuts that are not returned to earth and inspected (PMA3), these are limited to six cycles. The Powered Bolt Nut Assembly uses a cotter pinned castle nut to retain integrity of the assembly. This retention mechanism will prevent vibration from loosening the assembly. Thus far, 202 Powered Bolt Nut Assembly flight units have been acceptance tested to date with no evidence of chatter or clearance problems noted.

PG3–80:

ITEM:

Capture Latch Assembly Part Number 683–13434–6

SSP 41172 REQUIREMENT:

Paragraph 4.2.5.3: Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 4.2.5.3B: Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The same Power Spectral Density was used for Qualification and Acceptance. Therefore, Qualification levels did not envelope Acceptance levels plus test tolerances.

RATIONALE:

The Capture Latch Assembly delta qualification unit (Part Number 683–13434–2, Serial Number 000002) was subjected to random vibration acceptance testing three times at 1 minute per axis (3 minutes per axis total), and random vibration qualification testing once, for one minutes per axis, with a flight actuator and once, for four minutes per axis, with a mass simulator. The total vibration time on the qualification unit was eight minutes per axis. 20 Capture Latch Assemblies have been acceptance tested to date with no chatter or clearance problems noted.

Since Qualification and Acceptance Tests included vibration testing to the same level, and no amplitude margin exists, the eight minute total duration of applied stress effectively screens the Capture Latch Assembly for failures due to fatigue with acceptable qualification margin.

PG3–81:

ITEM:

Deployable Meteoroid Debris Mechanism Part Number 683–14599–3

SSP 41172 REQUIREMENT:

Paragraph 4.1: Qualification test levels and duration shall in all cases envelope worst–case service life environments including acceptance test levels and duration (including test tolerances) and accommodate acceptance retesting.

EXCEPTION:

The same Power Spectral Density was used for Qualification and Acceptance Acoustic Vibration Testing. Therefore, Qualification levels did not envelope Acceptance levels plus test tolerances.

RATIONALE:

The Deployable Meteoroid Debris Mechanism Qualification Unit (Serial Number 000001) was subjected to acoustic vibration acceptance testing for 1 minute, subjected to acoustic vibration qualification testing once for 3 minutes, and subjected to a delta acoustic vibration requalification for 3 additional minutes. The total acoustic vibration test time on the qualification unit is 7 minutes acoustic vibration at 141 dB overall sound pressure level.

Since Qualification and Acceptance Tests included vibration testing to the same level and no amplitude margin exists, the seven minute total duration of applied stress effectively screens the Deployable Meteoroid Debris Cover for failures due to fatigue.

PG3–82:

ITEM:

Powered Bolt Assembly   Part Number 683–13450
Powered Bolt Nut Assembly   Part Number 683–13503

SSP 41172 REQUIREMENT:

Paragraph 4.2.2.3C, Qualification Thermal Vacuum, Test Levels and Duration: A minimum of three temperature cycles shall be used.

EXCEPTION:

One qualification thermal vacuum cycle will be performed from the required hot temperature extremes (170 degrees F nonoperating, 150 degrees F operating) to the required cold temperature extreme (–50 degrees F nonoperating and operating).

RATIONALE:

The purpose of three qualification cycles is to qualify the hardware for two additional acceptance thermal vacuum tests. The Powered Bolt Assembly and Powered Bolt Nut Assembly does not contain any internal soft goods or other materials sensitive to thermal extreme cycling and the operation of these items for one cycle at the thermal extremes demonstrates a lack of mechanical interferences. Additionally, the lubricant performance data indicates that its worst case is in the ambient temperature ranges eliminating lubricant as a driver for the thermal extremes.

Additionally, testing as listed has been performed on the Common Berthing Mechanism components.

A. 69 Assembly Level Qualification Test (ALQT) functional cycles at varying temperature ranges under vacuum, including one ALQT thermal extreme cycle [Active Common Berthing Mechanism (ACBM) flange temperature –50 degrees F to 150 degrees F (Bolt) and Passive Common Berthing Mechanism (PCBM) flange temperature -90 degrees F to 190 degrees F (Nut)].

B. Five Functional Cycles (75 bolts) have been conducted with the ACBM Flange temperature at or above 120 degrees F. The qualification unit met all objectives with no failures.

CBM Assembly Qualification Test Powered Bolt Cycles	Active Ring Temp/Passive Ring Temp
15 Bolts achieved preload	120 degrees F/119 degrees F
14 Bolts achieved preload	120 degrees F/70 degrees F
16 Bolts achieved preload	150 degrees F/190 degrees F
16 Bolts achieved preload	130 degrees F/170 degrees F
14 Bolts achieved preload	130 degrees F/–70 degrees F

C. Six Functional Cycles (88 bolts) have been conducted with the ACBM Flange temperature at or below –10 degrees F including four functional cycles (59 bolts) at or below -30 degrees F. The qualification unit met all objectives with no failures.

CBM Assembly Qualification Test Powered Bolt Cycles	Active Ring Temp/Passive Ring Temp
15 Bolts achieved preload	–23 degrees F/84 degrees F
14 Bolts achieved preload	–30 degrees F/–67 degrees F
14 Bolts achieved preload	–10 degrees F/90 degrees F
16 Bolts achieved preload	–50 degrees F/–90 degrees F
16 Bolts achieved preload	–30 degrees F/170 degrees F
13 Bolts achieved preload	–30 degrees F/–70 degrees F

Analytical predictions as well as the as measured temperature data from ALQT Powered Bolts show that the powered bolts achieve extreme temperatures approximating that of the Flange and this component has not shown sensitivity to repeated thermal use cycling as demonstrated in ALQT. Therefore, it is considered minimal risk to re–thermal vacuum acceptance test power bolts without having demonstrated cycle capability during qualification. Thus, one qualification thermal vacuum cycle is adequate due to the rationale and the additional testing described.

PG3–83:

ITEM:

Powered Bolt Assembly Part Number 683–13450

SSP 41172 REQUIREMENT:

Paragraph 5.1.2.2, Acceptance Thermal Vacuum Test, Test Description: ALL REQUIREMENTS

EXCEPTION:

An Acceptance Thermal Vacuum test will not be performed following the installation of the trim resistor for calibration of the load cell. This exception allows the use of the Powered Bolt Assembly without the execution of this test on the fully assembled unit.

RATIONALE:

Thermal vacuum is performed on the Powered Bolt Assembly prior to installation of the trim resistor required to trim the load cell. The resistor is soldered into the circuit and is easily inspected. The intent of the requirement is to screen and stress more complex assemblies of primarily electronic or electrical assemblies with vacuum sensitive and/or thermally sensitive installations such as conformal coatings, many solder joined surfaces, or bonding applications. There have been no failures of the load cell with trim resistors through 69 thermal vacuum cycles that have been performed during the Assembly Level Qualification Test.

Additionally, testing as listed has been performed on the ACBM components.

A. Five Functional Cycles (75 bolts) have been conducted with the ACBM Flange temperature at or above 120 degrees F.

CBM Assembly Qualification Test Powered Bolt Cycles	Active Ring Temp/Passive Ring Temp
15 Bolts achieved preload	120 degrees F/119 degrees F
14 Bolts achieved preload	120 degrees F/70 degrees F
16 Bolts achieved preload	150 degrees F/190 degrees F
16 Bolts achieved preload	130 degrees F/170 degrees F
14 Bolts achieved preload	130 degrees F/–70 degrees F

B. Six Functional Cycles (88 bolts) have been conducted with the ACBM Flange temperature at or below –10 degrees F including four functional cycles (59 bolts) at or below -30 degrees F.

CBM Assembly Qualification Test Powered Bolt Cycles	Active Ring Temp/Passive Ring Temp
15 Bolts achieved preload	–23 degrees F/84 degrees F
14 Bolts achieved preload	–30 degrees F/–67 degrees F
14 Bolts achieved preload	–10 degrees F/90 degrees F
16 Bolts achieved preload	–50 degrees F/–90 degrees F
16 Bolts achieved preload	–30 degrees F/170 degrees F
13 Bolts achieved preload	–30 degrees F/–70 degrees F

Analytical predictions as well as the measured temperature data from ALQT Powered Bolts show that the Powered Bolts achieve extreme temperatures approximating that of the Flange. Thus, an acceptance thermal vacuum cycle on the Powered Bolts without incorporation of the trim resistor is adequate.

PG3–84:

ITEM:

Powered Bolt Assembly Part Number 683–13450 Serial Numbers 001–165, 169–170, 172–174, 176–189, 191–194, 209–218, and 248–252

SSP 41172 REQUIREMENT:

Paragraph 4.2.2.3B, Qualification Thermal Vacuum, Test Levels and Duration: The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The designated Powered Bolt Assemblies are permitted to be Acceptance Thermal Vacuum tested within 10 degrees of the cold portion of the Qualification Thermal Vacuum test.

RATIONALE:

The Acceptance Thermal Vacuum testing for the designated Powered Bolts was performed over a range of -40 degrees F to 130 degrees F. The Qualification Thermal Vacuum Testing was performed over a range of -50 degrees F to 150 degrees F. For the designated Powered Bolt Assemblies above, the Qualification test limits should have been -60 degrees F to 150 degrees F to maintain the specified 20 degrees F margin.

Extensive thermal testing during both qualification and acceptance of the Powered Bolt Assemblies indicates that the power bolts are not temperature sensitive. Additionally, there have been no temperature–related failures. Therefore, the 10 degrees F margin for the cold portion of the test is considered to be a minimal risk.

The Acceptance Test Procedure will be modified to test from -30 degrees F to 130 degrees F. This change provides the required 20 degrees F qualification margin and envelopes the minimum and maximum predicted on–orbit operating temperatures (–30 degrees F to 120 degrees F). Therefore, these worst–case service environments will be enveloped in the modified acceptance test.

PG3–85:

ITEM:

Powered Bolt Assembly Part Number 683–13450 Serial Numbers 001–165, 169–170, 172–174, 176–189, 191–194, 209–218, and 248–252

SSP 41172 REQUIREMENT:

Paragraph 5.1.3.3B, Acceptance Thermal Cycling, Test Levels and Duration: The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

The designated Powered Bolt Assemblies were subjected to a temperature sweep of 0 degrees F to 100 degrees F which did not encompass the cold and hot acceptance limits of -40 degrees F and 130 degrees F as established by its acceptance test procedure.

RATIONALE:

The designated Powered Bolt Assemblies, except for the trim resistor, has been exposed to one temperature cycle of –40 degrees F to 130 degrees F during Thermal Vacuum prior to adding the trim resistor. The trim resistor is soldered into the circuit and is easily inspected.

The purpose of enveloping the minimum and maximum predicted flight temperatures with the 100 degrees F minimum workmanship sweep for electronics is to demonstrate that each flight unit will operate at its minimum and maximum predicted level prior to flight. This has been demonstrated for the Power Bolt Assemblies, with the exception of the trim resistor installation, during Thermal Vacuum Testing. Not exposing the trim resistor and its solder joints to the minimum and maximum predicted on–orbit temperature extremes is considered minimal risk.

The Acceptance Test Procedure will be modified to test from -30 degrees F to 130 degrees F. This change provides the required 20 degrees F qualification margin and envelopes the minimum and maximum predicted on–orbit operating temperatures (–30 degrees F to 120 degrees F). Therefore, these worst–case service environments will be enveloped in the modified acceptance test.

PG3–86:

ITEM:

SPDA Part Number 683–27000

SSP 41172 REQUIREMENT:

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittent performance during the test.

EXCEPTION:

The fully assembled SPDA is exempt from power–on and monitoring during Qualification Random Vibration testing.

RATIONALE:

The SPDA is a mechanical housing for mounting RPCMs in the USL endcones. All RPCMs internal to the SPDAs undergo individual ORU–level Qualification Random Vibration testing. The only additional electrical/electronic components in the SPDA other than RPCMs are wires and connectors. These harnesses are subjected to continuity and isolation testing, insulation resistance testing at 500 Vdc, and dielectric withstanding voltage testing at 1500 Vdc in accordance with NHB 5300.4. These tests are performed post–assembly and post–installation onto the SPDA. The SPDA does undergo random vibration testing with the RPCMs installed. These RPCMs were tested and operated successfully both pre– and post–vibration.

PG3–87:

ITEM:

SPDA Part Number 683–27000

SSP 41172 REQUIREMENT:

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test. Parameters shall be monitored for failures or intermittences during the test.

EXCEPTION:

The fully assembled SPDA is exempt from power–on and monitoring during Acceptance Random Vibration testing.

RATIONALE:

The SPDA is a mechanical housing for mounting RPCMs in the USL endcones. All RPCMs internal to the SPDAs undergo individual ORU–level Acceptance Random Vibration testing. The only additional electrical/electronic components in the SPDA other than RPCMs are wires and connectors. These harnesses are subjected to continuity and isolation testing, insulation resistance testing at 500 Vdc, and dielectric withstanding voltage testing at 1500 Vdc in accordance with NHB 5300.4. These tests are performed post–assembly and post–installation onto the SPDA. The SPDA does undergo random vibration testing with the RPCMs installed. These RPCMs were tested and operated successfully both pre– and post–vibration.

PG3–88:

ITEMS:

Audio Bus Coupler (ABC) Part Number 3000005–301
Video Switch Unit (VSU) Part Number 3000008–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification.

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

The ABC and VSU shall not be electrically energized and monitored during the qualification random vibration test.

RATIONALE:

Functional testing was performed on these components at the completion of the acceptance random vibration testing which provides some workmanship screening. The ABC and VSU not being electrically energized and monitored poses little performance risk as the hardware will not be powered on and functioned during the flight vibration environments. In both cases, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–89:

ITEMS:

Audio Bus Coupler (ABC) Part Number 3000005–301
Video Switch Unit (VSU) Part Number 3000008–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance.

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

The ABC and VSU shall not be electrically energized and monitored during the acceptance random vibration test.

RATIONALE:

Functional testing was performed on these components at the completion of the acceptance random vibration testing which provides some workmanship screening. The ABC and VSU not being electrically energized and monitored poses little performance risk as the hardware will not be powered on and functioned during the flight vibration environments. In both cases, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–90:

ITEMS:

Audio Bus Coupler (ABC) Part Number 3000005–301
Video Switch Unit (VSU) Part Number 3000008–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification. Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 4.2.5.3B, Test Levels and Duration. Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The Qualification Random Vibration Test of the ABC and VSU shall be performed at a level of 4.4 grms with a maximum spectral density of 0.04 G^2/Hz.

RATIONALE:

The Qualification Random Vibration environment enveloped the maximum predicted flight level of 4.3 grms for US–located uses. The ABC and the VSU were designed and analyzed to survive a 4.4 grms environment. The contractor Harris used conservative design practices including designing for stiffness with self–imposed safety factors for deflection and fatigue. All fasteners have locking features or are staked, and are inspected during assembly. Additionally, all circuit assemblies are conformal coated which precludes malfunctions due to shorting from conductive hardware loose in the unit. In both cases, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–91:

ITEMS:

Audio Bus Coupler (ABC) Part Number 3000005–301
Video Switch Unit (VSU) Part Number 3000008–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance. Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 5.1.4.3B, Test Levels and Duration. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime contractor.

EXCEPTION:

The Acceptance Random Vibration Test of the ABC and VSU shall be performed at a level of 3.1 grms with a maximum spectral density of 0.02 G^2/Hz.

RATIONALE:

The acceptance random vibration testing performed by the contractor Harris does not meet the required workmanship screening levels in Figure 5–2, yet it does provide value as a workmanship screen. The ABC and the VSU were designed and analyzed to survive a 4.4 grms environment, beyond the maximum predicted flight level of 4.3 grms for US–located uses. Harris used conservative design practices including designing for stiffness with self–imposed safety factors for deflection and fatigue. All fasteners have locking features or are staked, and are inspected during assembly. Additionally, all circuit assemblies are conformal coated which precludes malfunctions due to shorting from conductive hardware loose in the unit. In both cases, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–92:

ITEMS:

Audio Bus Coupler (ABC) Part Number 3000005–301
Video Switch Unit (VSU) Part Number 3000008–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

The ABC and VSU shall not be subjected to their maximum predicted non–operating temperature of 125 degrees F during acceptance thermal cycle testing.

RATIONALE:

The maximum temperature for the ABC and VSU during its life cycle is derived under ferry flight conditions. This predicted maximum non–operation temperature under these conditions is 125 degrees F. However, the actual operating environments of these ORUs are benign as they are within pressurized volumes and are coldplate–mounted which controls the operating environment experienced. The worst–case predicted maximum temperatures only occur during depot return activities or in the event of an aborted launch attempt with a contingency site landing.

The Acceptance Thermal Cycle testing was performed at the 90–degree coldplate temperature during the hot portion of the thermal cycle test. Analysis indicates that during the hot portion, operational testing resulted in internal EEE part case temperatures from 126 degrees F to 178 degrees F. Thus, internal electrical components were tested at temperatures that exceeded the required acceptance non–operating limit. In both cases, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–93:

ITEMS:

Audio Bus Coupler (ABC) Part Number 3000005–301
Video Switch Unit (VSU) Part Number 3000008–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.

Paragraph 4.2.3.3B, Test Levels and Duration. For electrical/electronic equipment where the minimum sweep (para. 5.1.3.3–2) does not encompass by the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F.

EXCEPTION:

The ABC and VSU shall experience a 97 degrees F sweep between the minimum and maximum coldplate–mounted qualification thermal cycle operating temperatures.

RATIONALE:

The coldplate–mounted operational temperatures during qualification thermal cycle testing were from 13 degrees F to 110 degrees F. This envelops the predicted operational temperatures of the ORUs with required margin of 20 degrees F. Additionally, non–operational acceptance thermal cycle testing from –30 degrees F to 110 degrees F provides a non–operating temperature sweep in compliance with the requirement. In both cases, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–94:

ITEMS:

Audio Bus Coupler (ABC) Part Number 3000005–301
Video Switch Unit (VSU) Part Number 3000008–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3B, Test Levels and Duration. For components identified in Table 5–1, with note 4, there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The ABC and VSU shall experience a 57 degrees F sweep between the minimum and maximum coldplate–mounted acceptance thermal cycle operating temperatures.

The minimum coldplate–mounted test temperature shall be 33 degrees F.

RATIONALE:

The coldplate–mounted operational temperatures during acceptance thermal cycle testing were from 33 degrees F to 90 degrees F. This envelops the predicted operational temperatures of the ORUs. Additionally, non–operational acceptance thermal cycle testing from –10 degrees F to 90 degrees F provides a non–operating temperature sweep in compliance with the requirement. In both cases, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–95:

ITEM:

Audio Terminal Unit Part Number 3000001–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification.

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

The Audio Terminal Unit shall not be electrically energized and monitored during the qualification random vibration test.

RATIONALE:

Functional testing was performed at the completion of the acceptance random vibration testing which provides some workmanship screening. The Audio Terminal Unit not being electrically energized and monitored poses little performance risk as the hardware will not be powered on and functioned during the flight vibration environments. In the event of failure due to undetected workmanship defects, multiple units are installed (including one on each end cone of the USL) to provide functional redundancy and an on–board spare is available in Avionics Rack 3. In addition, extra crew communication headset extension cables will be flown to accommodate relocating functionally to a different Audio Terminal Unit should one fail. Thus, the additional risk associated with this exception is limited.

PG3–96:

ITEM:

Audio Terminal Unit Part Number 3000001–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance.

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

The Audio Terminal Unit shall not be electrically energized and monitored during the acceptance random vibration test.

RATIONALE:

Functional testing was performed at the completion of the acceptance random vibration testing which provides some workmanship screening. The Audio Terminal Unit not being electrically energized and monitored poses little performance risk as the hardware will not be powered on and functioned during the flight vibration environments. In the event of failure due to undetected workmanship defects, multiple units are installed (including one on each end cone of the USL) to provide functional redundancy and an on–board spare is available in Avionics Rack 3. In addition, extra crew communication headset extension cables will be flown to accommodate relocating functionally to a different Audio Terminal Unit should one fail. Thus, the additional risk associated with this exception is limited.

PG3–97:

ITEM:

Audio Terminal Unit Part Number 3000001–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification. Component random vibration test levels and spectrums shall be the envelope of the following.

Paragraph 4.2.5.3B, Test Levels and Duration. Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The Qualification Random Vibration Test of the Audio Terminal Unit shall be performed at a level of 4.4 grms with a maximum spectral density of 0.04 G^2/Hz.

RATIONALE:

The Qualification Random Vibration environment enveloped the maximum predicted flight level of 4.3 grms for US–located uses. The Audio Terminal Unit was designed and analyzed to survive a 4.4 grms environment. The Audio Terminal Unit design complexity is low as it contains four circuit–card assemblies, a power supply module, and several panel–mounted components. The contractor Harris used conservative design practices including designing for stiffness with self–imposed safety factors for deflection and fatigue. All fasteners have locking features or are staked, and are inspected during assembly. Additionally, all circuit assemblies are conformal coated which precludes malfunctions due to shorting from conductive hardware loose in the unit. In the event of failure due to undetected workmanship defects, multiple units are installed (including one on each end cone of the US Laboratory) to provide functional redundancy and an on–board spare is available in Avionics Rack 3. In addition, extra crew communication headset extension cables will be flown to accommodate relocating functionally to a different Audio Terminal Unit should one fail. Thus, the additional risk associated with this exception is limited.

PG3–98:

ITEM:

Audio Terminal Unit Part Number 3000001–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance. Component random vibration test levels and spectrums shall be the envelope of the following.

Paragraph 5.1.4.3B, Test Levels and Duration. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime contractor.

EXCEPTION:

The Acceptance Random Vibration Test of the Audio Terminal Unit shall be performed at a level of 3.1 grms with a maximum spectral density of 0.02 G^2/Hz.

RATIONALE:

The acceptance random vibration testing performed by the contractor Harris does not meet the required workmanship screening levels in Figure 5–2, yet it does provide value as a workmanship screen. The Audio Terminal Unit was designed and analyzed to survive a 4.4 grms environment, beyond the maximum predicted flight level of 4.3 grms for US–located uses. The Audio Terminal Unit design complexity is low as it contains four circuit–card assemblies, a power supply module, and several panel–mounted components. Harris used conservative design practices including designing for stiffness with self–imposed safety factors for deflection and fatigue. All fasteners have locking features or are staked, and are inspected during assembly. Additionally, all circuit assemblies are conformal coated which precludes malfunctions due to shorting from conductive hardware loose in the unit. In the event of failure due to undetected workmanship defects, multiple units are installed (including one on each end cone of the USL) to provide functional redundancy and an on–board spare is available in Avionics Rack 3. In addition, extra crew communication headset extension cables will be flown to accommodate relocating functionally to a different Audio Terminal Unit should one fail. Thus, the additional risk associated with this exception is limited.

PG3–99:

ITEM:

Audio Terminal Unit Part Number 3000001–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

The Audio Terminal Unit shall not be subjected to their maximum predicted non–operating temperature of 125 degrees F during acceptance thermal cycle testing.

RATIONALE:

The maximum temperature for the Audio Terminal Unit during its life cycle is derived under ferry flight conditions. This predicted maximum non–operation temperature under these conditions is 125 degrees F. However, the actual operating environments of this ORU is benign as it is within pressurized volumes and is coldplate–mounted which controls the operating environment experienced. The worst–case predicted maximum temperatures only occur during depot return activities or in the event of an aborted launch attempt with a contingency site landing.

The Acceptance Thermal Cycle testing was performed at the 90–degree coldplate temperature during the hot portion of the thermal cycle test. Analysis indicates that during the hot portion, operational testing resulted in internal EEE part case temperatures from 126 degrees F to 178 degrees F. Thus, internal electrical components were tested at temperatures that exceeded the required acceptance non–operating limit. In the event of failure due to undetected workmanship defects, multiple units are installed (including one on each end cone of the USL) to provide functional redundancy and an on–board spare is available in Avionics Rack 3. In addition, extra crew communication headset extension cables will be flown to accommodate relocating functionally to a different Audio Terminal Unit should one fail. Thus, the additional risk associated with this exception is limited.

PG3–100:

ITEM:

Audio Terminal Unit Part Number 3000001–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.

Paragraph 4.2.3.3B, Test Levels and Duration. For electrical/electronic equipment where the minimum sweep (para. 5.1.3.3–2) does not encompass by the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F.

EXCEPTION:

The Audio Terminal Unit shall experience a 98 degrees F sweep between the minimum and maximum coldplate–mounted qualification thermal cycle operating temperatures.

RATIONALE:

The coldplate–mounted operational temperatures during qualification thermal cycle testing were from 12 degrees F to 110 degrees F. This envelops the predicted operational temperatures of the ORU with required margin of 20 degrees F. Additionally, non–operational acceptance thermal cycle testing from –30 degrees F to 110 degrees F provides a non–operating temperature sweep in compliance with the requirement. In the event of failure due to undetected workmanship defects, multiple units are installed (including one on each end cone of the USL) to provide functional redundancy and an on–board spare is available in Avionics Rack 3. In addition, extra crew communication headset extension cables will be flown to accommodate relocating functionally to a different Audio Terminal Unit should one fail. Thus, the additional risk associated with this exception is limited.

PG3–101:

ITEM:

Audio Terminal Unit Part Number 3000001–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3B, Test Levels and Duration. For components identified in Table 5–1, with note 4, there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The Audio Terminal Unit shall experience a 58 degrees F sweep between the minimum and maximum coldplate–mounted acceptance thermal cycle operating temperatures.

The minimum coldplate–mounted test temperature shall be 32 degrees F.

RATIONALE:

The coldplate–mounted operational temperatures during acceptance thermal cycle testing were from 32 degrees F to 90 degrees F. This envelops the predicted operational temperatures of the ORUs. Additionally, non–operational acceptance thermal cycle testing from –10 degrees F to 90 degrees F provides a non–operating temperature sweep in compliance with the requirement. In the event of failure due to undetected workmanship defects, multiple units are installed (including one on each end cone of the USL) to provide functional redundancy and an on–board spare is available in Avionics Rack 3. In addition, extra crew communication headset extension cables will be flown to accommodate relocating functionally to a different Audio Terminal Unit should one fail. Thus, the additional risk associated with this exception is limited.

PG3–102:

ITEM:

MBM Manual Bolt Assembly Part Number 683–55217–003

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.

Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Thermal Vacuum Test was not performed on the 16 MBM Manual Bolt Assemblies used on MBM–1.

RATIONALE:

The MBM Manual Bolt Assemblies will be flight accepted based on CBM Powered Bolt Acceptance Thermal Vacuum testing.

The CBM Powered Bolts undergo acceptance thermal vacuum testing at a pressure of 1E–04 Torr and temperatures from 0 to +100 degrees F for 24 Operational Cycles. Extensive acceptance thermal testing of the CBM Powered Bolts (approximately 200 bolts tested to date) demonstrate that the powered bolts are not temperature sensitive, as there have been no temperature–related failures. An MBM Manual Bolt have the following parts not residing in a CBM Powered Bolt: Manual Bolt Drive (Part Number 683–55217–002), 2 Washers (Part Number 683–55220–004), 1 Screw (Part Number NAS1351N3–8), 2 Screws (Part Number NAS1351N5–14), and a Cushion Clamp (Part Number NAS1715CT16IW). These changes are mechanical only and, thus, do not impact the validity of the application of the results of the CBM Powered Bolt thermal vacuum test.

A thermal analysis of the Manual Bolt Assembly was conducted to ensure part clearances were maintained at the MBM operational temperature extremes. The analysis considered rotational clearances due to size variations, axial clearances due to size variations, and position variations. The analysis determined that sufficient clearances are maintained at the temperature extremes, and the Manual Bolt Assembly will function properly at those temperatures.

PG3–103:

ITEM:

Manual Berthing Mechanism (MBM–1 and MBM–2) Part Numbers 683–55226–003 and 683–55226–004

SSP 41172 REQUIREMENT:

Paragraph 6.1.1 Assembly / Components Protoflight Tests.

When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification) with the following exceptions:

C. For the acoustic vibration qualification test, the test level shall be the maximum predicted flight level, but not less than a level derived from an acoustic environment of 141 dB overall (whose spectrum is defined by NSTS 21000–IDD–ISS, paragraph 4.1.1.5). The duration of the test shall be limited to 1 minute.

EXCEPTION:

The duration of the Protoflight Vibration Test for MBM–1 and MBM–2 was 4 minutes 20 seconds.

RATIONALE:

Based on previous qualification testing and supporting analysis, significant fatigue life remains post–acceptance for all MBM components. The MBM design borrows several features from the qualified Common Berthing Mechanism (CBM) which were random vibration tested to qualification levels in excess of the predicted MBM flight environment. Two moving mechanical assemblies developed in–house for CBM were adapted for use on MBM; the capture latch and the powered bolt. A number of tests have been run during development, qualification, and acceptance of CBM bolt and latch hardware which have demonstrated no susceptibility to fatigue damage from vibration sensitivity. In short, there is no evidence of fatigue criticality or significant vibration–induced stress from the CBM qualification environment. Therefore, there are no fatigue concerns for these parts in the MBM protoflight environment, even at 4 minutes 20 seconds test duration.

The remainder of the MBM structure and mechanical components (the spider web cross beams, drive and latch tie rods, center crank, and drive screw) were also thoroughly addressed in the MBM Fatigue and Fracture Analysis (D683–34973, MBM Structural Analysis, Section 7). Based on structural analysis of the equipment exposed to flight–level cyclic loading during the protoflight vibroacoustic test, the following conclusions were drawn: Four minutes 20 seconds of exposure (more than 3 minutes over the requirement) did not significantly degrade the fatigue life of MBM hardware; and MBM hardware stresses affected by the test are not enough to be significant for fracture or fatigue.

PG3–104:

ITEM:

Common Video Interface Unit (CVIU) Part Number 3000028–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

The CVIU shall not be electrically energized and monitored during the qualification random vibration test.

RATIONALE:

Functional testing was performed on these components at the completion of the acceptance random vibration testing which provides some workmanship screening. The CVIU not being electrically energized and monitored poses little performance risk as the hardware will not be powered on and functioned during the flight vibration environments. With the CVIU, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–105:

ITEM:

Common Video Interface Unit Part Number 3000028–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

The CVIU shall not be electrically energized and monitored during the acceptance random vibration test.

RATIONALE:

Functional testing was performed on these components at the completion of the acceptance random vibration testing which provides some workmanship screening. The CVIU not being electrically energized and monitored poses little performance risk as the hardware will not be powered on and functioned during the flight vibration environments. With the CVIU, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–106:

ITEM:

Common Video Interface Unit Part Number 3000028–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification. Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 4.2.5.3B, Test Levels and Duration. Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The Qualification Random Vibration Test of the CVIU shall be performed at a level of 4.4 grms with a maximum spectral density of 0.04 G^2/Hz.

RATIONALE:

The Qualification Random Vibration environment enveloped the maximum predicted flight level of 4.3 grms for US–located uses. The CVIU was designed and analyzed to survive a 8.6 grms environment. The contractor Harris used conservative design practices including designing for stiffness with self–imposed safety factors for deflection and fatigue. All fasteners have locking features or are staked, and are inspected during assembly. Additionally, all circuit assemblies are conformal coated which precludes malfunctions due to shorting from conductive hardware loose in the unit. With the CVIU, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–107:

ITEM:

Common Video Interface Unit Part Number 3000028–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance. Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 5.1.4.3B, Test Levels and Duration. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime contractor.

EXCEPTION:

The Acceptance Random Vibration Test of the CVIU shall be performed at a level of 3.1 grms with a maximum spectral density of 0.02 G^2/Hz.

RATIONALE:

The acceptance random vibration testing performed by the contractor Harris does not meet the required workmanship screening levels in Figure 5–2, yet it does provide value as a workmanship screen. The CVIUs were designed and analyzed to survive a 4.4 grms environment, beyond the maximum predicted flight level of 4.3 grms for US–located uses. Harris used conservative design practices including designing for stiffness with self–imposed safety factors for deflection and fatigue. All fasteners have locking features or are staked, and are inspected during assembly. Additionally, all circuit assemblies are conformal coated which precludes malfunctions due to shorting from conductive hardware loose in the unit. In both cases, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–108:

ITEM:

Common Video Interface Unit Part Number 3000028–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

The CVIU shall not be subjected to their maximum predicted non–operating temperature of 125 degrees F during acceptance thermal cycle testing.

RATIONALE:

The maximum temperature for the CVIU during its life cycle is derived under ferry flight conditions. This predicted maximum non–operation temperature under these conditions is 125 degrees F. However, the actual operating environments of the CVIU is benign as they are within pressurized volumes and are coldplate–mounted which controls the operating environment experienced. The worst–case predicted maximum temperatures only occur during depot return activities or in the event of an aborted launch attempt with a contingency site landing.

The Acceptance Thermal Cycle testing was performed at the 90–degree coldplate temperature during the hot portion of the thermal cycle test. Analysis indicates that during the hot portion, operational testing resulted in internal EEE part case temperatures from 126 degrees F to 178 degrees F. Thus, internal electrical components were tested at temperatures that exceeded the required acceptance non–operating limit. In addition, all parts and materials in the CVIU are certified beyond non–operational limit. With the CVIU, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–109:

ITEM:

Common Video Interface Unit Part Number 3000028–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.

Paragraph 4.2.3.3B, Test Levels and Duration. For electrical/electronic equipment where the minimum sweep (para. 5.1.3.3–2) does not encompass by the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F.

EXCEPTION:

The CVIU shall experience a 97 degrees F sweep between the minimum and maximum coldplate–mounted qualification thermal cycle operating temperatures.

RATIONALE:

The coldplate–mounted operational temperatures during qualification thermal cycle testing were from 13 degrees F to 110 degrees F. This envelops the predicted operational temperatures of the ORUs with required margin of 20 degrees F. Additionally, non–operational qualification thermal cycle testing from –30 degrees F to 110 degrees F provides a non–operating temperature sweep in compliance with the requirement. With the CVIU, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–110:

ITEM:

Common Video Interface Unit Part Number 3000028–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3B, Test Levels and Duration. For components identified in Table 5–1, with note 4, there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The CVIU shall experience a 57 degrees F sweep between the minimum and maximum coldplate–mounted qualification thermal cycle operating temperatures.

The minimum coldplate–mounted test temperature shall be 33 degrees F.

RATIONALE:

The coldplate–mounted operational temperatures during acceptance thermal cycle testing were from 33 degrees F to 90 degrees F. This envelops the predicted operational temperatures of the ORUs. Additionally, non–operational acceptance thermal cycle testing from –10 degrees F to 90 degrees F provides a non–operating temperature sweep in compliance with the requirement. With the CVIU, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–111:

ITEM:

Avionics Air Assembly Part Number SV809992

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.

Paragraph 5.1.4.3 Test levels and Duration. ALL REQUIREMENTS.

Paragraph 5.1.4.4 Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Random Vibration Test will not be performed at the AAA assembly level.

RATIONALE:

The Avionics Air Assembly consists of a Heat Exchanger, Acoustic Cover/Inlet Muffler Assembly, Outlet Muffler, Fan Assembly containing a Brushless DC Motor with a mounted Hall Effects Device board, and a Controller and Filter Board containing electronics. The mechanical assemblies in the AAA are not considered to have close tolerances requiring precise adjustment or that cannot be inspected properly, and therefore would not require acceptance random vibration test. Mechanical subassembly workmanship tests do include Heat Exchanger proof and helium leakage test.

The Controller and EMI filter boards experience an acceptance random vibration level of 7.0 grms while powered and monitored in accordance with SSP 41172. The simple Hall Effects Device board (mounted on the Brushless DC Motor) is a single–layer board with four mounted active parts, has lap solder connections, and can be fully inspected after soldering. Additionally, the board is subjected to electrical bonding, thermal cycling, 300–hour burn–in, and performance testing during subassembly test. The Brushless DC Motor has air bearings designed to work in a 0 or 1G environment, not in a vibration condition. The Motor is inspected during build and subjected to electrical continuity, dielectric withstanding voltage, isolation resistance, power cycle, and run–in workmanship testing.

The Fan Assembly, with both the Brushless DC Motor and the HED board, is screened via 89 separate inspections during buildup. It is then subjected to electrical bonding and an 8–hour burn–in that induces the maximum subsynchronous vibration to validate proper motor bearing assembly. These processes are consistent with workmanship and material screening by Hamilton Sunstrand for Shuttle hardware (Shuttle does not use air bearing motors, so vibration/run–in tests are different).

Due of the design, commonality to the Shuttle hardware, and screening and inspection processes in place, this exception should not affect risk related to the identified Criticality 1 failure mode (Moderate Temperature Loop Leakage with workaround identified in JSC 48532–5A). Additionally, the AAA did successfully pass qualification random vibration test, demonstrating design acceptability. Therefore, flight AAAs are deemed acceptable for flight.

PG3–112:

ITEM:

Avionics Air Assembly Part Number SV809992

SSP 41172 REQUIREMENT:

Paragraph 4.2.5 Random Vibration, Component Qualification.

Paragraph 4.2.5.4 Supplementary Requirements. Electrical and electronic hardware shall be powered on and electrically monitored for failures or intermittencies during random vibration test.

EXCEPTION:

The AAA will not be powered on and electrically monitored during the Qualification Random Vibration test at its assembly level.

RATIONALE:

The Avionics Air Assembly consists of a Heat Exchanger, Acoustic Cover/Inlet Muffler Assembly, Outlet Muffler, Fan Assembly containing a Brushless DC Motor with a mounted Hall Effects Device board, and a Controller and Filter Board containing electronics. The AAA did undergo qualification random vibration test at an overall level of 8.57 grms without failure; however, this testing was performed without power to the unit.

The Controller and EMI filter boards were tested at overall qualification vibration levels of 10 grms while powered and monitored at subassembly level. The Hall Effects Device board (mounted on the Brushless DC Motor) was qualification random vibration tested while powered and monitored at 8.33 grms at the subassembly level. Additionally, the simple, single–layer Hall Effects Device board with only four active parts is constructed using lap solder connections that are readily inspected, and the board is subjected to electrical bonding, thermal cycling, 300–hour burn–in, and performance testing.

The Brushless DC Motor has air bearings that are designed to operate in a 0 or 1G environment, but could be damaged if operated during qualification vibration testing. The Motor is inspected during build and subjected to electrical continuity, dielectric withstanding voltage, isolation resistance, power cycling, and burn–in workmanship testing. These workmanship screening processes are consistent with those used on hardware Hamilton Sunstrand builds for the Shuttle (Shuttle does not use air bearing motors, so vibration/burn–in tests are different). All solder connections are coated or potted to ensure isolation from conductive particles. Finally, the Avionics Air Assembly will be unpowered during vibration events (launch and ferry–flight environments); therefore, post–vibration functional testing best represents operational use.

PG3–113:

ITEM:

Inlet ORU Part Number SV811840

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. ”When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same.…” This requires that the protoflight hardware Random Vibration test levels and spectrum envelope maximum predicted flight level and spectrum minus 6 dB but not less than the workmanship screening level of 6.1 grms as defined in SSP 41172.

EXCEPTION:

Protoflight Random Vibration testing for the Inlet ORU shall be performed enveloping maximum flight levels of 4.3 Grms.

RATIONALE:

The Inlet ORU consists of a Cabin Air Fan Subassembly (Cabin Fan Motor Controller and Motor with mounted Hall Effects Board), and a Delta Pressure Sensor. The Inlet ORU Cabin Air Fan Housing Assembly is not considered a mechanical assembly requiring precise adjustment or that cannot be inspected properly and therefore would not require acceptance vibration test. However, the Fan Housing Assembly does receive a series of visual and test inspections to ensure workmanship and material conformance. All Motor Controllers, containing the significant electronics, are vibration tested with power and monitoring for one minute at 8.7 grms, well above minimum screening requirements. The Motor has been qualified individually to vibration levels of 6.9 grms for 20 minutes powered, and is tested and inspected at subassembly level to verify workmanship. Also, each Delta Pressure Sensor receives a powered temperature cycle/powered burn–in at the manufacturer, and has been qualified at 5.4 grms for 4 minutes in all three axes with post leakage test. Historically, there have been no workmanship issues on these Sensors during test at Eaton (the manufacturer) or Hamilton Sunstrand. Finally, testing is performed after Inlet ORU vibration to ensure proper unit operation.

Additionally, on–orbit redundancy or spares availability would provide operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–114:

ITEM:

Heat Exchanger ORU Part Number SV813900

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. ”When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same...” This requires that the protoflight hardware Random Vibration test levels and spectrum envelope maximum predicted flight level and spectrum minus 6 dB but not less than the workmanship screening level of 6.1 grms as defined in SSP 41172.

EXCEPTION:

Protoflight Random Vibration testing for the Heat Exchanger ORU shall be performed enveloping maximum flight levels of 4.3 grms.

RATIONALE:

The Heat Exchanger ORU consists of a Temperature Control Check Valve (TCCV) ORU (which includes the Electrical/Electronic Actuator subassembly and hinged doors that close/open a duct), the Temperature Sensor ORU (with the platinum wire wrapped ceramic temperature sensor), and the mechanical Heat Exchanger Condensing subassembly. All electrical/electronic subassemblies receive vibration test prior to higher assembly. The TCCV ORU actuator subassembly is random vibration tested for 3 minutes in all 3 axes while powered and monitored at 8.7 grms acceptance and 9.8 grms qualification, well above minimum screening requirements. The Temperature Sensor ORU is protoflight vibration tested at 15.4 grms for 1 minute in all 3 axes while powered and monitored, and a Temperature Sensor underwent qualification vibration at 16.8 grms for 20 minutes while powered and monitored. The mechanical assemblies of the TCCV ORU and the Heat Exchanger Condenser are not considered to require precise adjustment or that cannot be inspected properly, and therefore would not require an acceptance workmanship vibration test. These mechanical assemblies are tested at the final assembly level to the envelope of the flight vibration environment. During all testing, there have been no failures due to workmanship of the heat exchanger ORU.

Additionally, on–orbit redundancy or TCCV spares availability would provide operational capability for critical services in the event of a failure. Thus the additional risk associated with this exception is limited.

PG3–115:

ITEM:

Internal Self–Sealing Fluid Quick Disconnect (QD) Couplings, Boeing Part Numbers 683–16348 and 683–15179

SSP 41172 REQUIREMENT:

Paragraph 4.2.3 Thermal Cycling Test, Component Qualification.

Paragraph 4.2.3.3C, Test Levels and Duration. The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles.

EXCEPTION:

The duration of the qualification thermal cycle test for the Internal Self–Sealing Fluid Quick Disconnect Couplings shall be two cycles.

RATIONALE:

The performed qualification thermal cycle was designed to simulate actual thermal exposure of the bulkhead mounted male couplings and corresponding female halves. During the test, the QD couplings were operationally tested by mating and de–mating the coupling at the temperature extremes of 33 degrees F and 160 degrees F with the unit pressurized at 14.7 psia. Furthermore, the qualification test exposed the QD coupling to non–operating temperature of –50 degrees F for 30 minutes. Leak test demonstrated that exposure to this lower temperature extreme did not degrade the integrity of the QD coupling seals.

In addition to the above tests performed, a review of the Parker Symetrics Shuttle QD couplings (Part Number MC276–0020, equivalent to the Parker Symetrics ISS QD couplings 683–16348 in respect to design, material, and qualification process) qualification history and service life indicates the performed qualification thermal cycle test on the Station QD coupling meets the intent of SSP 41172. During the shuttle QD couplings qualification program, high pressure QD couplings experienced five thermal cycles and low pressure QD couplings experienced three thermal cycles to qualify for 1000 mate and demate cycles. The Shuttle QD couplings failure history report indicates that over 20 years, greater than 85 percent of the QD couplings failures were leakage and de–mate issues not associated with the stresses of the thermal environment. The QD couplings on the Space Station have a mate/demate life requirement of 250–mate/demate cycles and a service life requirement of 10 years. In light of the on–orbit predicted thermal environment and nominal QD couplings use, both requirements on Station QD couplings are less strenuous than those on Space Shuttle QD couplings. Thus, two qualification thermal cycles are adequate to qualify the ISS QD couplings design.

PG3–116:

ITEM:

Internal Self Sealing Fluid Quick Disconnect Couplings, Boeing Part Numbers 683–16348 (ALL) and 683–15179 (ALL except 683–15179–11, 683–15179–15, 683–15179–21, and 683–15179–25)

SSP 41172 REQUIREMENT:

Paragraph 4.2.2.5, Depress/Repress Vacuum Requirement – Internal components shall be subjected to a depressurization and repressurization test in accordance with either paragraph 4.2.2.5.1 or 4.2.2.5.2. A thermal vacuum qualification test in accordance with paragraphs 4.2.2.1 through 4.2.2.4 may be substituted for this depressurization/repressurization qualification test.

EXCEPTION:

Qualification depress/repress test for the Internal Self–Sealing Fluid Quick Disconnect Couplings is not required. This applies only to internal QDs within the module.

RATIONALE:

Through a combination of testing and analysis, QD couplings have met the intent of the depress/repress requirements. During acceptance testing, the flight QDs are proof–pressure tested in both mated and demated configurations at 2 times Maximum Operating Pressures from 30.4 psia to 500 psia. Testing performed has verified that there are no structural issues associated with the QD couplings. The additional depress/repress testing would not have increased the effectiveness of the screening process. Therefore, the QD couplings are considered acceptable for flight.

PG3–117:

ITEM:

Internal Self Sealing Fluid Quick Disconnect Couplings, Boeing Part Numbers 683–16348

SSP 41172 REQUIREMENT:

Paragraph 4.2.10, Pressure Test, Component Qualification.

Paragraph 4.2.10.2A, Proof Pressure. For such items as pressure vessels, pressure lines, and fittings, the temperature of the component shall be consistent with the critical–use temperature and subjected to a minimum of one cycle of proof pressure. A proof–pressure cycle shall consist of raising the internal pressure (hydrostatically or pneumatically, as applicable) to the proof pressure, maintaining it for five minutes, and then decreasing the pressure to ambient. Evidence of permanent set or distortion or failure of any kind shall indicate failure to pass the test.

EXCEPTION:

The duration of the qualification pressure test for the Internal Self–Sealing Fluid Quick Disconnect Couplings shall be three minutes.

RATIONALE:

The required proof pressure test in accordance with SSP 41172 and SSP 30559 is five minutes at 1.5 times (x) Maximum Operating Pressure (MOP). During qualification, proof pressure testing was performed on the QD couplings for three minutes at 2 x MOP. However, the flight QD couplings were tested for five minutes at 2 x MOP in accordance with specified requirements. In addition, the qualification QD couplings did pass the specified burst pressure test. Therefore, additional proof pressure testing on the qualification internal self sealing fluid QD couplings to fully achieve the required proof pressure test duration may not provide any significant benefit.

PG3–118:

ITEM:

Internal Self Sealing Fluid Quick Disconnect Couplings, Boeing Part Numbers 683–16348 and 683–15179

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.

Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Thermal Vacuum test will not be performed on the Internal Self–Sealing Fluid Quick Disconnect Couplings.

RATIONALE:

Each QD coupling will undergo acceptance leak testing in both mated and unmated conditions at operational pressure with Helium as the test fluid. Acceptance leak testing is the most cost–effective test to screen each QD coupling for workmanship defects upon review of the failure history of the Space Shuttle quick disconnects. The Shuttle QD couplings failure history report indicates that over 20 years, greater than 85 percent of the QD couplings failures were leakage and de–mate issues not associated with the stresses of the thermal environment. Additionally, the flight low pressure QD couplings were coupled and tested at differential pressures and temperatures from 33 degrees F and 160 degrees F in coupler half, mated and de–mated configurations and the high pressure QDs were also tested at differential pressures and temperatures from –50 degrees F and 160 degrees F in coupler half, mated and de–mated configurations. This approach is identical to the environmental screening used for the Parker Symetrics QD couplings used on the Space Shuttle Program in which the Parker Symetrics Shuttle QD couplings (Part Number MC276–0020) is equivalent to the Parker Symetrics ISS QD couplings 683–16348 in respect to design, material, and qualification process. Therefore, as an acceptance thermal vacuum test would not increase the effectiveness of the screening process, the omission of said testing is permitted.

PG3–119:

ITEM:

Internal Self Sealing Fluid Quick Disconnect Couplings, Boeing Part Numbers 683–16348 and 683–15179

SSP 41172 REQUIREMENT:

Paragraph 5.1.3 Thermal Cycle Test, Component Acceptance.

Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Thermal Cycle test will not be performed on the Internal Self–Sealing Fluid Quick Disconnect Couplings.

RATIONALE:

Each QD coupling will undergo acceptance leak testing in both mated and unmated conditions at operational pressure with Helium as the test fluid. Acceptance leak testing is the most cost–effective test to screen each QD coupling for workmanship defects upon review of the failure history of the Space Shuttle quick disconnects. The Shuttle QD couplings failure history report indicates that over 20 years, greater than 85 percent of the QD couplings failures were leakage and de–mate issues not associated with the stresses of the thermal environment. Additionally, the flight low pressure QD couplings were coupled and tested at differential pressures and temperatures from 33 degrees F and 160 degrees F in coupler half, mated and de–mated configurations and the high pressure QDs were also tested at differential pressures and temperatures from –50 degrees F and 160 degrees F in coupler half, mated and de–mated configurations. This approach is identical to the environmental screening used for the Parker Symetrics QD couplings used on the Space Shuttle Program in which the Parker Symetrics Shuttle QD couplings (Part Number MC276–0020) is equivalent to the Parker Symetrics ISS QD couplings 683–16348 in respect to design, material, and qualification process. Therefore, as an acceptance thermal cycle test would not increase the effectiveness of the screening process, the omission of said testing is permitted.

PG3–120:

ITEM:

Internal Self Sealing Fluid Quick Disconnect Couplings, Boeing Part Numbers 683–16348 and 683–15179

SSP 41172 REQUIREMENT:

Paragraph 5.1.4 Random Vibration Test, Component Acceptance.

Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Random Vibration test will not be performed on the Internal Self–Sealing Fluid Quick Disconnect Couplings.

RATIONALE:

Each QD coupling will undergo acceptance leak testing in both mated and unmated conditions at operational pressure with Helium as the test fluid. Acceptance leak testing is the most cost–effective test to screen each QD coupling for workmanship defects. Review of the failure history of the Space Shuttle quick disconnects was completed as the Parker Symetrics Shuttle QD couplings (Part Number MC276–0020) is equivalent to the Parker Symetrics ISS QD couplings 683–16348 in respect to design, material, and qualification process. The Shuttle QD couplings failure history report indicates that over 20 years, greater than 85 percent of the QD couplings failures were leakage and de–mate issues not associated with the stresses of the vibration environment. Additionally, acceptance random vibration testing is not an effective workmanship–screening program as there are no moving mechanical parts. Finally, this exception is in line with the Parker Symetrics Shuttle QD couplings screening as acceptance random vibration tests are not performed. Therefore, as an acceptance random vibration test would not increase the effectiveness of the screening process, the omission of said testing is permitted.

PG3–121:

ITEMS:

Oxygen/Nitrogen Isolation Valve
Nitrogen Isolation Valve (NIV) Part Number 2353052–2–1
Dual Seal Oxygen Isolation Valve (OIV) Part Number 2365618–1–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.

Paragraph 4.2.5.3, Test Levels and Duration. The test duration in each of the three orthogonal axes shall be three times the expected flight exposure time to the maximum predicted level and spectrum or three times the component random vibration acceptance test time if that is greater, but not less than three minutes per axis.

EXCEPTION:

The test duration in the z–axis of the OIV/NIV qualification unit shall be the same as the flight component random vibration acceptance test time (three minutes).

RATIONALE:

The flight OIV/NIVs underwent an in–process Reliability Acceptance Test before entering into acceptance testing. The qualification test unit underwent Reliability Acceptance Testand Acceptance Test (AT) before starting qualification testing. However, no additional run time was included during the qualification test program to provide demonstrated margin in the z–axis for the as–run Reliability Acceptance Test performed on the flight units.

All three tests were conducted at the same level of assembly for the OIV/NIV (i.e., there were no configuration changes or higher level assemblies between RAT, ATP, and QTP). Analysis show that the remaining Demonstrated Fatigue Life Expended on the Flight OIV/NIVs is greater than 80 percent prior to first flight. This result from the overall random vibration levels experienced (12.1 grms) during qualification testing was 6 dB above ATP levels (instead of the standard 3 dB). The OIV/NIV did pass qualification random vibration at this level.

PG3–122:

ITEMS:

Oxygen/Nitrogen Isolation Valve
Nitrogen Isolation Valve (NIV) Part Number 2353052–2–1
Dual Seal Oxygen Isolation Valve (OIV) Part Number 2365618–1–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.5.4, Random Vibration, Component Qualification, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The Oxygen/Nitrogen Isolation Valve will not be powered on and monitored during the qualification random vibration testing.

RATIONALE:

ATP and QTP testing was conducted at the same level of assembly for the OIV/NIV (i.e., there were no configuration changes or higher level assemblies). During qualification testing, the vibration test levels were 3 dB higher than normally used, resulting in a 12.2 grms value with no failures detected. The only parameters available for monitoring are the input current and valve position. These data provide little technical confidence in terms of detecting intermittences. However, additional full functional tests were performed after the random vibration testing, at extreme temperatures, which greatly increases the probability of detecting intermittent failures.

Therefore powering on the Oxygen/Nitrogen Isolation Valve and monitoring for intermittences during qualification random vibration tests may not provide significant value.

PG3–123:

ITEMS:

Oxygen/Nitrogen Isolation Valve
Nitrogen Isolation Valve (NIV) Part Number 2353052–2–1
Dual Seal Oxygen Isolation Valve (OIV) Part Number 2365618–1–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.4.4, Random Vibration, Component Acceptance, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The Oxygen/Nitrogen Isolation Valve will not be powered on and monitored during the acceptance random vibration testing.

RATIONALE:

Reliability Acceptance Testing was performed at the same assembly level as the ATP, with high and low temperature vibration for one minute in the maximum deflection axis. ATP and QTP testing were conducted at the same level of assembly for the OIV/NIV (i.e., there were no configuration changes or higher level assemblies). OIV and NIV units assembled in the PCP underwent ATP testing. The only parameters available for monitoring are the input current and valve position. These data provide little technical confidence in terms of detecting intermittences. However, additional full functional tests were performed after the random vibration testing, at extreme temperatures, which greatly increases the probability of detecting intermittent failures

Therefore powering on the Oxygen/Nitrogen Isolation Valve and monitoring for intermittences during acceptance random vibration tests may not provide significant value.

PG3–124:

ITEM:

Cold Cathode Transducer Part Number 220F01083–001

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Leakage Test will not be performed on the Cold Cathode Transducer.

RATIONALE:

The Cold Cathode Transducer is criticality 3 hardware and is only operated during Vacuum System operation. Limited ORU–level leakage testing was performed; however, no conclusions can be made from the ORU–level test performed on the Qualification unit as Helium levels no greater than the mass spectrometer background level were recorded during the duration (three minutes) testing was performed.

Prior to the vacuum system regression testing conducted at Kennedy Space Center November – December, 1999, a system–level vacuum retention test was completed. The system–level vacuum retention rate was measured and compared to the allowable leak rate as documented on the Vacuum System envelope drawing.

Vacuum Exhaust System was measured at 4.1E–6 sccs; Vacuum Resource System was measured at 4.5E–5 sccs. These are both significantly better than the allowable system–level leak rate of 4.0E–4 sccs for each system.

Upon review of the test procedure and this system–level leak data, Boeing and NASA have judged acceptable the ORUs of the Vacuum System without need for additional leakage testing.

PG3–125:

ITEM:

Cold Cathode Transducer Part Number 220F01083–001

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Leakage Test will not be performed on the Cold Cathode Transducer. Any subsequent leakage tests on flight units shall require testing in full compliance with SSP 41172.

RATIONALE:

The Cold Cathode Transducer is criticality 3 hardware and is only operated during Vacuum System operation. Limited ORU–level leakage testing was performed; however, no conclusions can be made from the ORU–level test performed on the flight units as Helium levels no greater than the mass spectrometer background level was recorded during the duration (three minutes) testing was performed.

Prior to the vacuum system regression testing conducted at Kennedy Space Center November – December, 1999, a system–level vacuum retention test was completed. The system–level vacuum retention rate was measured and compared to the allowable leak rate as documented on the Vacuum System envelope drawing.

Vacuum Exhaust System was measured at 4.1E–6 sccs; Vacuum Resource System was measured at 4.5E–5 sccs. These are both significantly better than the allowable system–level leak rate of 4.0E–4 sccs for each system.

Upon review of the test procedure and this system–level leak data, Boeing and NASA have judged acceptable the ORUs of the Vacuum System without need for additional leakage testing.

Any subsequent leakage tests on the flight units or additional procurements shall require testing in full compliance with SSP 41172.

PG3–126:

ITEM:

2.5–Inch Vacuum Valve Part Number 220F01082–003

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Leakage Test will not be performed on the 2.5–Inch Vacuum Valve.

RATIONALE:

Though the lack of a fully–qualified 2.5–Inch Vacuum Valve design for leakage introduces risk, as–performed tests on the flight units do provide some confidence in the design. Prior to the vacuum system regression testing conducted at Kennedy Space Center November – December, 1999, a system–level vacuum retention test was completed. The system–level vacuum retention rate was measured and compared to the allowable leak rate as documented on the Vacuum System envelope drawing.

Vacuum Exhaust System was measured at 4.1E–6 sccs Helium; Vacuum Resource System was measured at 4.5E–5 sccs Helium. These are both significantly better than the allowable system–level leak rate of 4.0E–4 sccs Helium for each system.

The Program accepts the flight units installed without the need of additional qualification testing. On future flight units, via an update to the Acceptance Test Procedure, leakage testing will be performed by a SSP 41172–compliant Method II leak test method to mitigate the risk associated.

PG3–127:

ITEM:

2.5–Inch Vacuum Valve Part Number 220F01082–003

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Leakage Test will not be performed on the 2.5–Inch Vacuum Valve. Any subsequent leakage tests on flight units or additional procurements shall require testing in full compliance with SSP 41172.

RATIONALE:

Prior to the vacuum system regression testing conducted at Kennedy Space Center November – December, 1999, a system–level vacuum retention test was completed. The system–level vacuum retention rate was measured and compared to the allowable leak rate as documented on the Vacuum System envelope drawing.

Vacuum Exhaust System was measured at 4.1E–6 sccs Helium; Vacuum Resource System was measured at 4.5E–5 sccs Helium. These are both significantly better than the allowable system–level leak rate of 4.0E–4 sccs Helium for each system.

Upon review of the test procedure and this system–level leak data, Boeing and NASA have judged acceptable the ORUs of the Vacuum System without need for additional leakage testing. With an update to the Acceptance Test Procedure, future leakage testing will be performed by a SSP 41172–compliant Method II leak test method.

PG3–128:

ITEM:

Non Propulsive Vent (NPV) Part Number 220F01009

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.

Paragraph 4.2.11.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The duration of the Qualification Leakage Test of the NPV shall be five minutes.

RATIONALE:

The Qualification NPV is connected to a pressure–regulated Helium source and placed under a bell jar that is plumbed to a Helium Leak Detector. An auxiliary vacuum pump is used to evacuate the bell jar to low pressure (1E–03 Torr). At this time, the auxiliary pump is valved off, the leak detector is valved on, and the system is allowed to stabilize. When the system pressure has stabilized, Helium is applied to the NPV at the specified pressure and this pressure is maintained for the specified duration. At the end of the test duration the leakage rate indicated by the leak detector is recorded.

A reasonable duration for the leakage test described can be determined based on the time constant of the system. The time constant of the system described is

T = V/S

where T is the time constant;

V is the volume of the bell jar; and

S is the effective pumping speed of the leak detector combined with its plumbing to the bell jar.

A test duration of three time constants is acceptable. After three time constants, the Helium partial pressure in the bell jar will reach 95 percent of its final (equilibrium) value. Extending the test duration beyond three time constants would have little effect on the result. However, decreasing the test duration to less than three time constants would reduce the leakage rate reported by the leak detector to something less than the actual leakage rate.

The test system time constant calculated for the NPV was 234 seconds. The test duration of the NPV was 300 seconds.

As tested, the NPV exhibited a leakage rate of 1.0E–07 sccs He, well below the specified maximum allowable leakage rate of 1.0E–04 sccs He. Any variation in this leakage rate between the optimum time constant calculated for the system and the actual test time is negligible with respect to total allowable leakage of cabin air from the USL Module.

PG3–129:

ITEM:

Non Propulsive Vent (NPV) Part Number 220F01009

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.

Paragraph 5.1.7.3B, Test Levels and Duration. The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration of the test shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The duration of the Acceptance Leakage Test of the NPV shall be five minutes.

RATIONALE:

The Flight NPV is connected to a pressure–regulated Helium source and placed under a bell jar that is plumbed to a Helium Leak Detector. An auxiliary vacuum pump is used to evacuate the bell jar to low pressure (1E–03 Torr). At this time, the auxiliary pump is valved off, the leak detector is valved on, and the system is allowed to stabilize. When the system pressure has stabilized, Helium is applied to the NPV at the specified pressure and this pressure is maintained for the specified duration. At the end of the test duration the leakage rate indicated by the leak detector is recorded.

A reasonable duration for the leakage test described can be determined based on the time constant of the system. The time constant of the system described is

$T = V/S$

where T is the time constant;

V is the volume of the bell jar; and

S is the effective pumping speed of the leak detector combined with its plumbing to the bell jar.

A test duration of three time constants is acceptable. After three time constants, the Helium partial pressure in the bell jar will reach 95 percent of its final (equilibrium) value. Extending the test duration beyond three time constants would have little effect on the result. However, decreasing the test duration to less than three time constants would reduce the leakage rate reported by the leak detector to something less than the actual leakage rate.

The test system time constant calculated for the NPV was 234 seconds. The test duration of the NPV was 300 seconds.

As tested, the NPVs exhibited an average leakage rate of 6.2E–07 sccs He, well below the specified maximum allowable leakage rate of 1.0E–04 sccs He. Any variation in this leakage rate between the optimum time constant calculated for the system and the actual test time is negligible with respect to total allowable leakage of cabin air from the USL Module.

PG3–130:

ITEM:

1.0–inch Vacuum Valve Assembly, Part Numbers 220F01087–001 and 220F01087–003

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle. The minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The minimum cold thermal limit of the Qualification Thermal Cycling Test of the 1.0–Inch Vacuum Valve Assembly shall be 33 degrees F.

RATIONALE:

The 1.0–Inch Vacuum Valve Assembly successfully operated over a 140–degree F temperature sweep. The qualification temperature range tested was 33 degrees F to 173 degrees F. The minimum qualification temperature tested did not encompass the minimum actual acceptance test procedure temperature of 30 degrees F. Also, it did not encompass the minimum transportation non–operation requirement of – 40 degrees F.

The 1.0–Inch Vacuum Valve Assembly only operates in the nominal US Laboratory environment, with an operating temperature range of +55 degrees F to 109 degrees F. The qualification test temperature range encompassed the on–orbit predicted operating temperature with a 22 degree F margin (cold) / 64 degree F margin (hot). The test encompassed the actual acceptance test procedure temperature with a 43 degrees F margin at the maximum temperature.

At the low non–operation transportation temperature, Seal Type S383–70 in the 1.0–Inch Vacuum Valve has not been verified to be temperature–resistant via testing. However, military specification ZZ–R–765B indicates that Seal Type S383–70 is low temperature and low compression set resistant and is certified for use from – 103 degrees F to 400 degrees F. This range encompasses potential non–operation transportation and launch environments.

The 1.0–Inch Vacuum Valve Assembly has a calculated MTBF of 226,623 hours. The 1.0–Inch Vacuum Valve Assembly is nominally closed. It operates only during rack payload experiment pump down or exhaust. This operation occurs only during nominal USL environmental conditions. The 1.0–Inch Vacuum Valve Assembly is a Criticality 3 ORU. The Vacuum Exhaust System redundancy created by the inclusion of the 2.5–inch Valve Assembly and the Non–Propulsive Vent allows the system to maintain functionality if a 1.0–inch Vacuum Valve Assembly failure would occur.

PG3–131:

ITEM:

1.0–inch Vacuum Valve Assembly, Part Numbers 220F01087–001 and 220F01087–003

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.

Paragraph 5.1.3.3B, Test Levels and Duration. Temperature. (See Figure 5–1.) The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle. For components identified in Table 5–1, with note 4, there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The minimum cold thermal limit of the Acceptance Thermal Cycling Test of the 1.0–Inch Vacuum Valve Assembly shall be 30 degrees F.

The maximum hot thermal limit of the Acceptance Thermal Cycling Test of the 1.0–Inch Vacuum Valve Assembly shall be 130 degrees F.

RATIONALE:

The 1.0–Inch Vacuum Valve Assembly successfully operated over a 100–degree F sweep for 9 cycles. The temperature range tested was +30 degrees F to 130 degrees F. The minimum acceptance temperature tested was not 20 degrees F above the Qualification minimum temperature nor met the minimum transportation non–operation requirement of – 40 degrees F. The maximum acceptance temperature did not meet the maximum transportation non–operation requirement of 160 degrees F. Yet, the 1.0–Inch Vacuum Valve Assembly only operates in the nominal USL environment. The USL operating temperature range is +55 degrees F to +109 degrees F. Therefore the temperature range tested encompasses the operating environment of the valve.

At the low non–operation transportation temperature, Seal Type S383–70 in the 1.0–Inch Vacuum Valve has not been verified to be temperature resistant via testing. However, military specification ZZ–R–765B indicates that Seal Type S383–70 is low temperature and low compression set resistant and is certified for use from – 103 degrees F to 400 degrees F. This range encompasses potential non–operation transportation and launch environments.

The 1.0–Inch Vacuum Valve Assembly has a calculated MTBF of 226,623 hours and is nominally closed. It operates only during rack payload experiment pump down or exhaust. This operation occurs only during nominal USL environmental conditions. The Vacuum Exhaust System redundancy created by the inclusion of the 2.5–inch Valve Assembly and the Non–Propulsive Vent allows the system to maintain functionality if a 1.0–Inch Vacuum Valve Assembly failure (valve stuck Open) occurs.

PG3–132:

ITEM:

1.0–inch Vacuum Valve Assembly, Part Numbers 220F01087–001 and 220F01087–003

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification.

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The 1.0–Inch Vacuum Valve Assembly will not be electrically energized and monitored during the qualification random vibration testing.

RATIONALE:

The 1.0–Inch Vacuum Valve Assembly contains electronics consisting only of motors. The 1.0–Inch Vacuum Valve Assembly electrical test connector contains only a minimal number of electronic components, and does not utilize a printed circuit board with a card edge connector. The electrical components are potted internally to dampen excitation. This electrical excitation powers the valve with an actuation time of 7 seconds and provides an output signal (open/closed). The valves are located in the USL standoffs with each valve dedicated to an ISPR rack. The Vacuum Exhaust System redundancy allows that a failure by the 1.0–Inch Vacuum Valve Assembly does not prevent the exhaust of waste gas. The 2.5–inch Vacuum Valve and the Non–Propulsive Vent must also fail before the USL ability to vent payload gas is lost.

The 1.0–Inch Vacuum Valve Assembly has a calculated MTBF of 226,623 hours and is nominally closed. It operates only during rack payload experiment pump down or exhaust. The pump down operation occurs only during nominal USL environmental conditions. The Vacuum Exhaust System redundancy created by the inclusion of the 2.5–inch Valve Assembly and the Non–Propulsive Vent allows the system to maintain functionality if a 1.0–Inch Vacuum Valve Assembly failure (valve fails Open) occurs. The 1.0–Inch Vacuum Valve Assembly has a Criticality 3 rating. Factoring the hardware criticality and the 1.0–Inch Vacuum Valve Assembly successfully completing functional test following the qualification random vibration test, the powering on the 1.0–Inch Vacuum Valve Assembly and monitoring for intermittences during qualification random vibration tests may not provide significant value.

PG3–133:

ITEM:

1.0–inch Vacuum Valve Assembly, Part Numbers 220F01087–001 and 220F01087–003

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance.

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The 1.0–Inch Vacuum Valve Assembly will not be electrically energized and monitored during the acceptance random vibration testing.

RATIONALE:

The 1.0–Inch Vacuum Valve Assembly contains electronics consisting only of motors that are not required to operate in the random vibration environment. The 1.0–Inch Vacuum Valve Assembly does not have an electrical test connector, contain only a minimal number of electronic components, and do not utilize a printed circuit board with a card edge connector.

Because of the low number of electrical parts, the specified item has an extremely low probability of intermittent defects. As the 1.0–Inch Vacuum Valve Assembly does not have an electrical test connector and is not required to demonstrate mechanical functionality during the random vibration test, only a limited number of the possible intermittent defects could be detected during the test. Therefore, powering on the specified item and monitoring for intermittences during acceptance random vibration tests may not provide significant value. Thus, there is little added risk to the program associated with this exception.

The 1.0–Inch Vacuum Valve Assembly is nominally closed and it operates only during rack payload experiment pump down or exhaust. This operation occurs only during nominal USL environmental conditions. The Vacuum Exhaust System redundancy created by the inclusion of the 2.5–inch Valve Assembly and the Non–Propulsive Vent allows the system to maintain functionality if 1.0–Inch Vacuum Valve Assembly failure (valve fails Open) occurs. The 1.0–Inch Vacuum Valve Assembly has a Criticality 3 rating. Factoring the hardware criticality and the 1.0–Inch Vacuum Valve Assembly successfully completing functional test following the acceptance random vibration test, the powering on the 1.0–Inch Vacuum Valve Assembly and monitoring for intermittences during acceptance random vibration tests may not provide significant value.

PG3–134:

ITEM:

1.0–inch Vacuum Valve Assembly, Part Numbers 220F01087–001 and 220F01087–003

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Leakage Test will not be performed on the 1.0–Inch Vacuum Valve.

RATIONALE:

Though the lack of a fully–qualified 1.0–Inch Vacuum Valve design for leakage introduces risk, as–performed tests on the flight units do provide some confidence in the design. Prior to the vacuum system regression testing conducted at Kennedy Space Center November – December, 1999, a system–level vacuum retention test was completed. The system–level vacuum retention rate was measured and compared to the allowable leak rate as documented on the Vacuum System envelope drawings.

Vacuum Exhaust System was measured at 4.1E–6 sccs; Vacuum Resource System was measured at 4.5E–5 sccs. These are both significantly better than the allowable system–level leak rate of 4.0E–4 sccs for each system.

The Program accepts the flight units installed without the need of additional qualification testing. On future flight units, via an update to the Acceptance Test Procedure, leakage testing will be performed by a SSP 41172–compliant Method I leak test method to mitigate the risk associated.

Only one of the thirteen 1.0–Inch Vacuum Valve Assemblies in the USL operates at a given time. Additionally, the Vacuum Exhaust System redundancy created by the inclusion of the 2.5–inch Valve Assembly and the Non–Propulsive Vent allows the system to maintain functionality if there is a 1.0–Inch Vacuum Valve Assembly failure (valve fails open). Thus, as the 1.0–Inch Vacuum Valve Assembly is a Criticality 3 component and the overall performance of the Vacuum System as shown through Vacuum System Regression Testing and the vacuum retention test, additional testing of the 1.0–Inch Vacuum Valve Assembly will not yield any significant benefit.

PG3–135:

ITEM:

1.0–inch Vacuum Valve Assembly, Part Numbers 220F01087–001 and 220F01087–003

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Leakage Test will not be performed on the 1.0–Inch Vacuum Valve. Any subsequent leakage tests on flight units shall require testing in full compliance with SSP 41172.

RATIONALE:

Prior to the vacuum system regression testing conducted at Kennedy Space Center November – December, 1999, a system–level vacuum retention test was completed. The system–level vacuum retention rate was measured and compared to the allowable leak rate as documented on the Vacuum System envelope drawings.

Vacuum Exhaust System was measured at 4.1E–6 sccs; Vacuum Resource System was measured at 4.5E–5 sccs. These are both significantly better than the allowable system–level leak rate of 4.0E–4 sccs for each system.

Upon review of the test procedure and this system–level leak data, Boeing and NASA have judged acceptable the ORUs of the Vacuum System without need for additional leakage testing. With an update to the Acceptance Test Procedure, future leakage testing will be performed by a SSP 41172–compliant Method I leak test method.

Only one of the thirteen 1.0–Inch Vacuum Valve Assemblies in the USL operates at a given time. Additionally, the Vacuum Exhaust System redundancy created by the inclusion of the 2.5–inch Valve Assembly and the Non–Propulsive Vent allows the system to maintain functionality if there is a 1.0–Inch Vacuum Valve Assembly failure (valve fails open). Thus, as the 1.0–Inch Vacuum Valve Assembly is a Criticality 3 component and the overall performance of the Vacuum System as shown through Vacuum System Regression Testing and the vacuum retention test, additional testing of the 1.0–Inch Vacuum Valve Assembly will not yield any significant benefit.

PG3–136:

ITEM:

Vent and Relief Valve Part Number 2353026–1–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.

Paragraph 4.2.11.3F, Test Levels and Duration, Method VI. The duration of the test shall be no less than 60 minutes.

EXCEPTION:

The duration of the Qualification Leakage Test of the Vent and Relief Valve shall be at least 15 minutes.

RATIONALE:

The as–performed leakage test on the Qualification Vent and Relief Valve (Serial Number D0001) was 15 minutes. To confirm that this is sufficient time for the leakage rate to become stable, a spare Vent and Relief Valve (Serial Number D0008) was tested in accordance with a modified acceptance test procedure for a duration of 60 minutes with a Varian leak detector connected to a strip–chart recorder. For both the isolation and control valves within the Vent and Relief Valve, the leakage rate was observed to be stable in a period less than 15 minutes. In addition, each valve in the D0008 spare ORU met its leakage requirement of less than 1.8E–02 sccs/sec Helium, with measured rates of 2.3E–03 sccs/sec Helium and 3.4E–05 sccs/sec Helium, respectively. Thus, accumulated data does indicate that the duration of 15 minutes for leakage testing is sufficient for the Vent and Relief Valve.

PG3–137:

ITEM:

Vent and Relief Valve Part Number 2353026–1–1, Serial Numbers D0003, D0004, D0005, D0007 (Spare Unit)

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.

Paragraph 5.1.7.3F, Test Levels and Duration, Method VI. The duration of the test shall be no less than 60 minutes.

EXCEPTION:

The duration of the Acceptance Leakage Test of the Vent and Relief Valve shall be at least 15 minutes.

RATIONALE:

The as–performed leakage test on the Flight and Spare Vent and Relief Valves as indicated were 15 minutes. To confirm that this is sufficient time for the leakage rate to become stable, a spare Vent and Relief Valve (Serial Number D0008) was tested in accordance with a modified acceptance test procedure for a duration of 60 minutes with a Varian leak detector connected to a strip–chart recorder. For both the isolation and control valves within the Vent and Relief Valve, the leakage rate was observed to be stable in a period less than 15 minutes. In addition, each valve in the D0008 spare ORU met its leakage requirement of less than 1.8E–02 sccs/sec Helium, with measured rates of 2.3E–03 sccs/sec Helium and 3.4E–05 sccs/sec Helium, respectively. Thus, accumulated data does indicate that the duration of 15 minutes for leakage testing is sufficient for the Vent and Relief Valve.

PG3–138:

ITEM:

Positive Pressure Transducer Part Number 220F01101

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

Qualification Random Vibration Testing of the Positive Pressure Transducer without power on and monitoring is permitted.

RATIONALE:

The Positive Pressure Transducer contains electronics consisting only of an electro–mechanical pressure sensor, contains a minimal number of electronic components that are internally potted (thereby minimizing the probability of intermittent defects), and does not utilize a printed circuit board with a card edge connector. Additionally, Positive Pressure Transducer functional testing at the completion of the random vibration testing was successfully performed with no anomalies detected. Positive Pressure Transducer follow–on spares will be random vibration tested with power on and monitoring which gains additional confidence in the Positive Pressure Transducer manufacturing processes. Therefore, retesting the Positive Pressure Transducer qualification unit powered on and monitored for intermittences during random vibration test may not provide significant value.

Failure of the Positive Pressure Transducer only causes loss of the capability to measure upper range vacuum pressures. The USL will still be able to measure vacuum system pressures in the mid– and low ranges with the Pirani gauge and Cold Cathode Transducer.

PG3–139:

ITEM:

Positive Pressure Transducer Part Number 220F01101, Serial Numbers 102 and 103

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

Acceptance Random Vibration Testing of the Positive Pressure Transducer without power on and monitoring is permitted.

RATIONALE:

The Positive Pressure Transducer contains electronics consisting only of an electro–mechanical pressure sensor, contains a minimal number of electronic components that are internally potted (thereby minimizing the probability of intermittent defects), and does not utilize a printed circuit board with a card edge connector. Additionally, Positive Pressure Transducer functional testing at the completion of the random vibration testing was successfully performed with no anomalies detected. Positive Pressure Transducer follow–on spares will be random vibration tested with power on and monitoring which gains additional confidence in the Positive Pressure Transducer manufacturing processes. Therefore, removing from installed locations and retesting delivered Positive Pressure Transducers with power on and monitoring for intermittences during acceptance random vibration tests may not provide significant value.

Failure of the Positive Pressure Transducer only causes loss of the capability to measure upper range vacuum pressures. The USL will still be able to measure vacuum system pressures in the mid– and low ranges using the Pirani gauge and Cold Cathode Transducer.

PG3–140:

ITEM:

Positive Pressure Transducer Part Number 220F01101

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Leakage Test will not be performed on the Positive Pressure Transducer.

RATIONALE:

The Positive Pressure Transducer is criticality 3 hardware and is only operated during Vacuum System operation. Limited ORU–level leakage testing was performed; however, no conclusions can be made from the ORU–level test performed on the Qualification unit as Helium levels no greater than the mass spectrometer background level was recorded during the duration (three minutes) testing was performed.

Prior to the vacuum system regression testing conducted at Kennedy Space Center November – December, 1999, a system–level vacuum retention test was completed. The system–level vacuum retention rate was measured and compared to the allowable leak rate as documented on the Vacuum System envelope drawing.

Vacuum Exhaust System was measured at 4.1E–6 sccs; Vacuum Resource System was measured at 4.5E–5 sccs. These are both significantly better than the allowable system–level leak rate of 4.0E–4 sccs for each system.

Upon review of the test procedure and this system–level leak data, Boeing and NASA have judged acceptable the ORUs of the Vacuum System without need for additional leakage testing.

PG3–141:

ITEM:

Positive Pressure Transducer Part Number 220F01101, Serial Numbers 102 and 103

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Leakage Test will not be performed on the Positive Pressure Transducer. Any subsequent leakage tests on the units indicated shall require testing in full compliance with SSP 41172.

RATIONALE:

The Positive Pressure Transducer is criticality 3 hardware and is only operated during Vacuum System operation. Limited ORU–level leakage testing was performed; however, no conclusions can be made from the ORU–level test performed on the flight units as Helium levels no greater than the mass spectrometer background level was recorded during the duration (three minutes) testing was performed.

Prior to the vacuum system regression testing conducted at Kennedy Space Center November – December, 1999, a system–level vacuum retention test was completed. The system–level vacuum retention rate was measured and compared to the allowable leak rate as documented on the Vacuum System envelope drawing.

Vacuum Exhaust System was measured at 4.1E–6 sccs; Vacuum Resource System was measured at 4.5E–5 sccs. These are both significantly better than the allowable system–level leak rate of 4.0E–4 sccs for each system.

Upon review of the test procedure and this system–level leak data, Boeing and NASA have judged acceptable the ORUs of the Vacuum System without need for additional leakage testing.

Any subsequent leakage tests on the units indicated or additional procurements shall require testing in full compliance with SSP 41172.

PG3–142:

ITEM:

Three–Way Mix Valve, Part Number 2365504–1–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperature of the Three–Way Mix Valve shall be 33 degrees F during qualification thermal cycle testing.

RATIONALE:

The minimum predicted operating temperature of the Three–Way Mix Valve is 55 degrees F. Three–Way Mix Valves were tested while operating to a minimum temperature of 33 degrees F. The qualification unit was tested to the same temperature limit of 33 degrees F. As the Three–Way Mix Valve employs water as the operational fluid, environmental testing to temperatures lower than 32 degrees F would causes the fluid to freeze. Thus, the resulting thermal expansion would damage the Three–Way Mix Valve. Yet, since the qualification thermal cycle temperature did envelop (without margin) the acceptance thermal cycle minimum temperature limit, the risk associated with this exception is minimal.

The Material Identification and Usage List shows that the mechanical portions of the assembly are insensitive to a non–operational low temperature of 0 degrees F.

In the event that there is any operational malfunction due to some factor that would have been screened out by non–operational cold testing of the unit, then it should be noted that the components within the Thermal Control System loops have manual operational capability. Additionally, an operational workaround has been developed and documented to recover partial functionality should the valve seize. A workaround requires the crew jumpering loads to the Moderate Temperature Loop. The workaround has been demonstrated and accepted by NASA as documented in JSC 48532–5A. Finally, there is a pre–positioned spare on–orbit.

PG3–143:

ITEMS:

Audio Interface Unit, Part Number 3000002–301
Internal Audio Controller, Part Number 3000016–301
Assembly Contingency System/UHF Communication Subsystem Interface Unit, Part Number 3000022–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification.

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

The Audio Interface Unit, Internal Audio Controller, and the Assembly Contingency System/UHF Communication Subsystem Interface Unit shall not be electrically energized and monitored during the qualification random vibration test.

RATIONALE:

Functional testing was performed on these components at the completion of the acceptance random vibration testing which provides some workmanship screening. The Internal Audio System ORUs not being electrically energized and monitored poses little performance risk as the hardware will not be powered on and functioned during the flight vibration environments. In all cases, except for the Audio Interface Unit prior to Flight 5A.1, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–144:

ITEMS:

Audio Interface Unit, Part Number 3000002–301
Internal Audio Controller, Part Number 3000016–301
Assembly Contingency System/UHF Communication Subsystem Interface Unit, Part Number 3000022–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

The Audio Interface Unit, Internal Audio Controller, and the Assembly Contingency System/UHF Communication Subsystem Interface Unit shall not be electrically energized and monitored during the acceptance random vibration test.

RATIONALE:

Functional testing was performed on these components at the completion of the acceptance random vibration testing which provides some workmanship screening. The Internal Audio System ORUs not being electrically energized and monitored poses little performance risk as the hardware will not be powered on and functioned during the flight vibration environments. In all cases, except for the Audio Interface Unit prior to Flight 5A.1, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–145:

ITEMS:

Audio Interface Unit, Part Number 3000002–301
Internal Audio Controller, Part Number 3000016–301
Assembly Contingency System/UHF Communication Subsystem Interface Unit, Part Number 3000022–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification. Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 4.2.5.3B, Test Levels and Duration. Acceptance test levels and spectrum plus test tolerances.

EXCEPTION:

The Qualification Random Vibration Test of the Audio Interface Unit, Internal Audio Controller, and the Assembly Contingency System/UHF Communication Subsystem Interface Unit shall be performed at a level of 4.4 grms with a maximum spectral density of 0.04 G^2/Hz.

RATIONALE:

The Qualification Random Vibration environment enveloped the maximum predicted flight level of 4.3 grms for US–located uses. The Internal Audio System ORUs were designed and analyzed to survive a 4.4 grms environment. The contractor Harris used conservative design practices including designing for stiffness with self–imposed safety factors for deflection and fatigue. All fasteners have locking features or are staked, and are inspected during assembly. Additionally, all circuit assemblies are conformal coated which precludes malfunctions due to shorting from conductive hardware loose in the unit. In all cases, except for the Audio Interface Unit prior to Flight 5A.1, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–146:

ITEMS:

Audio Interface Unit, Part Number 3000002–301
Internal Audio Controller, Part Number 3000016–301
Assembly Contingency System/UHF Communication Subsystem Interface Unit, Part Number 3000022–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance. Component random vibration test levels and spectrums shall be the envelope of the following:

Paragraph 5.1.4.3B, Test Levels and Duration. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime contractor.

EXCEPTION:

The Acceptance Random Vibration Test of the Audio Interface Unit, Internal Audio Controller, and the Assembly Contingency System/UHF Communication Subsystem Interface Unit shall be performed at a level of 3.1 grms with a maximum spectral density of 0.02 G^2/Hz.

RATIONALE:

The acceptance random vibration testing performed by the contractor Harris does not meet the required workmanship screening levels in Figure 5–2, yet it does provide value as a workmanship screen. The Internal Audio System ORUs were designed and analyzed to survive a 4.4 grms environment, beyond the maximum predicted flight level of 4.3 grms for US–located uses. Harris used conservative design practices including designing for stiffness with self–imposed safety factors for deflection and fatigue. All fasteners have locking features or are staked, and are inspected during assembly. Additionally, all circuit assemblies are conformal coated which precludes malfunctions due to shorting from conductive hardware loose in the unit. In all cases, except for the Audio Interface Unit prior to Flight 5A, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–147:

ITEMS:

Audio Interface Unit, Part Number 3000002–301
Internal Audio Controller, Part Number 3000016–301
Assembly Contingency System/UHF Communication Subsystem Interface Unit, Part Number 3000022–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

The Audio Interface Unit, Internal Audio Controller, and the Assembly Contingency System/UHF Communication Subsystem Interface Unit shall not be subjected to their maximum predicted non–operating temperature of 125 degrees F during acceptance thermal cycle testing.

RATIONALE:

The maximum temperature for the Internal Audio System ORUs during its life cycle is derived under ferry flight conditions. This predicted maximum non–operation temperature under these conditions is 125 degrees F. However, the actual operating environment of the Internal Audio System ORUs is benign as they are within pressurized volumes and are coldplate–mounted which controls the operating environment experienced. The worst–case predicted maximum temperatures only occur during depot return activities or in the event of an aborted launch attempt with a contingency site landing.

The Acceptance Thermal Cycle testing was performed at the 90–degree coldplate temperature during the hot portion of the thermal cycle test. Analysis indicates that during the hot portion, operational testing resulted in internal EEE part case temperatures from 126 degrees F to 178 degrees F. Thus, internal electrical components were tested at temperatures that exceeded the required acceptance non–operating limit. In addition, all parts and materials in the Internal Audio System ORUs are certified beyond non–operational limit. In all cases, except for the Audio Interface Unit prior to Flight 5A.1, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–148:

ITEMS:

Audio Interface Unit, Part Number 3000002–301
Internal Audio Controller, Part Number 3000016–301
Assembly Contingency System/UHF Communication Subsystem Interface Unit, Part Number 3000022–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Acceptance.

Paragraph 4.2.3.3B, Test Levels and Duration. For electrical/electronic equipment where the minimum sweep (paragraph 5.1.3.3–2) does not encompass by the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F.

EXCEPTION:

The Audio Interface Unit, Internal Audio Controller, and the Assembly Contingency System/UHF Communication Subsystem Interface Unit shall experience a 98 degrees F sweep between the minimum and maximum coldplate–mounted qualification thermal cycle operating temperatures.

RATIONALE:

The coldplate–mounted operational temperatures during qualification thermal cycle testing were from 12 degrees F to 110 degrees F. This envelops the predicted operational temperatures of the ORUs with required margin of 20 degrees F. Additionally, non–operational qualification thermal cycle testing from –30 degrees F to 110 degrees F provides a non–operating temperature sweep in compliance with the requirement. In all cases, except for the Audio Interface Unit prior to Flight 5A.1, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–149:

ITEMS:

Audio Interface Unit, Part Number 3000002–301
Internal Audio Controller, Part Number 3000016–301
Assembly Contingency System/UHF Communication Subsystem Interface Unit, Part Number 3000022–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3B, Test Levels and Duration. For components identified in Table 5–1, with note 4, there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The Audio Interface Unit, Internal Audio Controller, and the Assembly Contingency System/UHF Communication Subsystem Interface Unit shall experience a 57 degrees F sweep between the minimum and maximum coldplate–mounted qualification thermal cycle operating temperatures.

The minimum coldplate–mounted test temperature shall be 33 degrees F.

RATIONALE:

The coldplate–mounted operational temperatures during acceptance thermal cycle testing were from 33 degrees F to 90 degrees F. This envelops the predicted operational temperatures of the ORUs. Additionally, non–operational acceptance thermal cycle testing from –10 degrees F to 90 degrees F provides a non–operating temperature sweep in compliance with the requirement. In all cases, except for the Audio Interface Unit prior to Flight 5A.1, redundancy or additional units are available on–board that would provide the necessary operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–150:

ITEM:

Water Separator ORU Part Number SV813920

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same..." This requires that the protoflight hardware Random Vibration test levels and spectrum envelope maximum predicted flight level and spectrum minus 6 dB but not less than the workmanship screening level of 6.1 grms as defined in SSP 41172.

EXCEPTION:

Protoflight Random Vibration testing for the Water Separator ORU shall be performed enveloping maximum flight levels of 4.3 grms.

RATIONALE:

The Water Separator ORU consists of an Air Check Valve, Liquid Check Valve, Water Separator, and electrical/electronic subassemblies consisting of a Solenoid Valve, Controller, Motor, Pressure Sensor, and Liquid Sensor. The flight unit was protoflight vibration tested to envelope the maximum predicted flight level, based on an OASPL of 141 dB, as defined by ECP–068 for rack mounted equipment. The as–run test data over the X–, Y–, and Z–axes for Serial Number 0002 did not encompass the minimum SSP 41172 screening level from 200 to 2000 Hertz. However, based on test, the natural harmonics of this ORU occur at approximately 190 Hertz and below, so the variation above 200 Hertz is considered insignificant.

As the Water Separator does not contain mechanical assemblies requiring precise adjustment or that cannot be inspected properly, the mechanical components do not require a vibration–screening test. The Motor Controller subassembly, with its electronics, experience protoflight random vibration tests while powered and monitored for one minute at 11.8 grms, well above minimum screening requirements. The Solenoid Valve, the Motor, and the Pressure Sensor have been qualified individually to vibration levels above 6.1 grms. The flight components receive tests and inspections at their subassembly level for material and workmanship defects. The liquid sensor consists of a wire harness terminated in a mechanical housing with the wire soldered to a terminal lug brazed to an electrode; it is also easily inspectable for workmanship defects. Furthermore, all ORU solder connections are coated or potted to ensure isolation from conductive particles.

There have been no failures during test of the Water Separator ORU. On–orbit redundancy or spares availability would provide operational capability for critical services in the event of a failure. Thus, the additional risk associated with this exception is limited.

PG3–151:

ITEM:

Water Separator ORU Part Number SV813920

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same..." This requires that the Protoflight hardware be exposed to a minimum of eight thermal cycles.

EXCEPTION:

The duration of the Protoflight Thermal Cycle Test for the CCAA Water Separator ORU shall be one cycle.

RATIONALE:

The CCAA Water Separator ORU consists of an Air Check Valve, Liquid Check Valve, Water Separator, and electrical/electronic subassemblies consisting of a Solenoid Valve, Controller, Motor, Pressure Sensor, and Liquid Sensor. At the ORU level, only one non–operating thermal cycle was run to demonstrated operability after exposure to non–operation temperatures.

The moving mechanical subassemblies do not require precise adjustment, are inspected and tested effectively at subassembly level, and therefore would not require temperature cycling to detect workmanship or material defects. The Controller, with its electronics, has experienced eight thermal cycles in accordance with SSP 41172. The Solenoid Valve, Motor, and Pressure Sensor have been qualified over 24 thermal cycles at its component level. Additionally, the solenoid is simple and inspectable. The Motor is workmanship screened via burn–in, on/off cycle, functional, dielectric withstanding voltage, and isolation resistance testing at subassembly level. The Pressure Sensor experience thermal cycling, burn–in, and temperature extreme testing at the manufacturer; no workmanship issues have been uncovered during test. The liquid sensor consists of a wire harness terminated in a mechanical housing with the wire soldered to a terminal lug brazed to an electrode; it is also easily inspectable. Thus, the one–cycle protoflight thermal cycle test to demonstrate survivability at the non–operating temperature requirement is sufficient in the context of all performed lower–level testing.

PG3–152:

ITEM:

1.75–inch Three–Way Valve Part Number B40205–1 (Boeing Part Number 683–13024)

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperature of the 1.75–inch Three–Way Valve shall be 10 degrees F during qualification thermal cycle testing.

RATIONALE:

The qualification thermal cycle test for the 1.75–inch Three–Way Valve was performed from a minimum temperature of 10 degrees F to a maximum temperature of 180 degrees F. The acceptance thermal cycle test was previously performed from a minimum temperature of 20 degrees F to a maximum temperature limit of 160 degrees F. The valves were cycled at these temperature extremes; however, only 10 degrees F margin was demonstrated on the cold side during qualification thermal cycle testing as indicated.

There is no loss of valve integrity from performing the acceptance test with a minimum temperature level that did not provide the required 20 degrees F temperature margin beyond the qualification test level. The on–orbit operating temperature environment for the 1.75–inch Three–Way Valve is 55 degrees F to109 degrees F. The on–orbit non–operating temperature environment is 40 degrees F to 125 degrees F. The seal material is silicon (S614–80, AMS 3305) with a temperature performance rating of –85 degrees F to 300 degrees F, encompassing both the as–tested and predicted on–orbit operating temperatures.

Additionally, the 0.125–inch Three–Way Valves (Part Numbers B40204–11 and B40402–12) were successfully qualification thermal tested at –60 degrees F and acceptance thermal tested at –40 degrees F. The 1.75–inch Three–Way Valve, having a common valve design and materials, could be qualified by similarity. Both are made by the same supplier, use the exact same EEE parts (i.e.; diodes, position sensor switches, and wires), same type solenoid motors, same material in their Aluminum bodies, and same type of Silicon seals. The valve design is also the same; a bellow closes the circular sealing surface controlled by a central shaft.

The valves have an extremely low duty cycle and a very high MTBF. The Three–Way valves are normally unpowered and remain in one open position or another (the valve does not have a closed position) to allow continuous air flow through the duct systems. When the valve is cycled, it is powered for 100 milliseconds only. If the valve were to fail electronic actuation, it can be manually configured.

Finally, the 1.75–inch Three–Way Valve cycle operation is not critical at 10 degrees F or 20 degrees F, since the minimum operating temperature environment is 55 degrees F.

PG3–153:

ITEM:

1.75–inch Three–Way Valve Part Number B40205–1 (Boeing Part Number 683–13024)

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.

Paragraph 4.2.3.3C, Test Levels and Duration. The transitions shall be at a rate no less than 1.0 degree F (0.56 degree C).

EXCEPTION:

The transition ramp rate during the qualification thermal cycle test of the 1.75–inch Three–Way Valve has been verified by analysis as the thermocouple mounted on the unit under test was monitored without recording of the actual test data.

RATIONALE:

By comparison to a very conservative similar thermal cycling test where both chamber and hardware ramp rates were monitored, the Fluid Systems Servicer (FSS), using the General Equation for Transient Thermal response:

$\Delta T/\Delta t = Q/mc_p$, where Q = heat input to chamber = BTU/min

Then, even for similar chamber ramp rates
$(Q/mc_p)_{Valve} >> (Q/mc_p)_{FSS}$

mass = $(m)_{Valve}$ = 24 lbs, heat capacity of Aluminum = $(c_p)_{Valve}$ = .22
mass = $(m)_{FSS}$ = 114 lbs, heat capacity of Titanium = $(c_p)_{FSS}$ = .15

Intuitively and by analysis of physical law, smaller thermal mass sweeps faster for a given rate of change in background temperature.

The valves are much smaller than the FSS
The valve material, Al, has slightly higher C_p or heat capacity
The chamber the valves were tested in was much smaller
And, the chamber ramp rate was much faster for the valves: 15–30 degrees/min vs. 3–4 degrees/min for the FSS

Therefore, the valve ramp rate necessarily exceeded that of the FSS. Since the FSS hardware met the requirement, at 1.4 degrees/minute, the valves must also have met the requirement.

PG3–154:

ITEM:

1.75–inch Three–Way Valve Part Number B40205–1 (Boeing Part Number 683–13024)

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.

Paragraph 5.1.3.3C, Test Levels and Duration. The transitions shall be at a rate no less than 1.0 degree F (0.56 degree C).

EXCEPTION:

The transition ramp rate during the acceptance thermal cycle test of the 1.75–inch Three–Way Valve has been verified by analysis as the thermocouple mounted on the unit under test was monitored without recording of the actual test data.

RATIONALE:

By comparison to a very conservative similar thermal cycling test where both chamber and hardware ramp rates were monitored, the Fluid Systems Servicer (FSS), using the General Equation for Transient Thermal response:

$\Delta T/\Delta t = Q/mc_p$, where Q = heat input to chamber = BTU/min

Then, even for similar chamber ramp rates
$(Q/mc_p)_{Valve} >> (Q/mc_p)_{FSS}$

mass = $(m)_{Valve}$ = 24 lbs, heat capacity of Aluminum = $(c_p)_{Valve}$ = .22
mass = $(m)_{FSS}$ = 114 lbs, heat capacity of Titanium = $(c_p)_{FSS}$ = .15

Intuitively and by analysis of physical law, smaller thermal mass sweeps faster for a given rate of change in background temperature.

The valves are much smaller than the FSS
The valve material, Al, has slightly higher C_p or heat capacity
The chamber the valves were tested in was much smaller
And, the chamber ramp rate was much faster for the valves: 15–30 degrees/min vs. 3–4 degrees/min for the FSS

Therefore, the valve ramp rate necessarily exceeded that of the FSS. Since the FSS hardware met the requirement, at 1.4 degrees/minute, the valves must also have met the requirement.

PG3–155:

ITEM:

1.75–Inch Three–Way Valve Part Number B40205–1 (Boeing Part Number 683–13024)

SSP 41172 REQUIREMENT:

Paragraph 4.2.11.3F, Leakage Test, Component Qualification, Test Duration. Method VI. The duration of the test shall be no less than 60 minutes.

EXCEPTION:

The duration of the test for the 1.75–Inch valve shall be until the leak rate stabilizes (less than or equal to 30 minutes).

RATIONALE:

The design leakage requirement for the 1.75–Inch valve is less than 1 x 10^{-3} scc/second GHe. The valve leakage rate values predicted in the verification analysis are 2 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-4} scc/second GHe for internal leakage. The results from the qualification leak tests (3 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-5} scc/second GHe for internal leakage) are consistent with the predicted values from analysis. Likewise, the average results from the acceptance tests (6 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-5} scc/second GHe for internal leakage) are consistent with the predicted value from analysis. The verification analysis, qualification test, and acceptance test leakage results are all well below the valve requirement by a factor of 10 to 100.

The procedure that was used for this test included observing the leakage rate with a calibrated helium leak detector until the rate stabilized and then recording the stable leakage rate. This method follows the ASME standards. Leak test duration time depends on the exposure time that allows the seals to fully permeate with helium. The test duration can only be determined real time by reviewing the mass spectrometer readings, since seal permeation time depends on seal and body material, length, diameter, temperature, and pressure factors. Test experience has observed that some seals will permeate in 10 minutes, while others may take 3 to 4 hours. For the 1.75–Inch Valve, the average times for the leak rate to stabilize during acceptance tests on the flight units (20 minutes for external leakage, and 23 minutes for internal leakage) demonstrated repeatability with the stabilization times of 25 minutes observed during the qualification test. Any variation in leakage rate after the rate stabilizes and the required method VI duration of 60 minutes is negligible with respect to the total allowable leakage of cabin air from the USL Module.

ISS leak test experts concur that the method used on these items is valid at only a 30 minute or less duration and meets the intent of the SSP 41172 requirement.

PG3–156:

ITEM:

1.75–Inch Three–Way Valve Part Number B40205–1 (Boeing Part Number 683–13024)

SSP 41172 REQUIREMENT:

Paragraph 5.1.7.3F, Leakage Test, Component Acceptance, Test Duration. Method VI. The duration of the test shall be no less than 60 minutes.

EXCEPTION:

The duration of the test for the 1.75–Inch valve shall be until the leak rate stabilizes (less than or equal to 30 minutes).

RATIONALE:

The design leakage requirement for the 1.75–Inch valve is less than 1 x 10^{-3} scc/second GHe. The valve leakage rate values predicted in the verification analysis are 2 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-4} scc/second GHe for internal leakage. The results from the qualification leak tests (3 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-5} scc/second GHe for internal leakage) are consistent with the predicted values from analysis. Likewise, the average results from the acceptance tests (6 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-5} scc/second GHe for internal leakage) are consistent with the predicted value from analysis. The verification analysis, qualification test, and acceptance test leakage results are all well below the valve requirement by a factor of 10 to 100.

The procedure that was used for this test included observing the leakage rate with a calibrated helium leak detector until the rate stabilized and then recording the stable leakage rate. This method follows the ASME standards. Leak test duration time depends on the exposure time that allows the seals to fully permeate with helium. The test duration can only be determined real time by reviewing the mass spectrometer readings, since seal permeation time depends on seal and body material, length, diameter, temperature, and pressure factors. Test experience has observed that some seals will permeate in 10 minutes, while others may take 3 to 4 hours. For the 1.75–Inch Valve, the average times for the leak rate to stabilize during acceptance tests on the flight units (20 minutes for external leakage, and 23 minutes for internal leakage) demonstrated repeatability with the stabilization times of 25 minutes observed during the qualification test. Any variation in leakage rate after the rate stabilizes and the required method VI duration of 60 minutes is negligible with respect to the total allowable leakage of cabin air from the USL Module.

ISS leak test experts concur that the method used on these items is valid at only a 30 minute or less duration and meets the intent of the SSP 41172 requirement.

PG3–157:

ITEM:

0.5–Inch Valve, Solenoid Dual Coil Part Numbers B40202–1 and B40202–2

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.

Paragraph 4.2.3.3B. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The Qualification Thermal Cycle Test maximum temperature for the 0.5–inch Solenoid Dual Coil Valve will be 150 degrees F. This does not envelope the maximum acceptance thermal cycle test temperature of 160 degrees F for the following serial numbers:
B40202–1: 001001, 001002, 001003, 001004, and 001005;
B40202–2: 001001, 001002, 001003, and 001004.

RATIONALE:

The initial qualification temperature cycle test for the 0.5–Inch Valve was performed at a maximum temperature of 180 degrees F. However, as indicated in Failure Analysis 12085–002 (Appendix IX of Qualification Test Report VTR–12085), due to tolerance stack–up, the qualification valve would not close at this temperature. Analysis indicated that the qualification valve would not respond to electronic command to close the valve until the temperature was dropped to 165 degrees F. The qualification unit did function at 165 degrees F. The qualification test parameters were revised to conduct remaining testing at 150 degrees F during the hot portion of the cycle. However, this does not envelope the acceptance thermal cycle test performed at a maximum temperature of 160 degrees F on 5 CO2 Vent (Type –1) Valves and 4 H2O Vent (Type–2) Valves.

There is no loss of valve material integrity from performing the acceptance test with a maximum temperature level that did not provide the required 20 degrees F temperature margin below the qualification test level. The maximum on–orbit operating temperature environment for the 0.5–inch Valve is 109 degrees F. The maximum on–orbit non–operating temperature is 125 degrees F. The seal material is silicon (S614–80, AMS 3305) with a temperature performance rating of –85 to 300 degrees F, encompassing both the as–tested and predicted on–orbit operating temperature.

The valves have an extremely low duty cycle and a very high MTBF. The CO2 Vent valves are normally unpowered and remain open to allow continuous vent of CO2 from the AR rack. The H2O Vent valves are normally unpowered and remain closed, except for the non–normal vent of wastewater. The valves are used in series to provide function redundancy and a have a manual override capability so that if the valve were to fail electronic actuation, it can be manually configured. Finally, there are two water dump vent lines, which provides another path for the water vent function.

PG3–158:

ITEM:

0.5–Inch Valve, Solenoid Dual Coil Part Number B40202–2

SSP 41172 REQUIREMENT:

Paragraph 4.2.10, Ultimate Pressure Component Qualification.

Paragraph 4.2.10.3C, Ultimate Pressure for Valves. Ultimate pressure shall be as specified in SSP 30559, section 3. (2.5 x MDP)

EXCEPTION:

Ultimate pressure for the H2O Vent Valve shall verified by analysis to the required 2.5 times the Maximum Operation Pressure (MOP).

RATIONALE:

Qualification testing for both B40202–1 and B40202–2, was performed with a CO2 vent valve (B40202–1) which has an aluminum valve body. The H2O vent valve (B40202–2) has the identical design and internal components with a titanium valve body. The B40202–2 valve was qualified by similarity with the –1 qualification unit; no –2 qualification unit was manufactured.

The B40202–1 part has a maximum design pressure of 15.2 psia; the qualification unit was successfully proof pressure tested at 30.4 psig (2 times MDP) and burst pressure tested at 70 psig (4 times MDP). The SSP 30559 Table 3.3.1–1C.3d Minimum Factor of Safety for Proof Pressure Test of pneumatic and hydraulic valves is 1.50 x MDP and Ultimate Pressure is 2.5 x MDP.

The B40202–2 MDP requirement was initially 60 psig and was later, after the valves were built, revised to 85 psig, when the JEM determined a failure to its pump could result in the higher MDP.

All four installed B40202–2 flight units were proof pressure acceptance tested at 123 psig, which was based on the original MDP requirement with a safety factor of 2.0. The B40202–2 valve acceptance testing provides a safety factor of 1.45 over the MDP requirement of 85 psig, which is within tolerance for SSP 41172 requirement of 1.5 x MDP. The acceptance proof pressure testing for the –2 valve satisfies the qualification proof test requirement with the minimum of one pressure cycle at minimum safety factor.

The original design burst analysis was done for the B40202–2 valves at 4 x MDP of 60 psig, which is 240 psig. The required ultimate test/analysis is only 212.5 psig. There fore, the original analysis encompasses this level.

The B40202–2 part nominal waste water dump operating pressure range is 0 to 8 psig, which makes ultimate pressure analysis of 2.5 x MDP, for the unlikely failure scenario which could result in the 85 psig condition, a logical exception for verification.

PG3–159:

ITEM:

0.5–Inch Valve, Solenoid Dual Coil Part Number B40202–1 and B40202–2

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification.

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The 0.5–Inch Valve component will not be powered on and monitored for failures or intermittences during the random vibration test.

RATIONALE:

The 0.5–inch valve is a simple solenoid actuator device and does not contain any electronic circuitry. The valve is controlled by a remote computer. The solenoid has a 100–millisecond stroke, which makes detection of a fault or intermittent failure highly unlikely. The valves only operate on–orbit; it is not required to perform under launch or landing environments. The valve is normally unpowered except to change the valve position.

The valves have an extremely low duty cycle and a very high MTBF. The CO2 Vent valves are normally unpowered and remain open to allow continuous vent of CO2 from the AR rack. The H2O Vent valves are normally unpowered and remain closed, except for the non–normal vent of wastewater. The valves are used in series to provide functional redundancy and a have a manual override capability so that if the valve were to fail electronic actuation, it can be manually configured. Finally, there are two water dump vent lines, which provides another path for the water vent function.

PG3–160:

ITEMS:

0.5–Inch Valve, Solenoid Dual Coil Part Number B40202–1, Serial Numbers 001001, 001002, 001003, 001004, and 001005

0.5–Inch Valve, Solenoid Dual Coil Part Number B40202–2, Serial Numbers 001001, 001002, 001003, and 001004

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance.
Paragraph 5.1.4.4, Random Vibration, Component Acceptance, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The 0.5–Inch Valve component will not be powered on and monitored for failures or intermittences during the random vibration test.

RATIONALE:

The 0.5–inch valve is a simple solenoid actuator device and does not contain any electronic circuitry. The valve is controlled by a remote computer. The solenoid has a 100–millisecond stroke, which makes detection of a fault or intermittent failure highly unlikely. The valves only operate on–orbit; it is not required to perform under launch or landing environments. The valve is normally unpowered except to change the valve position.

The valves have an extremely low duty cycle and a very high MTBF. The CO2 Vent valves are normally unpowered and remain open to allow continuous vent of CO2 from the AR rack. The H2O Vent valves are normally unpowered and remain closed, except for the non–normal vent of wastewater. The valves are used in series to provide functional redundancy and a have a manual override capability so that if the valve were to fail electronic actuation, it can be manually configured. Finally, there are two water dump vent lines, which provides another path for the water vent function.

PG3–161:

ITEM:

0.5–Inch Valve, Solenoid Dual Coil Part Number B40202–1 and B40202–2

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.

Paragraph 4.2.3.3C. The temperature transitions shall be at a rate no less than 1.0–degree F (0.56 degree C) per minute.

EXCEPTION:

The verification of temperature transition compliance is verified by analysis of compliant chamber ramp rate data instead of record of temperature sensors during temperature transitions.

RATIONALE:

The valve was instrumented for visual readings of Unit Under Test temperatures; however, continuous monitoring and recording of the temperatures did not occur. Yet, due to the small mass of the unit and rapid chamber temperature ramp rates, it is reasonable to expect that a 1.0–degree F per minute ramp rate requirement is met by the unit under test.

Data verifying the test chamber ramp rate used for the thermal cycling test was obtained from a calibrated temp sensor. These rates were presented in the test report.

By comparison of the chamber ramp rates of both the 0.5–inch valve under discussion and another ORU during a conservatively similar thermal cycling test, analysis proves that the temperature ramp rate experienced by the 0.5–inch valve must have exceeded the 1.0–degree F/minute requirement.

Consider the conservatively similar thermal cycling test data of the Fluid Systems Servicer (FSS) as compared to the known chamber temperature sweep rate of the 0.5–inch Valve and other ValveTech valves.

Using the General Equation for Transient Thermal response:
$\Delta T/\Delta t = Q/mc_p$, where Q = heat input to chamber = BTU/min

Then, even for similar chamber ramp rates
$(Q/mc_p)_{Valve} >> (Q/mc_p)_{FSS}$

mass = $(m)_{Valve}$ = 24 lbs, heat capacity of Aluminum = $(c_p)_{Valve}$ = .22
mass = $(m)_{FSS}$ = 114 lbs, heat capacity of Titanium = $(c_p)_{FSS}$ = .15

Intuitively, a smaller thermal mass sweeps faster for a given rate of change in background temperature.

The valves are much smaller than the FSS
The valve material, Al, has slightly higher C_p or heat capacity than the Ti FSS
The chamber in which the valves were tested was much smaller than the large chamber used for the FSS
The chamber ramp rate was much faster for the valves in the smaller chamber

Therefore, the valve ramp rate necessarily exceeded that of the FSS. Since the FSS met the requirement, at 1.3 degrees F per minute, the 0.5–inch Valves must also have met the requirement.

PG3–162:

ITEM:

0.5–Inch Valve, Solenoid Dual Coil Part Number B40202–1 and B40202–2

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3C. The temperature transitions shall be at a rate no less than 1.0–degree F (0.56 degree C) per minute.

EXCEPTION:

The verification of temperature transition compliance is verified by analysis of compliant chamber ramp rate data instead of record of temperature sensors during temperature transitions.

RATIONALE:

The valve was instrumented for visual readings of Unit Under Test temperatures; however, continuous monitoring and recording of the temperatures did not occur. Yet, due to the small mass of the unit and rapid chamber temperature ramp rates, it is reasonable to expect that a 1.0–degree F per minute ramp rate requirement is met by the unit under test.

Data verifying the test chamber ramp rate used for the thermal cycling test was obtained from a calibrated temp sensor. These rates were presented in the test report.

By comparison of the chamber ramp rates of both the 0.5–inch valve under discussion and another ORU during a conservatively similar thermal cycling test, analysis proves that the temperature ramp rate experienced by the 0.5–inch valve must have exceeded the 1.0–degree F/minute requirement.

Consider the conservatively similar thermal cycling test data of the Fluid Systems Servicer (FSS) as compared to the known chamber temperature sweep rate of the 0.5–inch Valve and other ValveTech valves.

Using the General Equation for Transient Thermal response:
$\Delta T/\Delta t = Q/mc_p$, where Q = heat input to chamber = BTU/min

Then, even for similar chamber ramp rates
$(Q/mc_p)_{Valve} >> (Q/mc_p)_{FSS}$

mass = $(m)_{Valve}$ = 24 lbs, heat capacity of Aluminum = $(c_p)_{Valve}$ = .22
mass = $(m)_{FSS}$ = 114 lbs, heat capacity of Titanium = $(c_p)_{FSS}$ = .15

Intuitively, a smaller thermal mass sweeps faster for a given rate of change in background temperature.

The valves are much smaller than the FSS
The valve material, Al, has slightly higher C_p or heat capacity than the Ti FSS
The chamber in which the valves were tested was much smaller than the large chamber used for the FSS
The chamber ramp rate was much faster for the valves in the smaller chamber

Therefore, the valve ramp rate necessarily exceeded that of the FSS. Since the FSS met the requirement, at 1.3 degrees F per minute, the 0.5–inch Valves must also have met the requirement.

PG3–163:

ITEM:

0.5–Inch Valve, Solenoid Dual Coil Part Number B40202–1 and B40202–2

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.

Paragraph 4.2.11.3F, Test Duration. Method VI. The duration of the test shall be no less than 60 minutes.

EXCEPTION:

The duration of the test for the 0.5–Inch valve shall be until the leak rate stabilizes (less than or equal to 30 minutes).

RATIONALE:

The design leakage requirement for the 0.5–Inch valve is less than 1 x 10^{-3} scc/second GHe. The valve leakage rate values predicted in the verification analysis are 2 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-4} scc/second GHe for internal leakage. The results from the qualification leak tests (3 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-5} scc/second GHe for internal leakage) are consistent with the predicted values from analysis. Likewise, the average results from the acceptance tests (6 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-5} scc/second GHe for internal leakage) are consistent with the predicted value from analysis. The verification analysis, qualification test, and acceptance test leakage results are all well below the valve requirement by a factor of 10 to 100.

The procedure that was used for this test included observing the leakage rate with a calibrated helium leak detector until the rate stabilized and then recording the stable leakage rate. This method follows the ASME standards. Leak test duration time depends on the exposure time that allows the seals to fully permeate with helium. The test duration can only be determined real time by reviewing the mass spectrometer readings, since seal permeation time depends on seal and body material, length, diameter, temperature and pressure factors. Test experience has observed that some seals will permeate in 10 minutes, while others may take 3 to 4 hours. For the 0.5–Inch Valve, the average times for the leak rate to stabilize during acceptance tests on the flight units (20 minutes for external leakage, and 23 minutes for internal leakage) demonstrated repeatability with the stabilization times of 25 minutes observed during the qualification test. Any variation in leakage rate after the rate stabilizes and the required method VI duration of 60 minutes is negligible with respect to the total allowable leakage of cabin air from the USL Module.

ISS leak test experts concur that the method used on these items is valid at only a 30 minute or less duration and meets the intent of the SSP 41172 requirement.

PG3–164:

ITEM:

0.5–Inch Valve, Solenoid Dual Coil Part Number B40202–1 and B40202–2

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.

Paragraph 5.1.7.3F, Test Duration. Method VI. The duration of the test shall be no less than 60 minutes.

EXCEPTION:

The duration of the test for the 0.5–Inch valve shall be until the leak rate stabilizes (less than or equal to 30 minutes).

RATIONALE:

The design leakage requirement for the 0.5–Inch valve is less than 1 x 10^{-3} scc/second GHe. The valve leakage rate values predicted in the verification analysis are 2 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-4} scc/second GHe for internal leakage. The results from the qualification leak tests (3 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-5} scc/second GHe for internal leakage) are consistent with the predicted values from analysis. Likewise, the average results from the acceptance tests (6 x 10^{-5} scc/second GHe for external leakage, and 2 x 10^{-5} scc/second GHe for internal leakage) are consistent with the predicted value from analysis. The verification analysis, qualification test, and acceptance test leakage results are all well below the valve requirement by a factor of 10 to 100.

The procedure that was used for this test included observing the leakage rate with a calibrated helium leak detector until the rate stabilized and then recording the stable leakage rate. This method follows the ASME standards. Leak test duration time depends on the exposure time that allows the seals to fully permeate with helium. The test duration can only be determined real time by reviewing the mass spectrometer readings, since seal permeation time depends on seal and body material, length, diameter, temperature and pressure factors. Test experience has observed that some seals will permeate in 10 minutes, while others may take 3 to 4 hours. For the 0.5–Inch Valve, the average times for the leak rate to stabilize during acceptance tests on the flight units (20 minutes for external leakage, and 23 minutes for internal leakage) demonstrated repeatability with the stabilization times of 25 minutes observed during the qualification test. Any variation in leakage rate after the rate stabilizes and the required method VI duration of 60 minutes is negligible with respect to the total allowable leakage of cabin air from the USL Module.

ISS leak test experts concur that the method used on these items is valid at only a 30 minute or less duration and meets the intent of the SSP 41172 requirement.

PG3–165:

ITEM:

Carbon Dioxide Removal Assembly, Part Number 2352630–1–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…” This requires that the component be powered off at the minimum and maximum temperature extremes during Protoflight Thermal Cycle testing in accordance with paragraph 4.2.3.2.

EXCEPTION:

The Carbon Dioxide Removal Assembly shall not be powered off at temperature extremes during protoflight thermal cycle testing.

RATIONALE:

The CDRA has a complex sequence of events during the transient start up phase. This sequence is software controlled and a serial operation of events is critical for the unit to meet performance requirements. Operating in off–nominal conditions poses risks that can result in damage to the unit. Powering off during this transient period would result in an excess of water in the air exchange beds, air pump, and blower. This operation in an off–nominal condition was shown to damage during Engineering development.

This test was timed so that the transitions between CDRA half–cycles occurred only after low temperature or high temperature dwells, allowing the selector valves to be powered on and off at these extremes. Already, flight rules exist that preclude shut down at temperature environments where condensation can occur. Thus, the risks associated with off–nominal operation are much greater than any value added by testing under condition of power cycling.

Test procedures will be updated to demonstrate the ability to start the unit at the hot and cold thermal extremes by ramping the unpowered unit to each thermal extreme, stabilizing at the specified level, and executing startup procedures.

PG3–166:

ITEM:

Carbon Dioxide Removal Assembly, Part Number 2352630–1–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same …" This requires Protoflight Random Vibration testing in accordance with paragraph 5.1.4.

EXCEPTION:

The Protoflight Random Vibration test on the Carbon Dioxide Removal Assembly shall be performed at a level of 4.3 grms with a maximum spectral density of .04 g2/Hz in all axes.

RATIONALE:

The CDRA is an electro–mechanical assembly consisting of a complex interconnected thermal transfer devises and electronics. These devises utilize cooling water from the Thermal Control System. There are seven major electrical/electronic assemblies on the CDRA. These are the Selector Valve Motor Controller, Pump Fan Motor Controller, Heater Controller, Blower, Pump, Absolute Pressure Sensor, and Differential Pressure Sensor.

The CDRA components have experienced a random vibration level of 6.1 grms for a duration of one minute at high and low temperature extremes in the critical axis during Reliability Acceptance Testing. The critical axis is normal to the plane of the circuit cards. The CDRA pressure sensors each went through additional component–level random vibration testing. This was done to a level of 3.1 grms for one minute in each of the two remaining axes. The CDRA did complete both a leak test and a full functional test after vibration.

Finally, only connectors meeting the Space Station Quality standards are used to integrate the electrical sub–assemblies. Locking features are use on fasteners and fastener heads are secured with RTV. All integrated water connections are either Symmetrics type Quick Disconnects or gamma fittings.

PG3–167:

ITEM:

Carbon Dioxide Removal Assembly, Part Number 2352630–1–1

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same …” This requires that the component be powered on and monitored for failures or intermittences during random vibration testing in accordance with paragraph 4.2.5.4.

EXCEPTION:

The CDRA shall not be powered on and monitored for failures or intermittences during the protoflight random vibration test.

RATIONALE:

The Carbon Dioxide Removal Assembly contains electronics consisting of an electro–mechanical actuator, does not utilize a printed circuit board with a card edge connector, or have high density populated circuit cards. Circuit Card Assemblies (CCA) are mounted at all four corners and contain a central mounting point, with standoffs between CCAs, to limit deflections due to vibrational loads. A limited portion of the components would be energized, as the Carbon Dioxide Removal Assembly is in standby mode. This is a non–enabled, non–operating condition, where minimal components, including mainly the position indication signals are powered. The only parameter available for monitoring would be input current at standby condition. The Carbon Dioxide Removal Assembly would not be operational during the vibration test. The Carbon Dioxide Removal Assembly would only be in standby mode. Also, electronic components are contained within as solder sealed cover, and are not accessible for inspection at any point during or after vibration test. The Carbon Dioxide Removal Assembly is not normally powered during launch vibrational loads, and to do this takes the unit out of its normal operating functional mode.

Additionally, reliability acceptance testing in a non–electrically energized condition was performed at a lower level assembly, which included random vibration testing for one minute in the maximum deflection axis at temperature extremes. This provides further workmanship screening on the electronics.

Thus, the consumption of additional fatigue life of the CDRA protoflight unit to repeat protoflight random vibration test outweighs the benefit of any additional random vibration testing with power on and monitoring for workmanship concerns.

PG3–168:

ITEM:

Trace Contaminant Control Subassembly (TCCS) Part Number 5823550–501

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…” This requires that electrical and electronic components be powered on and monitored for failures or intermittences during the protoflight random vibration test in accordance with paragraph 4.2.5.4.

EXCEPTION:

The TCCS protoflight unit will not be powered on and monitored for failures or intermittences during the protoflight random vibration test.

RATIONALE:

The TCCS contains only a minimal amount of electronic circuitry in the Electrical Interface Assembly, the blower, and the flow meter ORUs. These electronic components are designed to reduce the card–flexing concerns that can cause intermittencies during random vibration. The TCCS electronic components design includes: small circuit card assembly (about 3 x 5 inches), no high–profile components, low density packaging, staked or potted components, cards fastened at corners, and no card–edge connections. The TCCS is controlled by a remote MDM computer. The TCCS only operates on–orbit; there are no launch or landing operations.

The TCCS has a moderate duty cycle (0.5) and the electrical components have calculated MTBF values of over 100,000 hours. If a TCCS electrical component ORU fails, it will be replaced on–orbit with a spare ORU. Also, the Russian Air Revitalization in the Service Module provides redundant capability to remove trace contaminants to support a crew size of three. After Node 3 (Flight 20A) and Russian Universal Docking Module (Flight 3R) arrive, there are four redundant trace contaminant systems in the station of which any two can support a crew size of six. The charcoal filter in Node 1 also has the capability to remove trace contaminants and is used during assembly operation of the USL before the Air Revitalization rack is moved to its operating location and activated.

A functional test was successfully performed on each of the TCCS electrical ORUs after the individual component random vibration workmanship screening was performed with power off. Another functional test of the integrated TCCS was also performed after the powered–off random vibration test of the entire TCCS assembly. All of the TCCS electrical ORUs successfully demonstrated power–on performance during their individual thermal cycle workmanship screening tests. Power–on random vibration operational performance was later demonstrated with the third Electrical Interface Assembly, blower, and flow meter spares, while they were installed in the TCCS mass simulator assembly. Any electrical ORUs that are repaired or have follow–on production will have random vibration test conducted with power on while installed in the TCCS mass simulator. Thus, the additional fatigue on the TCCS protoflight unit outweighs the benefit of additional random vibration testing.

PG3–169:

ITEM:

Trace Contaminant Control Subassembly (TCCS) Part Number 5823550–501

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. ”When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same...”This requires that the protoflight hardware Random Vibration test levels and spectrum envelope maximum predicted flight level and spectrum minus 6 dB but not less than the workmanship screening level of 6.1 grms as defined in SSP 41172.

EXCEPTION:

The TCCS protoflight unit random vibration test will be performed to screening levels and spectrums as follows:

Frequency Range (Hz)	Power Spectral Density
20	0.01 g^2/Hz
20–70	3.3 dB/Octave
70–200	0.04 g^2/Hz
200–2000	– 4.0 dB/Octave
2000	0.002 g^2/Hz
Overall	4.3 grms

RATIONALE:

The three critical electrical ORU components (Electrical Interface Assembly, blower, and flow meter) each individually completed random vibration testing in all three axis at the required workmanship screening level and spectrum defined in Figure 5–2 (6.1 grms overall) prior to their integration into the TCCS protoflight unit. After the components were integrated into the TCCS, the protoflight assembly completed further random vibration testing with the screening level and spectrum (4.3 grms overall) defined in this exception. This spectrum is 3 dB higher than the TCCS launch and landing random vibration design requirement, and complies with the SSP 41172 Figure 5–2 levels and spectrums up to 200 Hertz.

The TCCS has a moderate duty cycle (0.5) and the electrical components have calculated MTBF values of over 100,000 hours. If a TCCS electrical component ORU fails, it will be replaced on–orbit with a spare ORU. Also, the Russian Air Revitalization in the Service Module provides redundant capability to remove trace contaminants to support a crew size of three. After Node 3 (Flight 20A) and Russian Universal Docking Module (Flight 3R) arrive, there are four redundant trace contaminant systems in the station of which any two can support a crew size of six. The charcoal filter in Node 1 also has the capability to remove trace contaminants and is used during assembly operation of the USL before the Air Revitalization rack is moved to its operating location and activated.

The spectrum that was used for TCCS assembly–level random vibration testing provides adequate screening for structural and mechanical integrity of the protoflight unit, since the critical electrical ORU components completed separate workmanship screening at the SSP 41172 spectrum prior to integration into the TCCS assembly. Additionally, the random vibration response data for the critical electrical Electrical Interface Assembly and Blower ORU components at their installed locations on the TCCS protoflight unit showed excitation levels (up to 6.2 grms overall) that were higher than the random vibration level (4.3 grms overall) on the entire TCCS assembly. Thus, the additional fatigue on the TCCS protoflight unit outweighs the benefit of additional random vibration testing.

PG3–170:

ITEM:

Trace Contaminant Control Subassembly Electrical/Electronic ORU Components

Electrical Interface Assembly	Part Number 5835398–503
Blower Assembly	Part Number 5835404–501
Flow Meter Assembly	Part Number 5835405–501

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same..." This requires at least 100 degrees F (55.6 degrees C) protoflight thermal sweep between the minimum and maximum test temperatures, and the minimum test temperature be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The temperature sweep performed during protoflight thermal cycling will be 66 degrees F (53 degrees F to 119 degrees F).

RATIONALE:

Along with the operating Protoflight Thermal Cycle test between 53 degrees F minimum and 119 degrees F maximum temperatures, the TCCS critical electrical components completed functional tests after experiencing non–operating thermal workmanship screening at 5 degrees F minimum and 160 degrees F maximum temperatures. Thermal cycle testing was performed over 24 cycles for both operational and non–operational testing, which is 3 times the protoflight thermal cycle requirement of 8 cycles.

Along with the additional thermal cycles performed, over 500 hours of operating time has been accumulated during supplier component/assembly and Boeing integrated testing. The additional thermal cycles and accumulated ground operating time on the protoflight equipment also helps reduce the risk of experiencing an early TCCS electronic component failure on–orbit from undetected latent defects in the electronic subassemblies.

The TCCS has a moderate duty cycle (0.5) and the electrical components have calculated MTBF values of over 100,000 hours. If a TCCS electrical component ORU fails, it will be replaced on–orbit with a spare ORU. Also, the Russian Air Revitalization in the Service Module provides redundant capability to remove trace contaminants to support a crew size of three. After Node 3 (Flight 20A) and Russian Universal Docking Module (Flight 3R) arrive, there are four redundant trace contaminant systems in the station of which any two can support a crew size of six. The charcoal filter in Node 1 also has the capability to remove trace contaminants and is used during assembly operation of the USL before the Air Revitalization rack is moved to its operating location and activated.

PG3–171:

ITEM:

Trace Contaminant Control System Part Number 5823550–501 Serial Number 0001

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.

Paragraph 6.1.1C. The duration of the test shall be limited to one minute.

EXCEPTION:

The duration of the protoflight vibration test in the z–axis for serial number 0001 shall be 95 seconds.

RATIONALE:

The test procedure for the Trace Contaminant Control System does specify a 60–second duration for each axis during protoflight random vibration tests. However, for serial number 0001 in the z–axis, the test was performed for 95 seconds.

Random Vibration Fatigue Evaluation Report TR001, Appendix H, shows total fatigue life accumulation is less than 5 percent of allowable fatigue life; that is, 95 percent of allowable fatigue/service life remains even with this additional time accumulation. This results from the z–axis random vibration test only accounts for 0.1 percent of the used service life. Thus, the overtest for duration is insignificant in view of the total available life remaining.

PG3–172:

ITEM:

Rack Flow Control Assembly, Part Number 2353180–1–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperature of the Rack Flow Control Assembly shall be 33 degrees F during qualification thermal cycle testing.

RATIONALE:

The minimum predicted operating temperature of the Rack Flow Control Assembly is 55 degrees F. Flight Rack Flow Control Assemblies underwent Acceptance Thermal Cycle testing while operating to a minimum temperature of 33 degrees F. However, the Qualification Rack Flow Control Assembly was tested while operating to the identical minimum temperature of 33 degrees F.

The Material Identification and Usage List shows that the mechanical portions of the Rack Flow Control Assembly assembly are insensitive to a non–operational low temperature of 0 degrees F. Also, the Rack Flow Control Assembly Valve Motor Controller did undergo operational testing to a minimum temperature of 0 degrees F. Finally, the EEE components of the Rack Flow Control Assembly Assembly are rated for temperatures below 0 degrees F.

The Rack Flow Control Assembly hardware is Criticality 1. The worst case is loss of coolant to Lab DDCUs that will shut down due to over–temperature resulting in loss of power distribution in the Lab, and the potential loss of crew and station critical functions. An operational workaround has been developed and documented to recover partial functionality. A workaround requires the crew jumpering loads to the low temperature loop. The workaround has been demonstrated and accepted by NASA as documented in JSC 48532–5A. For other potential failures, use of the manual override capability inherent in the Rack Flow Control Assembly would restore capability so long as fluid is present. In addition, a pre–positioned spare is available on–orbit.

The purpose of the 20–degree F qualification margin is to demonstrate that there is sufficient design margin such that the acceptance test to which the flight hardware is subjected does not excessively degrade the hardware's useful life. This cold temperature used for qualification and acceptance thermal cycle testing is sufficiently benign for space–quality hardware such that there is very little risk that acceptance testing at the same cold temperature as qualification (33 degrees F) excessively degrades the life of the hardware.

PG3–173:

ITEM:

Rack Flow Control Assembly, Part Number 2353180–1–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification.

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The Rack Flow Control Assembly will not be powered on and monitored during the qualification random vibration testing.

RATIONALE:

The Rack Flow Control Assembly contains electronics consisting of an electro–mechanical actuator, and does not utilize a printed circuit board with a card edge connector, or have high density populated circuit cards. CCAs are mounted at all four corners and contain a central mounting point, with standoffs between CCAs, to limit deflections due to vibrational loads. A limited portion of the components would be energized, as the Rack Flow Control Assembly is in standby mode. This is a non–enabled, non–operating condition, where minimal components, including mainly the position indication signals are powered. The only parameter available for monitoring would be input current at standby and "open" or "closed" position indication. The Rack Flow Control Assembly would not be operational during the vibration test. The Rack Flow Control Assembly would only be in standby mode. Also, electronic components are contained within a solder–sealed cover, and are not accessible for inspection at any point during or after vibration test. The Rack Flow Control Assembly is not normally powered during launch vibrational loads, and to do this takes the unit out of its normal operating functional mode.

Full functional testing was performed at temperature extremes after qualification random vibration testing, which does increase the likelihood of detecting any intermittent failures. Additionally, reliability acceptance testing in a non–electrically energized condition was performed at a lower level assembly, which included random vibration testing for one minute in the maximum deflection axis at temperature extremes. This provides further workmanship screening on the electronics.

In the event that there is any operational malfunction due to some factor that would have been screened out by power–on vibration testing of the unit, it should be noted that each of these components have manual override capability. Backing this up, should any component fail mechanically, there is sufficient redundancy in the system to ensure that the Thermal Control System loops are not shut down.

PG3–174:

ITEM:

Rack Flow Control Assembly, Part Number 2353180–1–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance.

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The Rack Flow Control Assembly will not be powered on and monitored during the acceptance random vibration testing.

RATIONALE:

The Rack Flow Control Assembly contains electronics consisting of an electro–mechanical actuator, and does not utilize a printed circuit board with a card edge connector, or have high density populated circuit cards. CCAs are mounted at all four corners and contain a central mounting point, with standoffs between CCAs, to limit deflections due to vibrational loads. A limited portion of the components would be energized, as the Rack Flow Control Assembly is in standby mode. This is a non–enabled, non–operating condition, where minimal components, including mainly the position indication signals are powered. The only parameter available for monitoring would be input current at standby and "open" or "closed" position indication. The Rack Flow Control Assembly would not be operational during the vibration test. The Rack Flow Control Assembly would only be in standby mode. Also, electronic components are contained within a solder–sealed cover, and are not accessible for inspection at any point during or after vibration test. The Rack Flow Control Assembly is not normally powered during launch vibrational loads, and to do this takes the unit out of its normal operating functional mode.

Full functional testing was performed at temperature extremes after acceptance random vibration testing, which does increase the likelihood of detecting any intermittent failures. Additionally, reliability acceptance testing in a non–electrically energized condition was performed at a lower level assembly, which included random vibration testing for one minute in the maximum deflection axis at temperature extremes. This provides further workmanship screening on the electronics.

In the event that there is any operational malfunction due to some factor that would have been screened out by power–on vibration testing of the unit, it should be noted that each of these components have manual override capability. Backing this up, should any component fail mechanically, there is sufficient redundancy in the system to ensure that the Thermal Control System loops are not shut down.

PG3–175:

ITEM:

Rack Flow Control Assembly, Part Number 2353180–1–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.

Paragraph 5.1.7.2, Test Description and Alternatives. One of the methods given in 4.2.11.2 shall be used.

4.2.11.2E. Method V. Leakage shall be detected using an appropriate method.

EXCEPTION:

The Soap Bubble method shall be an acceptable method under Method V. The duration of the test shall be no less than five minutes.

RATIONALE:

The Rack Flow Control Assembly underwent a Bubble Soap test for verification of external leakage during acceptance testing. Honeywell performs the Bubble Soap test on a bench top in a well–lighted, controlled laboratory environment at a maximum operating pressure of 90 psig with approximately 40–45 minutes required to stabilize. The valve body components are Dye Penetrant inspected after machining. The Rack Flow Control Assembly contains 3 welded joints that have nondestructive evaluation (X–Ray and Dye Penetrant) performed and 4 mechanical joints (actuator and sensor flange seals to assemble to the valve body). The QDs are qualified to 1.0E–04 scc/sec. Each joint and mechanical fitting is checked every time the bubble test is performed. The pass criteria is no bubbles visible in the 5–minute period, while reapplying soap solution as required. If bubbles are observed, the test fails and a nonconformance written.

The Bubble Soap method ensures an accuracy of 1.0E–04 scc/sec for each joint. Assuming the welded joints in the range of 1.0E–08 scc/sec range are negligible, that leaves 4 mechanical joints x 1.0E–04 per joint = 4.0E–04 scc/sec overall leak rate. The specification leakage requirement for the Rack Flow Control Assembly is 7.0E–04 scc/sec.

With an update to the Acceptance Test Procedure, future leakage testing will be performed by a SSP 41172–compliant pressure decay leak test method.

PG3–176:

ITEM:

Rack Flow Control Assembly, Part Number 2353180–1–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.

Paragraph 5.1.7.2, Test Description and Alternatives. The component leak checks shall be made before and after exposure to each environmental acceptance test.

EXCEPTION:

No Leakage Test will be performed between the Acceptance Vibration Test and the Acceptance Thermal Cycling Test, or at the end of environmental acceptance testing.

RATIONALE:

Upon integration into the USL, as part of a pressurized system checkout, the Rack Flow Control Assembly fittings were externally leak test verified with a Helium sniff check, as part of a USL Verification Objective 24732. The USL element test is conducted by pressurizing the entire coolant loop and using a calibrated Helium probe to 1.0E–08 scc/sec. Each joint is sniffed for leakage. If the background Helium concentration increases due to a gross leak in a valve body/joint, the Helium sniffer will detect. In addition, the valve bodies have a dye penetrant test to verify the integrity of the housings. In addition, the units in use in the USL have since undergone integration testing without notice of coolant fluid leak or indication from the Pump Package Assembly's quantity sensor of any coolant loss. On orbit, the quantity sensor will be used as an indication of any measurable coolant loss as a result of leakage. There are hardware and software detections to shutdown in the event of greater than 1 gallon of coolant loss.

With an update to the Acceptance Test Procedure, future leakage testing will be performed between environmental acceptance tests and at the completion of all environmental testing by a SSP 41172–compliant pressure decay leak test method.

PG3–177:

ITEM:

Rack Flow Control Assembly, Part Number 2353180–1–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.

Paragraph 4.2.11.2, Test Description and Alternatives. Component leak checks shall be made prior to initiation of, and following the completion of, component qualification thermal and vibration tests.

EXCEPTION:

No Leakage Test will be performed between the Qualification Vibration Test and the Qualification Thermal Cycling Test, or at the end of environmental qualification testing.

RATIONALE:

Though the lack of a fully–qualified Rack Flow Control Assembly design for leakage introduces risk, as–performed tests on the flight units do provide some confidence in the design. Upon integration into the USL, as part of a pressurized system checkout, the Rack Flow Control Assembly fittings were externally leak test verified with a Helium sniff check, as part of a USL Verification Objective. In addition, the units in use in the USL have since undergone integration testing, without notice of coolant fluid leak, or indications from the Pump Package Assembly's quantity sensor of any coolant loss.

The Program accepts the flight units installed without the need of additional qualification testing. On future flight units, via an update to the Acceptance Test Procedure, leakage testing will be performed between environmental acceptance tests and at the completion of all environmental testing by a SSP 41172–compliant pressure decay leak test method to mitigate the risks associated.

PG3–178:

ITEM:

System Flow Control Assembly, Part Number 2353190–101

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.

Paragraph 4.2.3.3.B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperature of the System Flow Control Assembly shall be 33 degrees F during qualification thermal cycle testing.

RATIONALE:

The minimum predicted operating temperature of the System Flow Control Assembly is 55 degrees F. Flight System Flow Control Assemblies underwent Acceptance Thermal Cycle testing while operating to a minimum temperature of 33 degrees F. However, the Qualification System Flow Control Assembly was tested while operating to the identical minimum temperature of 33 degrees F.

The MIUL shows that the mechanical portions of the System Flow Control Assembly assembly are insensitive to a non–operational low temperature of 0 degrees F. Also, the System Flow Control Assembly Valve Motor Controller did undergo operational testing to a minimum temperature of 0 degrees F. Finally, the EEE components of the System Flow Control Assembly Assembly are rated for temperatures below 0 degrees F.

The System Flow Control Assembly hardware is Criticality 1. The worst case is loss of coolant to Lab DDCUs that will shut down due to over–temperature resulting is loss of power distribution in the Lab, and the potential loss of crew and station critical functions. An operational workaround has been developed and documented to recover partial functionality. A workaround requires the crew jumpering loads to the Low Temperature Loop. The workaround has been demonstrated and accepted by NASA as documented in JSC 48532–5A. For other potential failures, use of the manual override capability inherent in the System Flow Control Assembly would restore capability so long as fluid is present. In addition, a pre–positioned spare is available on–orbit.

The purpose of the 20–degree F qualification margin is to demonstrate that there is sufficient design margin such that the acceptance test to which the flight hardware is subjected does not excessively degrade the hardware's useful life. This cold temperature used for qualification and acceptance thermal cycle testing is sufficiently benign for space–quality hardware such that there is very little risk that acceptance testing at the same cold temperature as qualification (33 degrees F) excessively degrades the life of the hardware.

PG3–179:

ITEM:

System Flow Control Assembly, Part Number 2353190–101

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification.

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The System Flow Control Assembly will not be powered on and monitored during the qualification random vibration testing.

RATIONALE:

The System Flow Control Assembly contains electronics consisting of an electro–mechanical actuator, does not utilize a printed circuit board with a card edge connector, or have high density populated circuit cards. CCAs are mounted at all four corners and contain a central mounting point, with standoffs between CCAs, to limit deflections due to vibrational loads. A limited portion of the components would be energized, as the System Flow Control Assembly is in standby mode. This is a non–enabled, non–operating condition, where minimal components, including mainly the position indication signals are powered. The only parameter available for monitoring would be input current at standby and “open” or “closed” position indication. The System Flow Control Assembly would not be operational during the vibration test. The System Flow Control Assembly would only be in standby mode. Also, electronic components are contained within a solder–sealed cover, and are not accessible for inspection at any point during or after vibration test. The System Flow Control Assembly is not normally powered during launch vibrational loads, and to do this takes the unit out of its normal operating functional mode.

Full functional testing was performed at temperature extremes after qualification random vibration testing, which does increase the likelihood of detecting any intermittent failures. Additionally, reliability acceptance testing in a non–electrically energized condition was performed at a lower level assembly, which included random vibration testing for one minute in the maximum deflection axis at temperature extremes. This provides further workmanship screening on the electronics.

In the event that there is any operational malfunction due to some factor that would have been screened out by power–on vibration testing of the unit, it should be noted that each of these components have manual override capability. Backing this up, should any component fail mechanically, there is sufficient redundancy in the system to ensure that the Thermal Control System loops are not shut down.

PG3–180:

ITEM:

System Flow Control Assembly, Part Number 2353190–101

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance.

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The System Flow Control Assembly will not be powered on and monitored during the acceptance random vibration testing.

RATIONALE:

The System Flow Control Assembly contains electronics consisting of an electro–mechanical actuator, and does not utilize a printed circuit board with a card edge connector, or have high density populated circuit cards. CCAs are mounted at all four corners and contain a central mounting point, with standoffs between CCAs, to limit deflections due to vibrational loads. A limited portion of the components would be energized, as the System Flow Control Assembly is in standby mode. This is a non–enabled, non–operating condition, where minimal components, including mainly the position indication signals are powered. The only parameter available for monitoring would be input current at standby and “open” or “closed” position indication. The System Flow Control Assembly would not be operational during the vibration test. The System Flow Control Assembly would only be in standby mode. Also, electronic components are contained within a solder–sealed cover, and are not accessible for inspection at any point during or after vibration test. The System Flow Control Assembly is not normally powered during launch vibrational loads, and to do this takes the unit out of its normal operating functional mode.

Full functional testing was performed at temperature extremes after acceptance random vibration testing, which does increase the likelihood of detecting any intermittent failures. Additionally, reliability acceptance testing in a non–electrically energized condition was performed at a lower level assembly, which included random vibration testing for one minute in the maximum deflection axis at temperature extremes. This provides further workmanship screening on the electronics.

In the event that there is any operational malfunction due to some factor that would have been screened out by power–on vibration testing of the unit, it should be noted that each of these components have manual override capability. Backing this up, should any component fail mechanically, there is sufficient redundancy in the system to ensure that the Thermal Control System loops are not shut down.

PG3–181:

ITEM:

System Flow Control Assembly, Part Number 2353190–101

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.

Paragraph 5.1.7.2, Test Description and Alternatives. One of the methods given in 4.2.11.2 shall be used.

4.2.11.2E. Method V. Leakage shall be detected using an appropriate method.

EXCEPTION:

The Bubble Soap method shall be an acceptable method under Method V. The duration of the test shall be no less than five minutes.

RATIONALE:

The System Flow Control Assembly underwent a Bubble Soap test for verification of external leakage during acceptance testing. Honeywell performs the Bubble Soap test on a bench top in a well–lighted, controlled laboratory environment at a maximum operating pressure of 90 psig with approximately 40–45 minutes required to stabilize. The valve body components are Dye Penetrant inspected after machining. The System Flow Control Assembly contains 10 welded joints that have nondestructive evaluation (X–Ray and Dye Penetrant) performed and 3 mechanical joints (actuator and sensor flange seals to assemble to the valve body). The QDs are qualified to 1.0E–04 scc/sec. Each joint and mechanical fitting is checked every time the bubble test is performed. The pass criteria is no bubbles visible in the 5–minute period, while reapplying soap solution as required. If bubbles are observed, the test fails and a nonconformance written.

The Bubble Soap method ensures an accuracy of 1.0E–04 scc/sec for each joint. Assuming the welded joints in the range of 1.0E–08 scc/sec range are negligible, that leaves 3 mechanical joints x 1.0E–04 per joint = 3.0E–04 scc/sec overall leak rate. The specification leakage requirement for the System Flow Control Assembly is 1.0E–03 scc/sec.

With an update to the Acceptance Test Procedure, future leakage testing will be performed by a SSP 41172–compliant pressure decay leak test method.

PG3–182:

ITEM:

System Flow Control Assembly, Part Number 2353190–101

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.

Paragraph 5.1.7.2, Test Description and Alternatives. The component leak checks shall be made before and after exposure to each environmental acceptance test.

EXCEPTION:

No Leakage Test will be performed between the Acceptance Vibration Test and the Acceptance Thermal Cycling Test, or at the end of environmental acceptance testing.

RATIONALE:

Upon integration into the USL, as part of a pressurized system checkout, the System Flow Control Assembly fittings were externally leak test verified with a Helium sniff check, as part of a USL Verification Objective 24732. The USL element test is conducted by pressurizing the entire coolant loop and using a calibrated Helium probe to 1.0E–08 scc/sec. Each joint is sniffed for leakage. If the background Helium concentration increases due to a gross leak in a valve body/joint, the Helium sniffer will detect. In addition, the valve bodies have a dye penetrant test to verify the integrity of the housings. In addition, the units in use in the USL have since undergone integration testing without notice of coolant fluid leak or indication from the Pump Package Assembly's quantity sensor of any coolant loss. On orbit, the quantity sensor will be used as an indication of any measurable coolant loss as a result of leakage. There are hardware and software detections to shutdown in the event of greater than 1 gallon of coolant loss.

With an update to the Acceptance Test Procedure, future leakage testing will be performed between environmental acceptance tests and at the completion of all environmental testing by a SSP 41172–compliant pressure decay leak test method.

PG3–183:

ITEM:

System Flow Control Assembly, Part Number 2353190–1–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.

Paragraph 4.2.11.2, Test Description and Alternatives. Component leak checks shall be made prior to initiation of, and following the completion of, component qualification thermal and vibration tests.

EXCEPTION:

No Leakage Test will be performed between the Qualification Vibration Test and the Qualification Thermal Cycling Test, or at the end of environmental qualification testing.

RATIONALE:

Though the lack of a fully–qualified System Flow Control Assembly design for leakage introduces risk, as–performed tests on the flight units do provide some confidence in the design. Upon integration into the USL, as part of a pressurized system checkout, the System Flow Control Assembly fittings were externally leak test verified with a Helium sniff check, as part of a USL Verification Objective. In addition, the units in use in the USL have since undergone integration testing, without notice of coolant fluid leak, or indications from the Pump Package Assembly's quantity sensor of any coolant loss.

The Program accepts the flight units installed without the need of additional qualification testing. On future flight units, via an update to the Acceptance Test Procedure, leakage testing will be performed between environmental acceptance tests and at the completion of all environmental testing by a SSP 41172–compliant pressure decay leak test method to mitigate the risks associated.

PG3–184:

ITEM:

Major Constituent Analyzer (MCA) Part Number 359800, Serial Numbers 0001 and 0002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…” This requires that each cycle have a one hour minimum dwell at the high and at the low temperature levels during which the article shall be turned off until the temperature stabilizes and then turned on in accordance with 4.2.3.3C.

EXCEPTION:

The MCA Protoflight 1 and 2 powered–off dwell during the cold portion of the protoflight thermal cycle testing shall be a minimum of 1 minute. Any subsequent thermal cycle tests on these units shall require testing in full compliance with SSP 41172.

RATIONALE:

The specified temperatures for the MCA during normal operations are 68 degrees F to 76 degrees F. The MCA was thermal cycle tested at operating temperatures from 53 degrees F to 90 degrees F and non–operating thermal cycle tested from 0 degrees F to 130 degrees F. The MCA has demonstrated cold start capability during integrated test when the internal rack temperatures have been as low as 36 degrees F.

As indicated, the MCA operates in a very benign environment. The equipment rack Avionics Air Assembly controls the MCA environment to temperature levels below the cabin air environment. If the AAA shutsdown, the MCA is automatically shutdown to prevent equipment damage due to operations beyond required operating environmental limits. The only time the MCA would be exposed to lower than expected temperatures is during depress and the MCA will be shutdown under these circumstances.

A review of 386 FMEAs indicated no critical functions or parts associated with the MCA. In addition, there are CO2 and O2 hand–held devices and Russian segment atmospheric monitoring equipment that provides redundant capability to the on–orbit functionality provided by the MCA. In the case of MCA failure, the O2 and Total pressure control would have to be manually controlled under contingency operation. This manual mode would continue until a replacement could be made, which was already accepted for South Atlantic Anomaly conditions.

PG3–185:

ITEM:

Major Constituent Analyzer Part Number 359800, Serial Numbers 0001 and 0002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. ”When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same...” This requires that the protoflight hardware Random Vibration test levels and spectrum envelope maximum predicted flight level and spectrum minus 6 dB but not less than the workmanship screening level of 6.1 grms as defined in SSP 41172.

EXCEPTION:

The MCA Protoflight 1 and 2 overall random vibration test level for manufacturing screening shall be 4.3 grms.

RATIONALE:

During random vibration testing both MCA protoflight units were workmanship–screened twice at an overall level of 4.3 grms. Both MCA protoflight units have accumulated over 500 hours of power–on testing conducted at the hardware provider's facility and during higher–level integration testing. The MCA is a protoflight design; thus, there is no qualification unit that has demonstrated by test the vibration fatigue life of the design. Additional vibration testing will subject the protoflight MCA to additional fatigue life expenditure without having test–demonstrated margin. Follow–on builds of internal Data and Control Assembly ORUs (Part Number 359650) with redesigned Electronic Data Processor boards will be tested to the 6.1 grms ORU equivalent vibration screen level with power on. In addition, there are CO2 and O2 hand–held devices and Russian segment atmospheric monitoring equipment that provides redundant capability to the on–orbit functionality provided by the MCA.

A review of 386 FMEAs indicated no critical functions or parts associated with the MCA. In addition, there are CO2 and O2 hand–held devices and Russian segment atmospheric monitoring equipment that provides redundant capability to the on–orbit functionality provided by the MCA. In the case of MCA failure, the O2 and Total pressure control would have to be manually controlled under contingency operation. This manual mode would continue until a replacement could be made, which was already accepted for South Atlantic Anomaly conditions.

PG3–186:

ITEM:

Major Constituent Analyzer Part Number 359800, Serial Numbers 0001 and 0002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…" This requires that electrical and electronic components be powered on and monitored for failures or intermittences during the protoflight random vibration test in accordance with paragraph 4.2.5.4.

EXCEPTION:

The MCA Protoflight 1 and 2 will not be powered on and monitored for failures or intermittences during the protoflight random vibration test.

RATIONALE:

During random vibration testing both MCA protoflight units were workmanship–screened twice at an overall level of 4.3 grms. Both MCA protoflight units have accumulated over 500 hours of power–on testing conducted at the hardware provider's facility and during higher–level integration testing. Board deflection analysis has been performed on the MCA and stiffeners added to prevent flexure during vibration testing. The MCA design does not incorporate the use of edge connectors on circuit card assemblies or the use of high profile components. All vertical mounted components used in the MCA design are staked and bonded to circuit boards to preclude vibration induced component movement or stress at the component solder joints. The MCA is a protoflight design; thus, there is no qualification unit that has demonstrated by test the vibration fatigue life of the design. Additional vibration testing will subject the protoflight MCA to additional fatigue life expenditure without having test–demonstrated margin. Follow–on builds of internal Data and Control Assembly ORUs (Part Number 359650) with redesigned Electronic Data Processor boards will be tested to the 6.1 grms ORU equivalent vibration screen level with power on.

A review of 386 FMEAs indicated no critical functions or parts associated with the MCA. In addition, there are CO2 and O2 hand–held devices and Russian segment atmospheric monitoring equipment provide redundant capability to the on–orbit functionality provided by the MCA. Finally, all spare ORUs are vibrated with power on and monitoring in accordance with Boeing Contract Letter 2–4450–NSR–5232–98 and Orbital SCP 246.

PG3–187:

ITEMS:

Major Constituent Analyzer Assembly Part Number 359675
Major Constituent Analyzer internal ORU # 2 (Mass Spectrometer)

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…" This requires that the duration of the protoflight leakage test be no less than 60 minutes in accordance with 4.2.11.3F.

EXCEPTION:

The Major Constituent Analyzer Assembly internal to MCA Protoflight 1 and 2 shall be leak tested via Method VI for a minimum of two minutes.

RATIONALE:

During the Method VI leak testing of the internal Major Constituent Analyzer Assembly conducted at the Orbital facility, no appreciable leak response was detected in 2 minutes of testing. This indicates there was no gross leak (no leakage rate above the Mass Spectrometer Leak Detector helium background of 2E–10 sccs Helium was detected) in the MCA design. The MCA Mass Spectrometer vacuum operating pressure is about 2E–06 torr. Following environmental testing, functional testing were conducted successfully. Therefore, there are no leaks with the leakage rate above the allowable level (i.e. less than 1E–08 sccs Air) that can compromise the MCA functionality. Furthermore, there are CO2 and O2 hand–held devices and Russian segment atmospheric monitoring equipment that provides redundant capability to the on–orbit functionality provided by the MCA.

PG3–188:

ITEM:

Major Constituent Analyzer Part Number 359800, Serial Numbers 0001 and 0002.

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same..." This requires at least 100 degrees F (55.6 degrees C) protoflight thermal sweep between the minimum and maximum test temperatures, and the minimum test temperature be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The temperature sweep performed during protoflight thermal cycle testing will be 37 degrees F (53 degrees F to 90 degrees F.)

RATIONALE:

Follow–on builds of internal Data and Control Assembly ORUs (Part Number 359650) with redesigned Electronic Data Processor boards will be tested to the 6.1 grms ORU equivalent vibration screen level with power on and thermal cycle testing conducted using a thermal sweep of 100 degrees F.

There are CO2 and O2 hand–held devices and Russian segment atmospheric monitoring equipment that provides redundant capability to the on–orbit functionality provided by the MCA. The Crew Health Care System Combustion Products Analyzer also provides an approximate O2 concentration measurement as a backup to the capability provided by the MCA. After Flight 20A, additional redundant capability will be provided via the Node 3 MCA. The MCA's highest criticality is during Airlock campout and this criticality is mitigated through the NASA exercise prebreathing protocol. In addition, CO2 levels prior to sleep periods would take tens of hours to reach unsafe levels.

PG3–189:

ITEM:

Major Constituent Analyzer Part Number 359800, Serial Numbers 0001 and 0002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…" This requires that component leak checks be made prior to initiation of, and following the completion of, component thermal and vibration tests in accordance with 4.2.11.2.

EXCEPTION:

The MCA Protoflight 1 and 2 will not undergo leakage testing after each environmental test.

RATIONALE:

Functional testing of the internal Major Constituent Analyzer Assembly can detect leaks greater than the calibrated leak rate via the ion pump current. Subsequent functional testing of the MCA has detected no out–of–specificationleak rates except for damaged hardware (i.e. internal Major Constituent Analyzer Assembly Part Number 359675, Serial Number Q0001). This indicates there are no leaks with the leakage rate above the allowable level (i.e. less than 1E–8 sccs Air). The internal Major Constituent Analyzer Assembly is a limited–life item and will be replaced periodically.

PG3–190:

ITEM:

Window Shutter Gearbox Assembly Part Number 683–13303–3

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.

Paragraph 5.1.2.3B, Test Levels and Duration. The component shall be at a maximum predicted temperature during the hot portion of the cycle and at the minimum predicted temperature during the cold portion of the cycle.

EXCEPTION:

The Acceptance Thermal Vacuum Test of the Window Shutter Gearbox Assembly shall be performed at 170 degrees F.

RATIONALE:

Updated predicted temperatures of the Window Shutter Gearbox Assembly in the US Laboratory indicated a maximum predicted temperature of 183 degrees F. However, the flight Window Shutter Gearbox Assembly experienced a maximum temperature of 170 degrees F during acceptance thermal vacuum testing. Yet, performed thermal testing did stress the Window Gearbox Shutter Assembly and torque and leak rate measurements appear insensitive to temperature.

Torque:

During the Qualification and Acceptance Thermal Vacuum Testing, torque values were insensitive to temperature extremes. Torque values increase only slightly at the cold extremes and the torque values at the tested hot extreme of 170 degrees F are at or below those measured at ambient conditions. Furthermore, the measurements at Ambient, Hot, and Cold extremes are well below requirements. The Pass/Fail specification requirement is 690 in–oz ; for the USL application, the maximum measured torque value was 48 in–oz.

Leak Rate:

During Qualification Thermal Vacuum Testing, Leak Rate was unchanged after exposure to temperature extremes. Leak Rate testing on the USL has been successfully performed (i.e. the Window Shutter Gearbox Assembly has passed its requirements). Furthermore, the USL–tested configuration was the worst case configuration as only the Inner Shaft was present (i.e. fewer seals were present).

PG3–191:

ITEM:

Pump Package Assembly Part Number 2353170–1–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.
Paragraph 5.1.3.4, Supplementary Requirements. Functional tests shall be conducted at the maximum and minimum predicted temperature levels during the first and last operating cycles after the dwell and after return of the component to ambient temperature.

EXCEPTION:

The Pump Package Assembly shall not have a full functional test performed at the hot and cold extremes during acceptance thermal cycle testing.

RATIONALE:

Only limited testing was performed during the high and low temperature extremes of acceptance thermal cycle testing. This consisted of a standby power consumption test, with sensors operating, and the power assumption test with operation of the pump. The intent of functional testing at temperature is both to verify functionality at temperature extremes and to act as a workmanship screen for the electronic components within the system. At lower levels of assemblies (Pump/Fan Motor Controller and sensors), the mechanical and electrical components within the Pump Package Assembly have undergone full functional testing at temperature extremes, as well as other workmanship screens for the electronics system. In the event that screening was not done, measures were taken to preclude the effects that these screens would precipitate.

During Reliability Acceptance Testing for the Pump/Fan Motor Controller contained within the Pump Package Assembly, the Pump/Fan Motor Controller undergoes thermal cycling in the range of 0 +/– 3 degrees F to 107 +/– 3 degrees F, with functional verification of the electronics at the temperature extremes. Also, during reliability acceptance thermal cycle testing, the Pump/Fan Motor Controller has a workmanship screen of 1 minute of random vibration in the worst axis for CCA deflections at each of the temperature extremes. These tests verify both functionality at temperature and workmanship screening.

All of the Pump Package Assembly sensing elements for temperature, pressure, or quantity, has electronics that are potted, which minimizes any damaging effects that a temperature workmanship screen would precipitate. The remainder of the assemblies contained within the Pump Package Assembly is mechanical in nature, either pressure relief valves, flow check valves, the accumulator, or other assemblies with no moving parts. These parts are all in the flow path of the circulating coolant, and as such will not be subject to the extreme temperatures imposed by ambient air.

PG3–192:

ITEM:

Pump Package Assembly Part Number 2353170–1–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperature of the Pump Package Assembly shall be 33 degrees F during qualification thermal cycle testing.

RATIONALE:

The minimum predicted operating temperature of the Pump Package Assembly is 55 degrees F. The Pump Package Assembly qualification unit was tested while operating to a minimum temperature of 33 degrees F.

The Material Identification and Usage List shows that the mechanical portions of the assembly are insensitive to a non–operational low temperature of 0 degrees F.

In the event that there is any operational malfunction due to some factor that would have been screened out by non–operational cold testing of the unit, then it should be noted that the components within the Thermal Control System loops have manual operational capability.

The Pump Package Assembly hardware is Criticality 1R. The worst case is loss of coolant to Lab DDCUs that will shut down due to over–temperature resulting is loss of power distribution in the Lab, and loss of all life and station critical functions. An operational workaround has been developed and documented to recover partial functionality. A workaround requires the crew jumpering loads to the Low Temperature Loop. The workaround has been demonstrated and accepted by NASA as documented in JSC 48532–5A, ISS Malfunction Check List. In addition, a pre–positioned spare is available on–orbit.

The purpose of the 20–degree F qualification margin is to demonstrate that there is sufficient design margin such that the acceptance test to which the flight hardware is subjected does not excessively degrade the hardware's useful life. This cold temperature used for qualification and acceptance thermal cycle testing is sufficiently benign for space–quality hardware such that there is very little risk that acceptance testing at the same cold temperature as qualification (33 degrees F) excessively degrades the life of the hardware.

PG3–193:

ITEM:

Pump Package Assembly Part Number 2353170–1–1, Serial Numbers 003 and 004

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.4, Supplementary Requirements. Qualification test input to the component shall in all cases envelope acceptance test levels plus test tolerances.

EXCEPTION:

Acceptance random vibration test levels exceeding qualification random vibration test levels shall be permitted as follows:

For Serial Number 003:

x–axis: at 310 Hz, 50 Hz bandwidth, 0.01 g^2/Hz magnitude
at 1500 and 1800 Hz; 20 Hz bandwidths, 0.003 g^2/Hz magnitude
y–axis: at 320 Hz, 20 Hz bandwidth, 0.001 g^2/Hz magnitude
z–axis: at 250 Hz, 40 Hz bandwidth, 0.003 g^2/Hz magnitude

For Serial Number 004:

y–axis: at 250 Hz, 20 Hz bandwidth, 0.001 g^2/Hz magnitude
z–axis: at 310 Hz, 30 Hz bandwidth, 0.003 g^2/Hz magnitude

RATIONALE:

The test tolerances were doubled, allowing for the possible overlap of as–run conditions of acceptance and qualification tests below 80 Hertz and above 275 Hertz. As indicated, the out of tolerance conditions resulted in qualification random vibration levels below acceptance random vibration levels. The approximate energy differential from the overlap is equivalent to 0.15 grms (maximum). The overall energy levels achieved were 8.18 grms during qualification, and 6.08 grms during acceptance. The hardware met all performance requirements and passed vibration testing without failure. Thus, due to the narrow frequency bandwidth and the low exceedance between acceptance random vibration and qualification random vibration, the risk associated with accepting the qualification test as–run is minimal.

PG3–194:

ITEM:

Overboard Water Vent Part Number 683–20217–5

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.
Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance temperature during the cold portion of the cycle.

EXCEPTION:

The maximum temperature of the Overboard Water Vent shall be 150 degrees F during acceptance thermal cycle testing.

RATIONALE:

The maximum predicted operating temperature of the Overboard Water Vent (OWV) is 167 degrees F. The component of the OWV assembly that has limiting thermal capacity is the flexhose (1F98653–509). The particular area of the flexhose in question is the sealing joint of the inner liner to the stainless steel collar. The inner liner of the flexhose is polytetrafluoroethylene, more commonly called Teflon. The Teflon has a maximum operating temperature of 275 degrees F. Beyond that temperature, the Teflon becomes plastic and structural sealing characteristics diminish. The flexhose chosen for the design of the OWV has been previously qualified for ISS internal use. The maximum operating temperature for the hose during internal use is 150 degrees F. The flexhose design was qualified to 170 degrees F and each flexhose procured from the supplier is acceptance tested to 150 degrees F. Thus, the flexhose in the OWV has been procured with an acceptance test of 150 degrees F. During a meeting prior to the OWV acceptance tests, a decision was made to limit the thermal cycling of the OWV assembly to the existing acceptance test limit of the flexhose, namely 150 degrees F. With the OWV predicted operating temperature of 167 degrees F and acceptance testing at 150 degrees F, there is a 17 degrees F difference in the compliance of SSP 41172. However, the flexhose has successfully completed a Delta Qualification Test composed of a Leakage Test and a Thermal Cycling Test. The thermal cycling test used a maximum temperature of 187 degrees F, 20 degrees F beyond the maximum predicted on–orbit temperature.

Heaters in the OWV cause the joint area of the flexhose to reach the 167 degrees F. Driving the heaters to this temperature limit provides the OWV with the capability to perform its intended function of venting wastewater. This temperature limit was derived from an extensive thermal analysis, CS–28V6C–WAB–005/00. Infrared scan data obtained at KSC while the OWVs were installed on the US Laboratory and operating to the 167 degrees F limit has been correlated to a thermal model to verify the thermal analysis. The flexhose joint is required to prevent leakage of the water in the flexhose during venting with a maximum pressure of 23 psid.

The purpose of the thermal cycle acceptance test is to screen the component for workmanship defects by subjecting it to the worst–case predicted on–orbit temperatures. The risk of a flight OWV failing on–orbit if it is subjected to the worst–case maximum predicted temperature (167 degrees F) due to a workmanship defect which was not detected at 150 degrees F during thermal cycle acceptance testing is considered minimal.

PG3–195:

ITEM:

Spacelab Logistics Pallet High Pressure Gas Tank ORU Adapter Assembly Part Number 683–55250–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test level temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

A 20 degree F margin was not obtained between the Qualification test temperatures and the Acceptance test temperatures for the gear assemblies (maximum and minimum temperatures) and for the handle assemblies (minimum temperature). The predicted operating temperatures for Flight 7A and the qualification and acceptance test temperatures for the adapter assemblies are summarized below:

Adapter Component	Test	Minimum Operating Temperature (degrees F)	Maximum Test Temperature (degrees F)
Gear Assemblies	Predicted Operating Temperature (Flight 7A)	–78.2	244.0
	Qualification	–115	270
	Acceptance	–110*	265*
Handle Assemblies	Predicted Operating Temperature (Flight 7A)	–83.9	210.0
	Qualification	–110	245
	Acceptance	–95*	225
Rods	Predicted Operating Temperature (Flight 7A)	–93.3	203.9
	Qualification	–115	230
	Acceptance	–95	210

* Exceeded Acceptance Test Temperature

RATIONALE:

For Spacelab Logistics Pallet High Pressure Gas Tank ORU Adapter Assembly components, the minimum qualification test temperature achieved is at least 20 degrees F less than the minimum predicted operating temperature for the components on Flight 7A. Similarly, the maximum qualification test temperature achieved is at least 20 degrees F greater than the maximum predicted operating temperature for the components on flight 7A.

The Spacelab Logistics Pallet Adapter Assembly successfully operated during the Acceptance Tests at the extreme maximum and minimum temperatures indicated. In conjunction with the successful qualification testing, the Adapter mechanisms are shown to be of robust design and not to be sensitive to temperature extremes. Therefore, the performed Qualification and Acceptance Thermal Vacuum testing is deemed adequate, and the design of the Adapter mechanisms for withstanding extreme temperature conditions is accepted.

PG3–196:

ITEM:

Pressure Cover Part Number 683–11403–4 Serial Number 000001

SSP 41172 REQUIREMENT:

Paragraph 5.1.1, Functional Test, Component Acceptance.
Paragraph 5.1.1.3, Supplementary Requirements. ALL REQUIREMENTS.

Paragraph 5.1.6, Pressure Test, Component Acceptance.
Paragraph 5.1.6.3, Test Levels. ALL REQUIREMENTS.
Paragraph 5.1.6.4, Supplementary Requirements. ALL REQUIREMENTS.

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The Internal Pressure Cover of the US Lab did not undergo Acceptance Functional, Pressure, and Leakage Testing.

RATIONALE:

The stress analysis (D683–29046–1–12) shows low stress levels and large margins of safety for this configuration. Additionally, this hardware is classified as fracture critical. The raw stock was ultrasonically inspected, the machined surfaces were dye penetrant inspected, and the holes were eddy current inspected. The risk of structural failure in this configuration is low.

This cover is mounted on the inside wall of the module over the window only in the event of damage to the window assembly. This produces compressive forces on the seals in the event this cover becomes the primary seal to vacuum. The seals are a silicone rubber O–ring in a groove. The risk of a major seal leak is minimal.

PG3–197:

ITEM:

Extra–Vehicular Mobility Unit Audio Control Panel, Part Number 312001–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. The test duration in each of the three orthogonal axes shall be three times the expected flight exposure time to the maximum predicted level and spectrum or three times the component random vibration acceptance test time if that is greater, but not less than three minutes per axis.

EXCEPTION:

For the Extravehicular Mobility Unit Audio Control Panel, the duration of the qualification random vibration test shall be equal to the duration of the acceptance random vibration test (three minutes in each of three orthogonal axes).

RATIONALE:

The Qualification unit was tested at three db above Acceptance test random vibration levels with power on. No anomalies were detected during testing. The Extravehicular Mobility Unit Audio Control Panel is Criticality 3 hardware. If an Extravehicular Mobility Unit Audio Control Panel failure occurs, the RF audio is available. The flight Extravehicular Mobility Unit Audio Control Panel has a calculated demonstrated fatigue life expended of 0.9530, where the calculation includes: Acceptance test, launches, landings, ferry flight, on–orbit operations, and transportation. This calculation shows a calculated positive margin of approximately 5 percent for the life of the Extravehicular Mobility Unit Audio Control Panel. Comparison of loads for the Extravehicular Mobility Unit Audio Control Panel shows the following: Design load = 142 Gs = 1,108 lbf, Qualification testing load = 110.3 Gs = 860 lbf, Acceptance testing load = 83.1 Gs = 648 lbf.

PG3–198:

ITEM:

Coupling, 0.125–inch, feedthrough–mounted, mated–pair Part Numbers 683–19485–3 and 683–19485–6

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3, Test Levels and Duration, Item B, Method II – The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The duration of the qualification leakage test of the QD coupling pressure cap will be at least five minutes.

RATIONALE:

The test unit was charged with Helium from a supply line to 14.7 psig. The male QD coupling and mated pressure cap was monitored for leakage with Helium leak detector (mass spectrometer) for a five–minute test period. Parker Symetrics procedure SYM 95–210 makes provision to extend the leak rate stabilization that is “If between the fourth and fifth minute, the leak rate is not stabilized within ± 5E–07 sccs or on a downward trend, continue stabilization for a maximum of 30 minutes. Leakage shall not exceed 1E–06 sccs.” Stabilization was attained between the fourth and fifth minute across this small 1/8–inch O–ring pair. Actual test data showed that the maximum leakage rate recorded was 6.2E–07 sccs. This is well within the acceptable limit.

PG3–199:

ITEM:

Coupling, 0.125–inch, feedthrough–mounted, mated–pair Part Numbers 683–19485–3 and 683–19485–6

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3B, Test Levels and Duration, Method II – The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

The duration of the acceptance leakage test of the QD coupling pressure cap will be at least five minutes.

RATIONALE:

The test unit was charged with Helium from a supply line to 14.7 psig. The male QD coupling and mated pressure cap was monitored for leakage with Helium leak detector (mass spectrometer) for a five–minute test period. Parker Symetrics procedure SYM 95–210 makes provision to extend the leak rate stabilization that is “If between the fourth and fifth minute, the leak rate is not stabilized within ± 5E–07 sccs or on a downward trend, continue stabilization for a maximum of 30 minutes. Leakage shall not exceed 1E–06 sccs.” Stabilization was attained between the fourth and fifth minute across this small 1/8–inch O–ring pair. Actual test data showed that the maximum leakage rate recorded was 6.2E–07 sccs. This is well within the acceptable limit.

PG3–200:

ITEMS:

Coupling, 0.125–inch, feedthrough–mounted, mated–pair Part Numbers 683–19485–3 and 683–19485–4
Coupling and Jumper Assembly, 0.125–inch, Part Number 683–19363–1
Coupling Half, Bulkhead, 0.125–inch, Part Number 683–19364–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.2, Test Description and Alternatives, Item A, Method I(gross leak test) – The component shall be completely immersed in a liquid so that the uppermost part of the test item is 2 +1/–0 inches (5 +2.5/–0 cm) below the surface of the liquid. The critical side or side of interest of the component shall be in a horizontal plane facing up. The liquid, pressurizing gas, and the test item shall be 73 + 18 degrees F (23 + 10 degrees C). The gas used for pressurizing shall be clean and dry with a dew point of at least –26 degrees F (–32 degrees C). Any observed leakage during immersion as evidenced by bubbles emanating from the component indicates a failure of seals.

EXCEPTION:

All items already delivered are qualified based on the successful characterization test in accordance with SSP 41172, paragraph 4.2.11.2, item A, Method I., of 27 July 2000.

RATIONALE:

A successful characterization test of the ARS QD couplings was perfomed in accordance with SSP 41172, paragraph 4.2.11.2, item A, Method I. Update the Parker Symetrics Qualification Test Procedure to perform a liquid immersion test of the mated QD coupling including the hose and hydraflow fitting for the required duration of 60 minutes. QualificationAnalysis to be provided to show that the coupling can limit the external leakage to 1E–04 sccs of Nitrogen at 14 psid at a temperature between 40 – 125 degrees F.

PG3–201:

ITEMS:

Coupling, 0.125–inch, feedthrough–mounted, mated–pair
Part Numbers 683–19485–3 Serial Numbers 1001 thru 1025 and 683–19485–4 Serial Numbers 1001 thru 1012
Coupling and Jumper Assembly, 0.125–inch, Part Number 683–19363–1 Serial Numbers 1001 thru 1005
Coupling Half, Bulkhead, 0.125–inch, Part Number 683–19364–1 Serial Numbers 1001 thru 1003

SSP 41172 REQUIREMENT:

Paragraph 5.1.7.2, Leakage Test, Component Acceptance.
Paragraph 5.1.7.2, Test Description and Alternatives, Item A, Method I(gross leak test) – The component shall be completely immersed in a liquid so that the uppermost part of the test item is 2 +1/–0 inches (5 +2.5/–0 cm) below the surface of the liquid. The critical side or side of interest of the component shall be in a horizontal plane facing up. The liquid, pressurizing gas, and the test item shall be 73 + 18 degrees F (23 + 10 degrees C). The gas used for pressurizing shall be clean and dry with a dew point of at least –26 degrees F (–32 degrees C). Any observed leakage during immersion as evidenced by bubbles emanating from the component indicates a failure of seals.

EXCEPTION:

All items already delivered are acceptable based on the successful characterization test in accordance with SSP 41172, paragraph 4.2.11.2, item A, Method I., of 27 July 2000.

RATIONALE:

A successful characterization test of the ARS QD couplings was perfomed in accordance with SSP 41172, paragraph 4.2.11.2, item A, Method I. Update the Parker Symetrics Acceptance Test Procedure to perform a liquid immersion test of the mated QD coupling including the hose and hydraflow fitting for the required duration of 60 minutes.

PG3–202:

ITEM:

Coupling, 0.125–inch, feedthrough–mounted, mated–pair Part Numbers 683–19485–3 and 683–19485–6

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3, Test Levels and Duration, Item F, Method VI – The duration of the test shall be no less than 60 minutes.

EXCEPTION:

The duration of the qualification leakage test of the QD coupling will be at least five minutes.

RATIONALE:

Parker Symetric procedure SYM95–210 makes the same provisions about leak rate stabilization as noted for PG3–198. The same 5–minute stabilization characteristic for the two 1/8–inch O–ring was already demonstrated. In addition, the ARS QD coupling successfully passed leak testing in flight configuration at the element/system level after installation in the lab. This demonstrated that the 5–minute duration of leak testing at the component level is acceptable.

PG3–203:

ITEM:

Sync and Control Unit (SCU) Part Number 136AE7010–302

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same…" This requires Protoflight Random Vibration testing in accordance with 5.1.4.

EXCEPTION:

The Protoflight Random Vibration test on the SCU shall be performed at a level of 4.3 grms.

RATIONALE:

The tested environment envelops the maximum predicted flight of 4.3 grms. The SCU, as a system, was designed to withstand a 4.3 grms test. Circuit cards, such as the 9 VFOT/VFOR cards, have been vibrated at the 6.1 grms workmanship screening level. All fasteners have locking features or are staked, and are inspected during assembly. All circuit assemblies are conformal coated which precludes malfunctions due to shorting from conductive hardware loose in the unit. Multiple units are installed (one each in USL AV1 and AV2 racks) to provide redundancy for all functions except Time Base Correction and Split Screen Processing simultaneous operation capabilities. SCU Serial Number 95002 has experienced two separate vibration tests and SCU Serial Number 95003 has gone through three separate vibration tests at the 4.3 grms level powered on. Burn–in time for these units is over 400 hrs without any problems. Finally, the SCUs are classified Criticality 2R with two flight spares available for replacement in 2001. Thus, the risk associated with this exception is limited.

PG3–204:

ITEM:

Sync and Control Unit (SCU) Part Number 136AE7010–302

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same…" This requires Protoflight Thermal Cycling testing with a minimum temperature sweep of 100 degrees F between the minimum and maximum test temperatures in accordance with 5.1.3.

EXCEPTION:

The SCU will undergo a minimum operational temperature sweep of 77 degrees F during protoflight thermal cycle testing.

RATIONALE:

The SCU has an operating temperature limit of 33 to 90 degrees F. The performed protoflight thermal cycle test on the SCU operated the units over the temperature range of 23 to 100 degrees F. The transition rate of SCU was approximately 2.5 degrees F per minute. As indicated, the test cycles encompassed the maximum expected operational environments with 10 degrees F margin. Twenty–four operating thermal cycles were conducted on the original flight units (Serial Numbers 95002 and 95003). Eight operating thermal cycles is the minimum number of operating thermal cycles required for protoflight testing in accordance with SSCN 003034. Multiple units are installed (one each in AV1 and AV2 racks in the USL) to provide redundancy for all functions except Time Base Correction and Split Screen Processing simultaneous operation capabilities. The SCUs are classified Criticality 2R with two flight spares available for replacement in 2001. Thus, the risk associated with this exception is limited.

PG3–205:

ITEM:

O2/N2 Pressure Sensor Part Number 683–16443–1 (Carleton B41397–1)

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The O2/N2 Pressure Sensor will not be powered on and monitored during the qualification random vibration testing.

RATIONALE:

The O2/N2 Pressure Sensor parameter available for monitoring during random vibration test is the output voltage based upon the pressure in the Airlock subsystem. The O2/N2 Pressure Sensor contains electronics consisting only of strain gages and compensating resistors, contains only a minimal number of electronic components, and does not utilize a printed circuit board with a card edge connector. The O2/N2 Pressure Sensor electrical parts are potted which dampens vibration excitation.

The O2/N2 Pressure Sensor is criticality 3 hardware and sensor failure results in no loss of functionality. If failure occurs, the system performance would be degraded with an inability to confirm delivery conditions to the regulator or relief assembly. The remaining pressure sensors could be used to determine and indicate pressure within the Airlock subsystem. Long term corrective action to a failure is to remove and replace the failed sensor.

Therefore, powering the O2/N2 Pressure Sensor and monitoring for intermittences during qualification random vibration tests may not provide significant value.

PG3–206:

ITEM:

O2/N2 Pressure Sensor Part Number 683–16443–1 (Carleton B41397–1)

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The O2/N2 Pressure Sensor will not be powered on and monitored during the acceptance random vibration testing.

RATIONALE:

The O2/N2 Pressure Sensor parameter available for monitoring during random vibration test is the output voltage based upon the pressure in the Airlock subsystem. The O2/N2 Pressure Sensor contains electronics consisting only of strain gages and compensating resistors, contains only a minimal number of electronic components, and does not utilize a printed circuit board with a card edge connector. The O2/N2 Pressure Sensor electrical parts are potted which dampens vibration excitation.

The O2/N2 Pressure Sensor is criticality 3 hardware and sensor failure results in no loss of functionality. If failure occurs, the system performance would be degraded with an inability to confirm delivery conditions to the regulator or relief assembly. The remaining pressure sensors could be used to determine and indicate pressure within the Airlock subsystem. Long term corrective action to a failure is to remove and replace the failed sensor.

Therefore, powering the O2/N2 Pressure Sensor and monitoring for intermittences during acceptance random vibration tests may not provide significant value.

PG3–207:

ITEM:

O2/N2 Pressure Sensor Part Number 683–16443–1 (Carleton B41397–1)

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle. The minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The temperatures experienced during the Qualification Thermal Cycle Test of the O2/N2 Pressure Sensor will be from a minimum temperature of – 40 degrees F to a maximum temperature of 150 degrees F.

RATIONALE:

The on–orbit operational environment of the O2/N2 Pressure Sensor will be from 35 degrees F to 125 degrees F. Three cycles of the qualification thermal cycle test for the O2/N2 Pressure Sensor were performed from a minimum temperature of – 40 degrees F to a maximum temperature of 150 degrees F; remaining cycles were performed from a minimum temperature of 15 degrees F to a maximum temperature of 145 degrees F. Two cycles of the acceptance thermal cycle test were previously performed from a minimum temperature of – 40 degrees F to a maximum temperature of 150 degrees F; remaining cycles were performed from a minimum temperature of 25 degrees F to a maximum temperature of 135 degrees F. Thus, due to the as–run conditions, 0 degrees F thermal margin was demonstrated during qualification thermal cycle testing as indicated.

Application and Derating Analysis shows that there is at least 76 degrees F (+24.6 degree C) thermal safety margin for all EEE components used in the O2/N2 Pressure Sensor in relation to the 150 degrees F (65.6 degrees C) maximum temperature experienced. Additionally, no concerns result from testing the O2/N2 Pressure Sensor to a minimum temperature of – 40 degrees F.

The O2/N2 Pressure Sensor is criticality 3 hardware and sensor failure results in no loss of functionality. If failure occurs, the system performance would be degraded with an inability to confirm delivery conditions to the regulator or relief assembly. The remaining pressure sensors could be used to determine and indicate pressure within the Airlock subsystem. Long term corrective action to a failure is to remove and replace the failed sensor.

Therefore, the risk associate with the performance of the qualification thermal cycling test as documented is minimal and thermal requalification is not needed. As additional O2/N2 Pressure Sensors are contracted on the ISS program, acceptance thermal cycle test will be performed only over the expected on–orbit environments.

PG3–208:

ITEM:

O_2N_2 Latching Motor Valve Part Numbers 683–16419–1 and 683–16419–2 (Carlton Part Numbers B41395–1 and B41395–3)

SSP 41172 REQUIREMENT:

Paragraph 4.2.5 Random Vibration Test, Component Qualification.
Paragraph 4.2.5.4 Supplementary Requirements. Electrical and electronic components shall be energized and monitored during the test. Parameters shall be monitored for failures or intermittence during the test.

EXCEPTION:

Qualification random vibration testing of the Latching Motor Valve without power on and monitoring is permitted.

RATIONALE:

Because of the low number of electrical parts, the Latching Motor Valve has an extremely low probability of intermittent defects. As the Latching Motor Valve is not required to demonstrate mechanical functionality during the random vibration test and because they do not have any electrical test connectors, only a limited number of the total possible intermittent defects could be detected during the test. Therefore, repeating random vibration tests with the Latching Motor Valve powered on and monitored is not technically warranted.

All Electronics are potted.

N_2 and O_2 spares are available on–orbit.

Functional testing was successfully performed at the completion of the random vibration testing. After testing, the unit was depressurized and disconnected from the test setup. There was no evidence of damage or deformation. The Latching Motor Valve will be functionally checked out on–orbit soon after module ingress. The Latching Motor Valve will not be powered on during the launch/ascent vibration environment.

PG3–209:

ITEM:

O_2N_2 Latching Motor Valve Part Numbers 683–16419–1 and 683–16419–2 (Carlton Part Numbers B41395–1 and B41395–3)

SSP 41172 REQUIREMENT:

Paragraph 5.1.4 Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.4 Supplementary Requirements. Electrical and electronic components shall be energized and monitored during the test. Parameters shall be monitored for failures or intermittence during the test.

EXCEPTION:

Acceptance random vibration testing of the Latching Motor Valve without power on and monitoring is permitted.

RATIONALE:

Because of the low number of electrical parts, the Latching Motor Valve has an extremely low probability of intermittent defects. As the Latching Motor Valve is not required to demonstrate mechanical functionality during the random vibration test and because they do not have any electrical test connectors, only a limited number of the total possible intermittent defects could be detected during the test. Therefore, repeating random vibration tests with the Latching Motor Valve powered on and monitored is not technically warranted.

All Electronics are potted.

N_2 and O_2 spares are available on–orbit.

Functional testing was successfully performed at the completion of the random vibration testing. After testing, the unit was depressurized and disconnected from the test setup. There was no evidence of damage or deformation. The Latching Motor Valve will be functionally checked out on–orbit soon after module ingress. The Latching Motor Valve will not be powered on during the launch/ascent vibration environment.

PG3–210:

ITEM:

O_2N_2 Latching Motor Valve Part Numbers 683–16419–1 and 683–16419–2 (Carlton Part Numbers B41395–1 and B41395–3).

SSP 41172 REQUIREMENT:

Paragraph 4.2.3 Thermal Cycling Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1. degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The maximum qualification temperature for the Latching Motor Valve shall be 155 +/– 5 degrees F.

RATIONALE:

The maximum predicted operating temperature of the Latching Motor Valve is 130 +/– 5 degrees F. Flight Latching Motor Valves were tested in accordance with Carleton acceptance test procedure CRA–1385, Revision C to a maximum temperature of 155 +/– 5 degrees F. However, the qualification Latching Motor Valve was tested to the identical maximum temperature of 155 +/– 5 degrees F.

The testing associated with the above Acceptance Test Procedure temperature does not create a risk to the hardware, since all materials employed in the design of the Latching Motor Valve are rated for significantly higher temperatures. The low number of electrical parts used in the design results in a low induced stress on the component. The EEE parts application analysis confirms that all parts have derated temperature limits of greater than 200 degrees F. In addition, mechanical components employed in the design of the Latching Motor Valve (i.e. Teflon and Silicon O–ring seals) are rated for a wide range of temperatures (Teflon seal: –100 to 400 degrees F; Silicon O–rings: – 45 to 200 degrees F). Thus, additional qualification testing at higher temperatures to qualify the design for the as–performed acceptance thermal testing is not warranted. However, the Acceptance Test Procedure will be updated to reflect appropriate temperature range for future testing.

PG3–211:

ITEMS:

Oxygen Latching Motor Valve Part Number 683–16419–2 Serial Numbers 001001 and 001003
Oxygen Low Pressure Regulator/Relief Valve Part Number 683–16421–6 Serial Number 001001
Oxygen Medium Pressure Regulator/Relief Valve Part Number 683–16421–5 Serial Number 001002
Oxygen Manual Isolation Valve Part Number 683–16439–3 Serial Numbers 001001 001002, 001003, 001004, 001005, 001006, and 001007
Nitrogen Manual Isolation Valve Part Number 683–16439–1 Serial Numbers 001003 and 001004

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Leak Test shall not be required for the indicated Nitrogen and Oxygen fluid hardware units for initial delivery. Any rework of these units that would normally dictate performance of a leak test shall require leak testing of these units in full compliance with SSP 41172.

RATIONALE:

The indicated components except for the Oxygen Low Pressure Regulator/Relief Valve and the Oxygen Medium Pressure Regulator/Relief Valve underwent a stand–alone vacuum chamber leak test via a bell jar. During this vacuum chamber testing, no gross leaks were found; however, the 5–minute duration of the test was not sufficient to allow helium permeation through the component such that an accurate fine leakage rate could be obtained to determine compliance with the individual component specifications. The Oxygen Low Pressure Regulator/Relief Valve and the Oxygen Medium Pressure Regulator/Relief Valve underwent acceptance leak testing via use of a Bubble–O–Meter; however, evaluation subsequently determined that the performed methodology could not determine a fine leakage rate to the same accuracy as a vacuum chamber leak test performed in compliance with SSP 41172.

An accumulation leak test was performed in place for components installed in the Airlock at the system level and for Manual Isolation Valves external to the Airlock at the Tank ORU level. The tests were performed in compliance with standard accumulation leak test methodology approved by the Test and Verification Control Panel via SSCN 5004 (Method VIII). These tests did indicate leakage rates in compliance with the individual specification requirements as detailed in SSCN 4652A.

Additionally, the Airlock has successfully completed a system–level gross leak test of the Oxygen Fluid System (pressure decay test at 2600 psia). This indicates the overall system meets its program end–item leakage rates.

Thus, all leak tests performed do provide confidence in the individual components' workmanship and additional acceptance leak tests are not warranted.

Finally, acceptance test procedures for all Carleton Technologies–supplied Oxygen and Nitrogen fluid components have been updated to perform leak testing via a vacuum chamber methodology in full compliance with SSP 41172.

PG3–212:

ITEMS:

Nitrogen Latching Motor Valve Part Number 683–16419–1 Serial Number 001001
Nitrogen Low Pressure Regulator/Relief Valve Part Number 683–16421–3 Serial Number 001002
Nitrogen/Oxygen Relief Valve Assembly Part Number 683–16425–1 Serial Number 001002
Nitrogen Manual Isolation Valve Part Number 683–16439–1 Serial Number 001002
Nitrogen/Oxygen Pressure Sensor Part Number 683–16443–1 Serial Numbers 001002, 001003, 001004, and 001005
Nitrogen Flow Restrictor Part Number 683–42331–2 Serial Number 001001
Oxygen Flow Restrictor Part Number 683–42331–1 Serial Number 001002
Oxygen Prebreathe Regulator/Relief Valve Part Number 683–16421–7 Serial Number 001001

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Leak Test shall not be required for the indicated Nitrogen and Oxygen fluid hardware units for initial delivery. Any rework of these units that would normally dictate performance of a leak test shall require leak testing of these units in full compliance with SSP 41172.

RATIONALE:

The indicated components except for the Nitrogen Low Pressure Regulator/Relief Valve underwent a stand–alone vacuum chamber leak test via a bell jar. During this vacuum chamber testing, no gross leaks were found; however, the 5–minute duration of the test was not sufficient to allow helium permeation through the component such that an accurate fine leakage rate could be obtained to determine compliance with the individual component specifications. The Nitrogen Low Pressure Regulator/Relief Valve underwent acceptance leak testing via use of a Bubble–O–Meter; however, evaluation subsequently determined that the performed methodology could not determine a fine leakage rate to the same accuracy as a vacuum chamber leak test performed in compliance with SSP 41172.

After evaluation of the installed locations of Oxygen and Nitrogen fluid components in the Airlock, NASA and Boeing–Huntsville leak test experts developed a resolution approach to perform an accumulation leak test on the installed flight Oxygen components except for the Oxygen Flow Restrictor, the Nitrogen/Oxygen Relief Valve Assembly, and one of the four Nitrogen/Oxygen Pressure Sensors in the Nitrogen distribution system. This test did indicate leakage rates in compliance with the individual specification requirements as detailed in SSCN 4652A. As these Oxygen components are identical in design and manufacture to the corresponding Nitrogen components above except for keying of the outlets, and the indicated Nitrogen components have been sufficiently tested to indicate no gross leak greater than 0.1 standard cubic centimeters per second, some confidence in the workmanship of these indicated Nitrogen components is inherent.

The Nitrogen/Oxygen Relief Valve Assembly, Nitrogen Flow Restrictor, and Oxygen Flow Restrictor are located only in the recharge lines of the respective Nitrogen and Oxygen distribution systems on the Airlock. As such, they are neither in continuous operation nor would any failure except a gross leak while in operation impact Space Station Nitrogen and Oxygen fluid recharge operation. As the original vacuum chamber test has indicated no gross leaks, use as–is disposition for these components is acceptable.

The Oxygen Prebreathe Regulator/Relief Valve was leak checked after the Oxygen Prebreathe Regulator Kit Assembly was damaged in shipping to KSC. The Varian leak detector used is capable of detecting leaks down to 1E–04 standard cubic centimeters per second in sniffer mode. White Sands Test Facility uses this probe–type detector as a leak/no leak indicator only by using it to sniff circumferentially around fittings, at welds, and at opening to items like Quick Disconnects. The complete Oxygen Prebreathe Regulator Kit (including the Oxygen Prebreathe Regulator/Relief Valve) was sniffed at all of the fittings, welds, and at the Quick Disconnect outlets and no detectable leak was found.

A Nitrogen Low Pressure Regulator/Relief Valve leakage to a magnitude of 0.1 standard cubic centimeters per second does not pose a concern or hazard as:

(a) All areas that have nitrogen components will be passively ventilated prior to crew working on/in those areas. This is to ensure the prevention of crew asphyxiation due to low oxygen partial pressure levels. Also for this specific area, the nitrogen component(s) are in an Airlock standoff in a difficult location for the crew to have unintended access. Thus, it is remote that a crewmember would become asphyxiated if the hazard ever exists.

(b) At Flight 7A, the Station itself will leak 0.0636 standard cubic centimeters per second, the BMP (Russian Trace Contaminant Control System) and Vozduhk will "leak" atmosphere at an average of 0.806 standard cubic centimeters per second, the USL CDRA will "leak" atmosphere at an average of 1.01 standard cubic centimeters per second, and for each EVA, at least 4.791 lbms will be lost to space. Therefore, the leakage of the Nitrogen Low Pressure Regulator/Relief Valve at the above stated rate will not cause the Station's total pressure to rise such that a positive pressure relief event would be required and hence a loss of Station resources.

Additionally, the Airlock has successfully completed a system–level gross leak test of the Oxygen Fluid System (pressure decay test at 2600 psia). This indicates the overall system meets its program end–item leakage rates. The Nitrogen Fluid System has undergone a system flow test at 3300 psia for verification of performance.

Further, the Nitrogen Latching Motor Valve, Nitrogen Manual Isolation Valve, and Nitrogen Low Pressure Regulator/Relief Valve had spares tested via a vacuum chamber methodology with a bell jar in compliance with SSP 41172. These spares are all manifested for launch on Flight 6A and will be available as on–orbit replacements.

Finally, acceptance test procedures for all Carleton Technologies–supplied Oxygen and Nitrogen fluid components have been updated to perform leak testing via a vacuum chamber methodology in full compliance with SSP 41172.

PG3–213:

ITEMS:

Oxygen Latching Motor Valve Part Number 683–16419–2
Oxygen Low Pressure Regulator/Relief Valve Part Number 683–16421–6
Oxygen Medium Pressure Regulator/Relief Valve Part Number 683–16421–5
Oxygen Prebreathe Regulator/Relief Valve Part Number 683–16421–7
Oxygen Manual Isolation Valve Part Number 683–16439–3
Oxygen Flow Restrictor Part Number 683–42331–1
Nitrogen Latching Motor Valve Part Number 683–16419–1
Nitrogen Low Pressure Regulator/Relief Valve Part Number 683–16421–3
Nitrogen/Oxygen Relief Valve Assembly Part Number 683–16425–1
Nitrogen Manual Isolation Valve Part Number 683–16439–1
Nitrogen/Oxygen Pressure Sensor Part Number 683–16443–1
Nitrogen Flow Restrictor Part Number 683–42331–2

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Leak Test shall not be performed on the indicated Nitrogen and Oxygen fluid hardware components.

RATIONALE:

The indicated components except for the Oxygen Low Pressure Regulator/Relief Valve, Nitrogen Low Pressure Regulator/Relief Valve, and the Oxygen Medium Pressure Regulator/Relief Valve underwent a stand–alone qualification vacuum chamber leak test via a bell jar. During this vacuum chamber testing, no gross leaks were found; however, the 5–minute duration of the test was not sufficient to allow helium permeation through the component such that an accurate fine leakage rate could be obtained to determine compliance with the individual component specifications. The Oxygen Low Pressure Regulator/Relief Valve, Nitrogen Low Pressure Regulator/Relief Valve, and the Oxygen Medium Pressure Regulator/Relief Valve underwent qualification leak testing via use of a Bubble–O–Meter; however, evaluation subsequently determined that the performed methodology could not determine a fine leakage rate to the same accuracy as a vacuum chamber leak test performed in compliance with SSP 41172.

An accumulation leak test was performed in place for the Oxygen Latching Motor Valve, Oxygen Low Pressure Regulator/Relief Valve, Oxygen Medium Pressure Regulator/Relief Valve, Oxygen Manual Isolation Valve, and three of the four Nitrogen/Oxygen Pressure Sensors in the Oxygen distribution system installed in the Airlock at the system level and for two Manual Isolation Valves external to the Airlock at the Tank ORU level. The tests were performed in compliance with standard accumulation leak test methodology approved by the Test and Verification Control Panel via SSCN 5004 (Method VIII). These tests did indicate leakage rates in compliance with the individual specification requirements as detailed in SSCN 4652A. All remaining components listed above are either similar in design and manufacture to Oxygen components which completed accumulation leak tests meeting their individual specification requirements or are located in Oxygen and Nitrogen recharge lines in limited operation as indicated in PG3–212. Based on the component–level leak tests performed, sufficient confidence in the design of these Oxygen and Nitrogen fluid components exist such that no additional qualification leak testing is warranted.

PG3–214:

ITEM:

Node 1 Ventilation Fan Inlet ORU Part Number SV811840 Serial Number 0001

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance. Component random vibration test levels and spectrums shall be the envelope of the following:
Paragraph 5.1.4.3B, Test Levels and Duration. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime contractor.

EXCEPTION:

The Acceptance Random Vibration Test of the Ventilation Fan Inlet shall be performed at a level of 3.1 grms with a maximum spectral density of 0.02 G^2 per Hz.

RATIONALE:

The ISS Ventilation Fan is 120 volts DC powered with motor control electronics to condition DC power and drive the motor. The fan motor is a typical wirewound brushless motor similar to fan motors in the Shuttle. The acceptance random vibration testing performed by the contractor Hamilton Standard on Serial Number 0001 was to a vibration level of 0.02 G^2 per Hz that was on contract at the time of the test. After Serial Number 0001 was delivered a change was processed that increased the Vibration Test Level to 0.04 G^2 per Hz.

The most sensitive component, the electronic controller, was tested to 0.057 G^2 per Hz. Also, Ventilation Fan Inlet ORU Serial Number 0002 was in process at Hamilton Standard and will have completed verification testing at a level of 0.04 G^2 per Hz prior to Launch of the Node containing the Ventilation Fan Inlet ORU Serial Number 0001. Additionally, a launch–induced failure could be detected by checkout on Flight 2A (fan is not normally used on Flight 2A). Operational Workarounds exist in the event the Ventilation Fan fails:

– Portable fans may be carried in by the crew;
– The Shuttle Air Revitalization System may be used to scrub the Node 1 air if required.

Finally, the motor used in the ventilator has good field reliability without an acceptance vibration test program.

PG3–215:

ITEM:

Active Common Berthing Mechanism Controller Panel Assembly
Part Number 2355260–1–1 Serial Numbers D0030, D0031, D0033, D0034 & D0048
Part Number 2355260–2–1 Serial Number D0037
Part Number 2355260–3–1 Serial Number D0047

SSP 41172 REQUIREMENT:

Paragraph 5.1.8, Burn–In Test, Component Acceptance.
Paragraph 5.1.8.3B, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The Acceptance Burn–In Test for the Controller Panel Assemblies identified shall be from –8 degrees F to 110 degrees F.

The Controller Panel Assemblies shall not be required to reach internal thermal equilibrium during the Acceptance Burn–In Test under Thermal Cycling conditions.

RATIONALE:

The flight Controller Panel Assemblies indicated were acceptance tested in accordance with a revision of the Acceptance Test Procedure that was not an ISS Program–baselined approved revision. The components indicated were bagged and purged with dry nitrogen to prevent potential damage from condensation during acceptance burn–in. This prevented the units from reaching the minimum and maximum operational temperatures of the acceptance thermal cycle test and achieving component internal thermal equilibrium. Components did experience the specified minimum and maximum temperatures during the acceptance thermal cycling testing in accordance with the Acceptance Test Procedure. Also, components did meet the minimum burn–in temperature sweep of 100 degrees F and received 13 temperature cycles instead of 10 required by SSP 41172.

There is low risk for the components in question incurring an early life failure. No early component failures were found during the complete Controller Panel Assembly component–level Qualification and Acceptance testing programs (41 flight and 5 qualification units delivered). In addition, each Controller Panel Assembly will receive several more hours of additional operation during the assembly–level Active Common Berthing Mechanism acceptance test performed following installation on the element.

PG3–216:

ITEM:

Active Common Berthing Mechanism Controller Panel Assembly Part Numbers 2355260–1–1, 2355260–2–1, and 2355260–3–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.8, Burn–In Test, Component Acceptance.
Paragraph 5.1.8.2, Test Description. ALL REQUIREMENTS.

EXCEPTION:

The Acceptance Burn–In Test for the Controller Panel Assemblies identified shall be powered off during transition from hot to cold temperature.

RATIONALE:

Components were not powered and monitored during transition from the hot temperature to the cold temperature during acceptance burn–in testing under thermal cycling conditions. The minimum Controller Panel Assembly transition rate could not be maintained when the unit is powered while transitioning to the low temperature in the Honeywell thermal chambers as the Controller Panel Assembly has a mass of 35 pounds and generates approximately 45 watts while powered.

There is low risk for the components in question incurring an early life failure. No early component failures were found during the complete Controller Panel Assembly component–level Qualification and Acceptance testing programs (41 flight and 5 qualification units delivered). The components were powered and monitored during transition from cold to hot temperatures. This would likely have detected temperature transition defects in the components. All Controller Panel Assemblies have accumulated a minimum of 100 hours powered burn–in duration. In addition, each Controller Panel Assembly not installed and on–orbit will receive several more hours of additional operation during the assembly–level Active Common Berthing Mechanism acceptance test performed on the element.

Additionally, Common Berthing Mechanism Controller Panel Assemblies installed on Node 1 and USL have successfully completed 11 berths/deberths without anomaly through flight 6A.

PG3–217:

ITEM:

Active Common Berthing Mechanism Controller Panel Assembly Part Numbers 2355260–1–1, 2355260–2–1, and 2355260–3–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.
Paragraph 5.1.3.2, Test Description. ALL REQUIREMENTS.

EXCEPTION:

The Acceptance Thermal Cycling Test for the Controller Panel Assemblies identified shall be powered off during transition from hot to cold temperature.

RATIONALE:

Components were not powered during transition from the hot temperature to the cold temperature during the acceptance thermal cycling test. The minimum Controller Panel Assembly transition rate cannot be maintained when the unit is powered while transitioning to the low temperature in the Honeywell thermal chambers as the Controller Panel Assembly has a mass of 35 pounds and generates approximately 45 watts while powered.

There is low risk for the components in question containing material or workmanship defects. The components were powered and monitored during transition from cold to hot temperatures. This would likely have detected temperature transition defects in the components. Controller Panel Assemblies have been installed and on–orbit for 2.5 years and have operated without anomaly through flight 6A. Additionally, Common Berthing Mechanism Controller Panel Assemblies on Node 1 and USL have successfully completed 11 berths/deberths to date.

PG3–218:

ITEM:

Active Common Berthing Mechanism Controller Panel Assembly Part Numbers 2355260–1–1, 2355260–2–1, and 2355260–3–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.2, Test Description. ALL REQUIREMENTS.

EXCEPTION:

The Acceptance Thermal Vacuum Test for the Controller Panel Assemblies identified shall be powered off during transition from hot to cold temperature.

RATIONALE:

Components were not powered during transition from the hot temperature to the cold temperature during the acceptance thermal vacuum test. The minimum Controller Panel Assembly transition rate cannot be maintained when the unit is powered while transitioning to the low temperature in the Honeywell thermal chambers as the Controller Panel Assembly has a mass of 35 pounds and generates approximately 45 watts while powered.

There is low risk for the components in question containing material or workmanship defects. The components were powered and monitored during transition from cold to hot temperatures. This would likely have detected temperature transition defects in the components. Controller Panel Assemblies have been installed and on–orbit for 2.5 years and have operated without anomaly through flight 6A. Additionally, Common Berthing Mechanism Controller Panel Assemblies on Node 1 and USL have successfully completed 11 berths/deberths to date.

PG3–219:

ITEMS:

Oxygen Latching Motor Valve Part Number 683–16419–2 Serial Numbers 001001, 001002, and 001003
Nitrogen Latching Motor Valve Part Number 683–16419–1 Serial Numbers 001001 and 001002
Oxygen Manual Isolation Valve Part Number 683–16439–3 Serial Numbers 001001, 001002, 001003, 001004, 001005, 001006, and 001007
Nitrogen Manual Isolation Valve Part Number 683–16439–1 Serial Numbers 001002, 001003, and 001004

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Leak Test for internal leakage shall not be required for the indicated Nitrogen and Oxygen fluid hardware units for initial delivery. Any rework of these units that would normally dictate performance of a leak test shall require leak testing of these units in full compliance with SSP 41172.

RATIONALE:

The indicated components underwent an internal leak test using helium from a pressure source connected to the inlet, and a mass spectrometer connected directly to the outlet. During this testing, no gross leaks were found; however, the 5–minute duration of the test was not sufficient to allow helium permeation through the component such that an accurate fine leakage rate could be obtained to determine compliance with the individual component specifications.

During Acceptance Testing of a returned Manual Isolation Valve and Latching Motor Valve, the units were tested six times for internal leakage. In all tests the reading would appear stable after 5 minutes and generally remain below 1E–4 sccs Helium. Additionally, in tests performed the reading was stable within 10 minutes and the values ranged from 1.4 to 2.5E–4 sccs Helium. This shows that a 5–minute duration test is adequate to provide confidence in the individual components' workmanship. However, acceptance test procedures for the items listed will be updated to perform internal leak testing with a minimum dwell time of 30 minutes in accordance with approved SSCN 5004 for Method X, paragraph 5.1.7.2J of SSP 41172.

The original leak rate requirement change of 1E–4 sccs Helium to 1E–3 sccs Helium was concurred by the Vehicle System Integration Panel on June 26, 2001. There is no safety hazard associated with the new internal leak rate requirement.

In the case of internal leakage through these components, the O2 or N2 is not released into the cabin but instead stays in the system. Therefore the safety concerns with external leakage of O2 or N2 are not applicable to this exception. In every application of these valves, there is a redundant method of isolation. On the High Pressure Gas Tank ORU, a quick disconnect is capped off downstream of the Manual Isolation Valve. In each leg of the O2/N2 distribution system inside the Airlock, a Manual Isolation Valve and a Latching Motor Valve are in series providing redundant isolation capability. The nominal position of all these components is open; therefore, internal leakage would only apply when these components are closed for maintenance or other contingency operations.

PG3–220:

ITEMS:

Oxygen Latching Motor Valve Part Number 683–16419–2
Nitrogen Latching Motor Valve Part Number 683–16419–1
Oxygen Manual Isolation Valve Part Number 683–16439–3
Nitrogen Manual Isolation Valve Part Number 683–16439–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Internal Leak Test shall not be performed on the indicated Nitrogen and Oxygen fluid hardware components.

RATIONALE:

The indicated components underwent a standalone qualification internal leak test using helium from a pressure source connected to the inlet, and a mass spectrometer connected directly to the outlet. During this testing, no gross leaks were found; however, the 5–minute duration of the test was not sufficient to allow helium permeation through the component such that an accurate fine leakage rate could be obtained to determine compliance with the individual component specifications.

During Acceptance Testing of a returned Manual Isolation Valve and Latching Motor Valve, the units were tested six times for internal leakage. In all tests the reading would appear stable after 5 minutes and generally remain below 1E–4 sccs Helium. Additionally, in tests performed the reading was stable within 10 minutes and the values ranged from 1.4 to 2.5E–4 sccs Helium. This shows that a 5–minute duration test is adequate to provide confidence in the individual components' workmanship.

In the case of internal leakage through these components, the O_2 or N_2 is not released into the cabin but instead stays in the system. Therefore the safety concerns with external leakage of O_2 or N_2 are not applicable to this exception. In every application of these valves, there is a redundant method of isolation. On the High Pressure Gas Tank ORU, a quick disconnect is capped off downstream of the Manual Isolation Valve. In each leg of the O_2/N_2 distribution system inside the Airlock, a Manual Isolation Valve and a Latching Motor Valve are in series providing redundant isolation capability. The nominal position of all these components is open; therefore, internal leakage would only apply when these components are closed for maintenance or other contingency operations.

Based on the leak tests performed, sufficient confidence in the design of these Oxygen and Nitrogen fluid components exist such that no additional qualification leak testing is warranted.

PG3–221:

ITEM:

Pump Bypass Assembly Part Number 2351169–1–1

SSP 41172 REQUIREMENT:

Paragragh 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperature of the Pump Bypass Assembly shall be 33 degrees F during qualification thermal cycle testing.

RATIONALE:

The minimum predicted operating temperature of the Pump Bypass Assembly is 55 degrees F. Flight Pump Bypass Assemblies were tested while operating to a minimum temperature of 33 degrees F. However, the qualification unit was tested to the same temperature limit. As the Pump Bypass Assembly employs water as the operational fluid, environmental testing to temperatures lower than 32 degrees F would cause the fluid to freeze. Thus, the resulting thermal expansion would damage the Pump Bypass Assembly. Yet, since the qualification thermal cycle temperature did envelop (without margin) the acceptance thermal cycle minimum temperature limit, the risk associated with this exception is minimal.

PG3–222:

ITEM:

Pump Bypass Assembly, Part Number: 2351169–1–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Random Vibration test on the Pump Bypass Assembly will not be performed.

RATIONALE:

The Pump Bypass Assembly does not have any electronics as it is a check valve only. There are no close tolerances on the Pump Bypass Assembly. As documented in Honeywell (Allied Signal) Stress Report 41–11535, Revision D, the Pump Bypass Assembly was non–responsive to the energy levels input during qualification random vibration testing. Thus, so far as providing value as a test of the component stresses and part functionality, the workmanship screen obtained by performing an acceptance random vibration test would be minimal.

PG3–223:

ITEM:

Loop Crossover Assembly Part Number 2353198–101

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration, Component Qualification.
Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The Loop Crossover Assembly will not be powered on and monitored during the qualification random vibration testing.

RATIONALE:

The Loop Crossover Assembly contains electronics consisting of an electro–mechanical actuator, and does not utilize a printed circuit board with a card edge connector, or have high density populated circuit cards. Circuit Card Assemblies (CCAs) are mounted at all four corners and contain a central mounting point, with standoffs between CCAs, to limit deflections due to vibrational loads. A limited portion of the components would be energized, as the Loop Crossover Assembly is in standby mode. This is a non–enabled, non–operating condition, where minimal components, including mainly the position indication signals, are powered. The only parameter available for monitoring would be input current at standby and “isolated” or “cross–connected” position indication. The Loop Crossover Assembly would not be operational during the vibration test. The Loop Crossover Assembly would only be in standby mode. Also, electronic components are contained within an solder–sealed cover, and are not accessible for inspection at any point during or after vibration test. The Loop Crossover Assembly is not normally powered during launch vibrational loads, and to do this takes the unit out of its normal operating functional mode.

Evidence was provided showing that these components did undergo reliability acceptance testing at a lower level assembly, which did high and low temperature vibration for one minute in one axis, providing further workmanship screening on the electronics.

In the event that there is any operational malfunction due to some factor that would have been screened out by power–on vibration testing of the unit, it should be noted that each of these components have manual override capability. Backing this up, should any component fail mechanically, there is sufficient redundancy in the system to ensure that the Thermal Control System loops are not shut down. The Loop Crossover Assembly is not classified by reliability and maintainability analysis to be of criticality 1.

PG3–224:

ITEM:

Loop Crossover Assembly Part Number 2353198–101

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration, Component Acceptance.
Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

The Loop Crossover Assembly will not be powered on and monitored during the acceptance random vibration testing.

RATIONALE:

The Loop Crossover Assembly contains electronics consisting of an electro–mechanical actuator, and does not utilize a printed circuit board with a card edge connector, or have high density populated circuit cards. Circuit Card Assemblies (CCAs) are mounted at all four corners and contain a central mounting point, with standoffs between CCAs, to limit deflections due to vibrational loads. A limited portion of the components would be energized, as the Loop Crossover Assembly is in standby mode. This is a non–enabled, non–operating condition, where minimal components, including mainly the position indication signals, are powered. The only parameter available for monitoring would be input current at standby and "isolated" or "cross–connected" position indication. The Loop Crossover Assembly would not be operational during the vibration test. The Loop Crossover Assembly would only be in standby mode. Also, electronic components are contained within an solder–sealed cover, and are not accessible for inspection at any point during or after vibration test. The Loop Crossover Assembly is not normally powered during launch vibrational loads, and to do this takes the unit out of its normal operating functional mode.

Evidence was provided showing that these components did undergo reliability acceptance testing at a lower level assembly, which did high and low temperature vibration for one minute in one axis, providing further workmanship screening on the electronics.

In the event that there is any operational malfunction due to some factor that would have been screened out by power–on vibration testing of the unit, it should be noted that each of these components have manual override capability. Backing this up, should any component fail mechanically, there is sufficient redundancy in the system to ensure that the Thermal Control System loops are not shut down. The Loop Crossover Assembly is not classified by reliability and maintainability analysis to be of criticality 1.

PG3–225:

ITEM:

Loop Crossover Assembly, Part Number 2353198–101

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperature of the Loop Crossover Assembly shall be 33 degrees F during qualification thermal cycle testing.

RATIONALE:

The minimum predicted operating temperature of the Loop Crossover Assembly is 55 degrees F. Loop Crossover Assemblies were tested while operating to a minimum temperature of 33 degrees F. The qualification unit was tested to the same temperature limit of 33 degrees F. As the Loop Crossover Assembly employs water as the operational fluid, environmental testing to temperatures lower than 32 degrees F would cause the fluid to freeze. Thus, the resulting thermal expansion would damage the Loop Crossover Assembly. Yet, since the qualification thermal cycle temperature did envelop (without margin) the acceptance thermal cycle minimum temperature limit, the risk associated with this exception is minimal.

The Material Identification and Usage List shows that the mechanical portions of the assembly are insensitive to a non–operational low temperature of 0 degrees F.

In the event that there is any operational malfunction due to some factor that would have been screened out by non–operational cold testing of the unit, then it should be noted that the components within the Thermal Control System loops have manual operational capability. Backing this up, should any component fail mechanically, there is sufficient redundancy in the system to ensure that the Thermal Control System loops are not shut down. None of these components are classified by reliability and maintainability analysis to be of criticality 1.

PG3–226:

ITEM:

Pirani Gauge Transducer Part Number 220F01084–001

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Leakage Test will not be performed on the Pirani Gauge Transducer.

RATIONALE:

The Pirani Gauge Transducer is criticality 3 hardware and is only operated during Vacuum System operation. Limited ORU–level leakage testing was performed; however, no conclusions can be made from the ORU–level test performed on the Qualification unit as Helium levels no greater than the mass spectrometer background level was recorded during the duration (three minutes) testing that was performed.

Prior to the vacuum system regression testing conducted at Kennedy Space Center Nov. – Dec. 1999, a system–level vacuum retention test was completed. The system–level vacuum retention rate was measured and compared to the allowable leakage rate as documented on the Vacuum System envelope drawing.

Vacuum Exhaust System was measured at 4.1E–6 sccs; Vacuum Resource System was measured at 4.5E–5 sccs. These are both significantly better than the allowable system level leakage rate of 4.0E–4 sccs for each system.

Upon review of the test procedure and this system–level leak data, Boeing and NASA have judged acceptable the ORUs of the Vacuum System without need for additional leakage testing.

PG3–227:

ITEM:

Pirani Gauge Transducer Part Number 220F01084–001

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Leakage Test will not be performed on the Pirani Gauge Transducer. Any subsequent leakage tests on flight units shall require testing in full compliance with SSP 41172.

RATIONALE:

The Pirani Gauge Transducer is criticality 3 hardware and is only operated during Vacuum System operation. Limited ORU–level leakage testing was performed; however, no conclusions can be made from the ORU–level test performed on the flight units as Helium levels no greater than the mass spectrometer background level was recorded during the duration (three minutes) testing that was performed.

Prior to the vacuum system regression testing conducted at Kennedy Space Center Nov. – Dec. 1999, a system–level vacuum retention test was completed. The system–level vacuum retention rate was measured and compared to the allowable leakage rate as documented on the Vacuum System envelope drawing.

Vacuum Exhaust System was measured at 4.1E–6 sccs; Vacuum Resource System was measured at 4.5E–5 sccs. These are both significantly better than the allowable system level leakage rate of 4.0E–4 sccs for each system.

Upon review of the test procedure and this system–level leak data, Boeing and NASA have judged acceptable the ORUs of the Vacuum System without need for additional leakage testing.

Any subsequent leakage tests on the flight units or additional procurements shall require testing in full compliance with SSP 41172.

PG3–228:

ITEM:

Internal Self–Sealing Fluid Quick Disconnect Couplings Boeing Part Numbers 683–16348 and 683–15179

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.
Paragraph 4.2.11.3B, Test Levels and Duration, Method II – The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

All items already delivered are qualified based on the successful characterization test in accordance with SSP 41172, paragraph 4.2.11.2, item B, Method II, of June 2000.

RATIONALE:

A characterization leak test was performed in June 2000 by Parker–Stratoflex on one pair of 683–16348 water QD, one female oxygen low pressure QD, and one pair of oxygen and nitrogen high pressure QDs in accordance with the test duration of SSP 41172, paragraph 4.2.11.2, item B, Method II. Although the test was performed for four hours in accordance with SSP 41172, the leak test results indicated that all the leakage rates of tested QDs stabilized within 45 minutes to one hour. This characterization test did not support the rationale for performing the original Qualification and Acceptance tests for 5 minutes, but a multiplication factor equal to the ratio of the leakage rate of four hours to the leakage rate of 5 minutes was applied to all the QDs that were tested for 5 minutes. The QDs used on ISS are similar in material and type of construction to the QDs used on the Space Shuttle. The testing done at Parker Symmetrics is consistent with testing done on Space Shuttle QDs. Both the Shuttle and Station QDs are Proof Pressure and MDP Pressure tested 100 percent for leakage in both the mated and de–mated modes. The specifications the Shuttle and Station are made and tested to are different, but the method of acceptance test is similar. No significant leak anomalies associated with manufacturing and acceptance testing have been reported for Shuttle QDs. This testing is sufficient to determine that the QDs can be used as–is for flight.

System engineers took into account in their leak specification the 1–decade rule as a safety factor to account for uncertainties such as operator technique, method/instrument errors, and normal variations in manufacturing. The calculated results show that 97 percent of the QDs are within this acceptable safety factor range and the O2 concentration limits for the USL and Airlock are within the respective specifications.

In addition, the calculated leakage rate ratio that was used (four hours leakage rate/5 minute leakage rate) is no longer the preferred methodology for assessing the acceptance data of the QDs since a characterization test indicated that the leakage rates of tested QDs stabilized within 45 minutes to one hour. If a calculated ratio of (one hour leakage rate/5 minute leakage rate) were used, all the QDs would fall well within the 1–decade rule.

Furthermore, during this characterization leak test, Parker–Stratoflex performed the leak testing at external pressure at 10 millitorr due to the limitation of the mass spectrometer, which would not allow the test to be performed at 1 millitorr. The difference between 1 millitorr and 10 millitorr is insignificant and therefore repeating the test at 1 millitorr would not provide any significant benefit.

Oxygen and Nitrogen High Pressure QDs
Oxygen maximum calculated leakage rate (3.19E–4 He) exceeded the specification requirement (1E–4 He) only by a small amount. Given that other oxygen components in the same area have a much higher specification leakage (i.e. 1E–3 He), this exceedance is considered to be in the noise range.

Vacuum Low Pressure QDs
The only condition under which external leakage into the cabin might be a concern is if a failure causing loss of vacuum in the VES occurred while a payload was venting a toxic gas. This is an extremely unlikely event, but if such an event did occur and toxic gas was trapped in the VES at a pressure above ISS atmosphere, leakage into the cabin would be at a minuscule quantity. Even with the worst case QD, the buildup of toxic gases in the cabin atmosphere would not be a concern.

Water Low Pressure QDs
Water coolant leakage is very small and the calculated total leakage for one year is equivalent to 8.2 cubic inch. The average calculated leakage rate for water QDs is 1.5E–4 He, which means practically no water leakage.

Oxygen Low Pressure QDs
Calculated leakage rate for five of the fifteen Oxygen QDs exceeded the specification requirement (2.8E–04 He) by a factor of two. Given that these QDs are not concentrated in any one area and we have other components which have a higher specification leakage (i.e. 1E–3 He), this leakage exceedance is acceptable. Calculated leakage rate for two of the fifteen Oxygen QDs exceeded the specification requirement by a factor of 9.3. By applying the 1–decade rule as a safety factor to account for uncertainties such as operator technique, method/instrument errors, and normal variations in manufacturing, ALL of the oxygen QDs are within this acceptable safety factor range.

Nitrogen Low Pressure QDs
Nitrogen leakage is extremely small and it is very doubtful that any excessive levels of nitrogen will build up behind the closeouts.

Procedures are in place that require any closeout area containing nitrogen lines/components to be vented with portable fans and/or let stand with the closeout off for a short period prior to performing maintenance.

PG3–229:

ITEM:

Internal Self–Sealing Fluid Quick Disconnect Couplings Boeing Part Numbers 683–16348 and 683–15179

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.
Paragraph 5.1.7.3B, Test Levels and Duration, Method II – The external test pressure shall be 0.001 Torr (0.133 Pa) or less, and the duration shall be four hours (for equipment that is operational in orbit for more than one day).

EXCEPTION:

All items already delivered are qualified based on the successful characterization test in accordance with SSP 41172, paragraph 5.1.7.3, item B, Method II, of June 2000.

RATIONALE:

A characterization leak test was performed in June 2000 by Parker–Stratoflex on one pair of 683–16348 water QD, one female oxygen low pressure QD, and one pair of oxygen and nitrogen high pressure QDs in accordance with the test duration of SSP 41172, paragraph 4.2.11.2, item B, Method II. Although the test was performed for four hours in accordance with SSP 41172, the leak test results indicated that all the leakage rates of tested QDs stabilized within 45 minutes to one hour. This characterization test did not support the rationale for performing the original Qualification and Acceptance tests for 5 minutes, but a multiplication factor equal to the leakage rate ratio of four hours to the leakage rate of 5 minutes was applied to all the QDs that were tested for 5 minutes. The QDs used on ISS are similar in material and type of construction to the QDs used on the Space Shuttle. The testing done at Parker Symmetrics is consistent with testing done on Space Shuttle QDs. Both the Shuttle and Station QDs are Proof Pressure and MDP Pressure tested 100 percent for leakage in both the mated and de–mated modes. The specifications the Shuttle and Station are made and tested to are different, but the method of acceptance test is similar. No significant leak anomalies associated with manufacturing and acceptance testing have been reported for Shuttle QDs. The QDs that did not meet the requirement will be used as is based on the above rationale. This testing is sufficient to determine that the QDs can be used as–is for flight.

System engineers took into account in their leak specification the 1–decade rule as a safety factor to account for uncertainties such as operator technique, method/instrument errors, and normal variations in manufacturing. The calculated results show that 97 percent of the QDs are within this acceptable safety factor range and the O2 concentration limits for the USL and Airlock are within the respective specifications. A comparative study has been conducted on Node 1 and Node 2. It has been determined that the leakage rates are acceptable due to the low amount of oxygen and nitrogen Quick Disconnects used in these modules as compared to the Airlock. The Airlock Y4 closeout volume is considered the worst–case volume (approximately 19.2 cubic ft and includes seven major O2 components) which is the worst volume to cumulative leakage rate. Therefore, there is no action required on the Node 1, 2, or 3 Quick Disconnects. MSFC/Hamilton Sundstrand will conduct an analysis using the Parker–Stratoflex test data to determine QD integrity for each specific application. If a QD cannot be shown by analysis to provide adequate leakage protection, a higher–level leak test will be performed at the integrated ORU level or flexhose assembly level by MSFC/Hamilton Sundstrand. NASDA has completed a system leak test and determined all leak requirements were successfully verified. No Quick Disconnect hardware changes are planned for the JEM.

In addition, the calculated leakage ratio that was used (four hours leakage rate/5 minute leakage rate) is no longer the preferred methodology for assessing the acceptance data of the QDs since a characterization test indicated that the leakage rates of tested QDs stabilized within 45 minutes to one hour. If a calculated ratio of one hour leakage rate/5 minute leakage rate were used, all the QDs would fall well within the 1–decade rule.

Furthermore, during this characterization leak test, Parker–Stratoflex performed the leak testing at external pressure at 10 millitorr due to the limitation of the mass spectrometer, which would not allow the test to be performed at 1 millitorr. The difference between 1 millitorr and 10 millitorr is insignificant and therefore repeating the test at 1 millitorr would not provide any significant benefit.

The Acceptance Test Procedure will include leak test duration of 15 minutes at external pressure of .01 torr. Each Quick Disconnect will be tested in the unmated configuration and the mated configuration. The mated configuration will utilize a surrogate Quick Disconnect. The surrogate Quick Disconnect does not have an external seal and will verify each seal independently. Vacuum Quick Disconnects (Category 3) will be tested in the same manner; however, 5 minutes will be used instead of 15 minutes. The pass/fail criteria are to remain the same. The 15–minute duration is based on risk assessment from the system–level analysis, cost, and technical concurrence of NASA, Boeing, and Parker–Stratoflex.

Oxygen and Nitrogen High Pressure QDs
Oxygen maximum leakage rate (3.19E–4 He) exceeded calculated requirement (specification 1E–4 He) only by a small amount. Given that other oxygen components in the same area have a much higher specification leakage (i.e., 1E–3 He), this exceedance is considered to be in the noise range.

Vacuum Low Pressure QDs
The only condition under which external leakage into the cabin might be a concern is if a failure causing loss of vacuum in the VES occurred while a payload was venting a toxic gas. This is an extremely unlikely event, but if such an event did occur and toxic gas was trapped in the VES at a pressure above ISS atmosphere, leakage into the cabin would be at a minuscule quantity. Even with the worst case QD, the buildup of toxic gases in the cabin atmosphere would not be a concern.

Water Low Pressure QDs
Water coolant leakage is very small and the calculated total leakage for one year is equivalent to 8.2 cubic inch. The average calculated leakage rate for water QDs is 1.5E–4 He, which means practically no water leakage.

Oxygen Low Pressure QDs
Calculated leakage rate for five of the fifteen oxygen QDs exceeded the specification requirement (2.8E–4 He) by a factor of two. Given that these QDs are not concentrated in any one area and we have other components which have a higher specification leakage (i.e., 1E–3 He), this leakage exceedance is acceptable. Two of the fifteen oxygen QDs exceeded the calculated leakage rate by a factor of 9.3. By applying the 1–decade rule as a safety factor to account for uncertainties such as operator technique, method/instrument errors, and normal variations in manufacturing, ALL of the oxygen QDs are within this acceptable safety factor range.

Nitrogen Low Pressure QDs
Nitrogen leakage is extremely small and it is very doubtful that any excessive levels of nitrogen will build up behind the closeouts.

Procedures are in place that require any closeout area containing nitrogen lines/components to be vented with portable fans and/or let stand with the closeout off for a short period prior to performing maintenance.

APPENDIX D GOVERNMENT–FURNISHED EQUIPMENT APPROVED EXCEPTIONS

The following is a list of exceptions to this document taken by Government–Furnished Equipment (GFE) vendors. The exceptions to this document in no way eliminate the vendor's responsibility for showing compliance to the requirements of the applicable specification.

GFE–01:

ITEM:

Airlock Servicing & Performance Checkout Equipment (SPCE) as follows:
Power Supply Assembly Part Number SEG39128211–303
Battery Charger Assembly Part Number SEG39128212–301
Battery Stowage Assembly Part Number SEG39128213–301
Fluid Pumping Unit Part Number SEG39128310–301
Umbilical Interface Assembly Part Number SEG39128214–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.8, Burn–In Test, Component Acceptance

Paragraph 5.1.8.3C, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

Allow accelerated burn–in hours at an elevated temperature to be combined with hours accumulated at room temperature to produce an "equivalent 300–hour" burn–in. An acceleration factor of F=5.26 for accelerated burn–in conducted at 110 degrees F is acceptable (i.e., each hour of accelerated burn–in conducted at 110 degrees F is equivalent to 5.26 hours at ambient temperature).

RATIONALE:

An accelerated burn–in can be characterized by the following equation:

$$F = \exp[(Ea/K)(1/Ta - 1/Tbi)]$$

where F = Time Acceleration Factor
Ea = Activation Energy (eV)
K = Boltzmann's constant (8.625E–05 eV/K)
Ta = ambient temperature (degrees K)
(For the purposes of this application, ambient temperature shall be considered to be 295.8 degrees K).
Tbi = elevated burn–in temperature (degrees K).

An activation energy of 0.6 eV was selected based on the makeup of the electronic components utilized and for an accelerated burn–in temperature of 110 degrees F, an acceleration factor of 5.26 is obtained. In other words, every hour of operation at 110 degrees F is equivalent to 5.26 hours of testing at 70 degrees F.

GFE–02:

ITEM:

Early Portable Computer System (EPCS) Laptop Assembly as follows:
Computer (760ED) Part Number SDG39129270–301
Payload Ground Support Computer (PGSC)
Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification

Paragraph 4.2.2.5, Depress/Repress Vacuum Requirements. ALL REQUIREMENTS.

EXCEPTION:

The EPCS Laptop Assembly will perform operational qualification depress/repress test in the range of 9.5 psi to 14.2 psi.

RATIONALE:

The EPCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760ED and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 48–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, "shake" testing, as well as functional burn–ins. The decision to perform a 48–hour burn in for EPCS was based on the previous experience of the Shuttle PGSC project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle's IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR since September, 97 and has experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. In addition there are several units available on board station that could assume the EPCS role in the event of a failure. On Flight 2A, two units are flown and one is required for the EPCS function. At the end of 2A.1, four units remain on board with one of these required for the EPCS function. On Flight 3A, five units remain on board with two required for the EPCS function. On Flight 4A, seven units remain on board and four are required for the EPCS function. Beginning with Flight 5A, a new platform is flown and the 760EDs are no longer used for EPCS functions. Given the shuttle experience, successful flight record of this hardware, and the availability of unit on board, additional acceptance testing is not deemed necessary.

In addition, it is known from both operational and evaluation testing that the COTS hardware will be damaged when depressed to less than 1 psi. Further, there are no EPCS requirements for the hardware to operate beyond the range of 10.1 to 14.7 psi.

GFE–03:

ITEM:

EPCS Laptop Assembly as follows:
Computer (760ED) Part Number SDG39129270–301
PGSC Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification

Paragraph 4.2.3.3, Thermal Cycling Test, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The EPCS Laptop Assembly will perform an operational test of a single cycle to manufacturer's specifications of 50 degrees F to 95 degrees F (10 degrees C to 35 degrees C).

RATIONALE:

The EPCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760ED and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 48–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, "shake" testing, as well as functional burn–ins. The decision to perform a 48–hour burn in for EPCS was based on the previous experience of the Shuttle PGSC project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle's IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR since September, 97 and has experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. In addition there are several units available on board station that could assume the EPCS role in the event of a failure. On Flight 2A, two units are flown and one is required for the EPCS function. At the end of 2A.1, four units remain on board with one of these required for the EPCS function. On Flight 3A, five units remain on board with two required for the EPCS function. On Flight 4A, seven units remain on board and four are required for the EPCS function. Beginning with Flight 5A, a new platform is flown and the 760EDs are no longer used for EPCS functions. Given the shuttle experience, successful flight record of this hardware, and the availability of unit on board, additional acceptance testing is not deemed necessary.

In addition, testing beyond manufacturer's specifications will damage this COTS equipment. Further, there are no EPCS requirements for the hardware to operate beyond manufacturer's specifications.

GFE–04:

ITEM:

EPCS Laptop Assembly as follows:
Computer (760ED) Part Number SDG39129270–301
PGSC Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification

(1) Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Qualification Random Vibration tests will not be performed on the EPCS Laptop Assembly.

RATIONALE:

The EPCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760ED and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 48–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, “shake” testing, as well as functional burn–ins. The decision to perform a 48–hour burn in for EPCS was based on the previous experience of the Shuttle PGSC project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle’s IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR since September, 97 and has experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. In addition there are several units available on board station that could assume the EPCS role in the event of a failure. On Flight 2A, two units are flown and one is required for the EPCS function. At the end of 2A.1, four units remain on board with one of these required for the EPCS function. On Flight 3A, five units remain on board with two required for the EPCS function. On Flight 4A, seven units remain on board and four are required for the EPCS function. Beginning with Flight 5A, a new platform is flown and the 760EDs are no longer used for EPCS functions. Given the shuttle experience, successful flight record of this hardware, and the availability of unit on board, additional acceptance testing is not deemed necessary.

In addition, the COTS equipment (hard drive, floppy drive, and CDROM) will be damaged if operating during vibration testing. Further, the hardware is not exposed to significant on–orbit vibration (it is not mounted). Operational flight experience has shown that the hardware (not operating) is capable of withstanding launch and landing vibration environments (through 4 EPCS Shuttle flights and over 150 similarly–launched models (PGSCs)).

GFE–05:

ITEM:

EPCS Laptop Assembly as follows:
Computer (760ED) Part Number SDG39129270–301
PGSC Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance

(1) Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Thermal Cycling tests will not be performed on the EPCS Laptop Assembly.

RATIONALE:

The EPCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760ED and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 48–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, “shake” testing, as well as functional burn–ins. The decision to perform a 48–hour burn in for EPCS was based on the previous experience of the Shuttle PGSC project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle’s IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR since September, 97 and has experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. In addition there are several units available on board station that could assume the EPCS role in the event of a failure. On Flight 2A, two units are flown and one is required for the EPCS function. At the end of 2A.1, four units remain on board with one of these required for the EPCS function. On Flight 3A, five units remain on board with two required for the EPCS function. On Flight 4A, seven units remain on board and four are required for the EPCS function. Beginning with Flight 5A, a new platform is flown and the 760EDs are no longer used for EPCS functions. Given the shuttle experience, successful flight record of this hardware, and the availability of unit on board, additional acceptance testing is not deemed necessary.

In addition, testing beyond manufacturer’s specifications will damage this COTS equipment. Further, there are no EPCS requirements for the hardware to operate beyond manufacturer’s specifications.

GFE–06:

ITEM:

EPCS Laptop Assembly as follows:
Computer (760ED) Part Number SDG39129270–301
PGSC Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance

(1) Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Random Vibration tests will not be performed on the EPCS Laptop Assembly.

RATIONALE:

The EPCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760ED and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 48–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, “shake” testing, as well as functional burn–ins. The decision to perform a 48–hour burn in for EPCS was based on the previous experience of the Shuttle PGSC project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle’s IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR since September, 97 and has experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. In addition there are several units available on board station that could assume the EPCS role in the event of a failure. On Flight 2A, two units are flown and one is required for the EPCS function. At the end of 2A.1, four units remain on board with one of these required for the EPCS function. On Flight 3A, five units remain on board with two required for the EPCS function. On Flight 4A, seven units remain on board and four are required for the EPCS function. Beginning with Flight 5A, a new platform is flown and the 760EDs are no longer used for EPCS functions. Given the shuttle experience, successful flight record of this hardware, and the availability of unit on board, additional acceptance testing is not deemed necessary.

In addition, the COTS equipment (hard drive, floppy drive, and CDROM) will be damaged if operating during vibration testing. Further, the hardware is not exposed to significant on–orbit vibration (it is not mounted). Operational flight experience has shown that the hardware (not operating) is capable of withstanding launch and landing vibration environments (through 4 EPCS Shuttle flights and over 150 similarly–launched models (PGSCs)). Finally, the JSC manufacturing and quality control processes has a proven track record of manufacturing flight units without a single in–flight failure that could be attributed to a defect that vibration testing would have detected.

GFE–07:

ITEM:

EPCS Laptop Assembly as follows:
Computer (760ED) Part Number SDG39129270–301
PGSC Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.8, Burn–In Test, Component Acceptance

Paragraph 5.1.8.3C, Test Levels and Duration. For constant temperature burn–in (either at ambient or accelerated via elevated temperature), the total operating time shall be equivalent to 300 hours at ambient temperature.

EXCEPTION:

The Acceptance Burn–In test on the EPCS Laptop Assembly will be 48 hours.

RATIONALE:

The EPCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760ED and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 48–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, “shake” testing, as well as functional burn–ins. The decision to perform a 48–hour burn in for EPCS was based on the previous experience of the Shuttle PGSC project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle’s IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR since September, 97 and has experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. In addition there are several units available on board station that could assume the EPCS role in the event of a failure. On Flight 2A, two units are flown and one is required for the EPCS function. At the end of 2A.1, four units remain on board with one of these required for the EPCS function. On Flight 3A, five units remain on board with two required for the EPCS function. On Flight 4A, seven units remain on board and four are required for the EPCS function. Beginning with Flight 5A, a new platform is flown and the 760EDs are no longer used for EPCS functions. Given the shuttle experience, successful flight record of this hardware, and the availability of unit on board, additional acceptance testing is not deemed necessary.

In addition to the COTS manufacturer burn–in tests and the 48–hour acceptance burn–in test, JSC performs additional functional tests upon receipt of the units, both prior to and after manufacturing into flight units. Finally, the JSC manufacturing and quality control processes has a proven track record of manufacturing flight units without a single in–flight failure that could be attributed to a defect that extended burn–in testing would have detected.

GFE–08:

ITEM:

Wireless Information System (WIS) components as follows:
Shuttle–based WIS Remote Sensor Unit (RSU) Part Numbers SEG16102888–301 and SEG16102888–303
Shuttle–based WIS Cargo Element Antenna Assembly Part Number SEG16102891–301
15 ft. Antenna Cable Assembly Part Number 41ZN2ZN2180.0
25–ft. Antenna Cable Assembly Part Number 41ZN2ZN2180.0
WIS Battery Pack Assembly Part Number IVC–0060–04–004

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification

Paragraph 4.2.2.3C, Test Levels and Duration. The time required to reach thermal equilibrium shall be determined by pre–qualification analysis or test or by measuring the component's internal thermal response during an extended dwell period of not less than 12 hours at each temperature extreme of the first qualification thermal vacuum cycle.

EXCEPTION:

The dwell time of the WIS components in the first cycle of Qualification Thermal Vacuum Test will be 1 hour after temperature stabilization.

RATIONALE:

The WIS units have two internal temperature sensors. One sensor is located on the main controller/CPU PC board at the top of the unit, and the other sensor is inside the battery pack at the bottom of the unit. With these two internal temperature sensors, the internal temperature of the WIS units can be accurately determined. Once temperate stabilization has been verified, remaining at that temperature for 12 hours will add little value to the test.

GFE–09:

ITEM:

Wireless Information System (WIS) components as follows:
Shuttle–based WIS RSU Part Numbers SEG16102888–301 and SEG16102888–303
Shuttle–based WIS Cargo Element Antenna Assembly Part Number SEG16102891–301
15 ft. Antenna Cable Assembly Part Number 41ZN2ZN2180.0
25–ft. Antenna Cable Assembly Part Number 41ZN2ZN2180.0
WIS Battery Pack Assembly Part Number IVC – 0060–04–004
Network Control Unit WIS Part Number SEG16102890–301
NCU to PGSC Parallel Interface Cable Part Number IVC–0060–04–014
Internal WIS Remote Sensor Unit Part Number SEG16102889–301
Strain Gauge Extension Cable Assembly Part Number IVC–0060–04–002
Accelerometer Assembly Part Number IVC–0060–04–012
Accelerometer Cable Assembly Part Number IVC–0060–04–013

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification

Paragraph 4.2.3.3C, Test Levels and Duration. The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total.

EXCEPTION:

The duration of the WIS Qualification Thermal Cycle Test shall be six cycles.

RATIONALE:

A duration of six thermal cycles is consistent with testing for noncritical hardware as proposed in SSCN 1379.

Depending on the test chamber capabilities and hardware components, the transition rate will be increased above the minimum temperature ramp rate of 1.0 degrees F per minute to provide a more effective screen.

GFE–10:

ITEM:

Wireless Information System (WIS) Battery Pack Assembly Part Number IVC–0060–04–004

SSP 41172 REQUIREMENT:

Paragraph 4.2.10, Pressure Test, Component Qualification

Paragraph 4.2.10.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 4.2.11, Leakage Test, Component Qualification

Paragraph 4.2.11.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The Qualification Pressure and Leakage Tests on the WIS Battery Pack Assembly will be performed by an alternative test method at the vendor.

RATIONALE:

The WIS battery pack contains three Shuttle–certified lithium BCXII primary D–cells connected in series with special safeguards to ensure safety. The cells are certified for Shuttle use per a test program for Shuttle use that includes the following tests:

A. Each cell is exposed to 160 + 10 degrees F for 2 hours during Acceptance Testing;

B. Sample cells are exposed to 200 + 10 degrees F for 2 hours during Acceptance Testing. and checked for leakage, and functional damage of shrink wrap and terminal assembly;

C. Load test for 90 minutes during Acceptance Testing with any cell below 3.4 volts rejected,

D. Dimensional and weight checks during Acceptance Testing;

E. X–Ray for positive pin defects and corrosion during Acceptance Testing; and

F. Sample cells are exposed to 300 + 5 degrees F for 1 hour during Certification Testing and visually inspected for damage, electrolyte leakage, bulging, and glass–to–metal seal damage.

Sample sizes for lot certification are 9 percent for capacity discharge, 3 percent for high temperature exposure, 4 percent for short circuit, and 3 percent for extreme temperature exposure.

In addition to the above test, the batteries will be certified as a part of the WIS Remote Sensing Unit in accordance with SSP 41172. Battery packs will be visually inspected after each environmental test.

GFE–11:

ITEM:

Wireless Information System (WIS) components as follows:
Shuttle–based WIS RSU Part Numbers SEG16102888–301 and SEG16102888–303
Shuttle–based WIS Cargo Element Antenna Assembly Part Number SEG16102891–301
15 ft. Antenna Cable Assembly Part Number 41ZN2ZN2180.0
25–ft. Antenna Cable Assembly Part Number 41ZN2ZN2180.0
WIS Battery Pack Assembly Part Number IVC – 0060–04–004
Network Control Unit WIS Part Number SEG16102890–301
NCU to PGSC Parallel Interface Cable Part Number IVC–0060–04–014
Internal WIS Remote Sensor Unit Part Number SEG16102889–301
Strain Gauge Extension Cable Assembly Part Number IVC–0060–04–002
Accelerometer Assembly Part Number IVC–0060–04–012
Accelerometer Cable Assembly Part Number IVC–0060–04–013

SSP 41172 REQUIREMENT:

Paragraph 4.2.12, Electromagnetic Compatibility Test, Component Qualification

Paragraph 4.2.12.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The following Electromagnetic Compatibility Test referenced in SSP 30237 will not be performed on the WIS components: CE01, CE03, CE07, LE01, RS02, RS03, CS01, CS02, CS06 and LE01.

Only approved emissions test RE02 will be performed for the electromagnet compatibility test.

RATIONALE:

All “C” or LE01 levels are not applicable to WIS Assembly since it is battery powered. An engineering evaluation test will be performed for information only to determine susceptibility. The WIS is a noncritical system and can be powered off if a problem is detected.

GFE–12:

ITEM:

Wireless Information System (WIS) components as follows:
Shuttle–based WIS RSU Part Numbers SEG16102888–301 and SEG16102888–303
Shuttle–based WIS Cargo Element Antenna Assembly Part Number SEG16102891–301
15 ft. Antenna Cable Assembly Part Number 41ZN2ZN2180.0
25–ft. Antenna Cable Assembly Part Number 41ZN2ZN2180.0
WIS Battery Pack Assembly Part Number IVC – 0060–04–004
Network Control Unit WIS Part Number SEG16102890–301
NCU to PGSC Parallel Interface Cable Part Number IVC–0060–04–014
Internal WIS Remote Sensor Unit Part Number SEG16102889–301
Strain Gauge Extension Cable Assembly Part Number IVC–0060–04–002
Accelerometer Assembly Part Number IVC–0060–04–012
Accelerometer Cable Assembly Part Number IVC–0060–04–013

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance

Paragraph 5.1.3.3C, Test Levels and Duration. The minimum number of thermal cycles shall be eight.

EXCEPTION:

The duration of the WIS Acceptance Thermal Cycle Test shall be three cycles.

RATIONALE:

A duration of three thermal cycles is consistent with testing for noncritical hardware as proposed in SSCN 1379.

Depending on the test chamber capabilities and hardware components, the transition rate will be increased above the minimum temperature ramp rate of 1.0 degrees F per minute to provide a more effective screen.

GFE–13:

ITEM:

Wireless Information System (WIS) Battery Pack Assembly Part Number IVC–0060–04–004

SSP 41172 REQUIREMENT:

Paragraph 5.1.6, Pressure Test, Component Acceptance

(1) Paragraph 5.1.6.3, Test Levels. ALL REQUIREMENTS.

Paragraph 5.1.7, Leakage Test, Component Acceptance

(1) Paragraph 5.1.7.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The Acceptance Pressure and Leakage Tests on the WIS Battery Pack Assembly will be performed by an alternative test method at the vendor.

RATIONALE:

The WIS battery pack contains three Shuttle–certified lithium BCXII primary D–cells connected in series with special safeguards to ensure safety. The cells are certified for Shuttle use per a test program for Shuttle use that includes the following tests:

A. Each cell is exposed to 160 + 10 degrees F for 2 hours during Acceptance Testing;

B. Sample cells are exposed to 200 + 10 degrees F for 2 hours during Acceptance Testing. and checked for leakage, and functional damage of shrink wrap and terminal assembly;

C. Load test for 90 minutes during Acceptance Testing with any cell below 3.4 volts rejected,

D. Dimensional and weight checks during Acceptance Testing;

E. X–Ray for positive pin defects and corrosion during Acceptance Testing; and

F. Sample cells are exposed to 300 + 5 degrees F for 1 hour during Certification Testing and visually inspected for damage, electrolyte leakage, bulging, and glass–to–metal seal damage.

Sample sizes for lot certification are 9 percent for capacity discharge, 3 percent for high temperature exposure, 4 percent for short circuit, and 3 percent for extreme temperature exposure.

In addition to the above test, the batteries will be certified as a part of the WIS Remote Sensing Unit in accordance with SSP 41172. Battery packs will be visually inspected after each environmental test.

GFE–14:

ITEM:

SPCE as follows:

Airlock SPCE Battery Charger Assembly (BCA) Part Number SEG39128212–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The Qualification Thermal Cycle Test for the Airlock SPCE BCA will be performed 10 degrees F beyond the maximum and minimum acceptance temperature extremes.

RATIONALE:

The Acceptance Thermal Cycle test temperature range of the Airlock SPCE BCA will be from 40 degrees F to 140 degrees F. The Qualification Thermal Cycle test of the Airlock SPCE BCA will be performed from 30 degrees F to 150 degrees F. This is due to components within the BCA are commercial parts which cannot withstand the minimum required qualification temperature sweep of 140 degrees F without potential damage to the component. Test tolerances will be controlled to less than the ±5.4 degrees F maximum required in accordance with SSP 41172 to ensure that the flight units will not be exposed to temperatures higher than that experienced by the qualification unit.

GFE–15:

ITEM:

SPCE as follows:

Airlock SPCE BCA Part Number SEG39128212–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification

Paragraph 4.2.3.4, Supplementary Requirements. Functional tests shall be conducted, as a minimum, at the maximum predicted temperature plus 20 degrees F (11.1 degrees C) and at the minimum predicted temperature minus 20 degrees F (11.1 degrees C) during the first and last thermal cycles and after return of the component to ambient.

EXCEPTION:

Functional tests during the Qualification Thermal Cycle Test for the Airlock SPCE BCA shall be conducted at 45 degrees F and 120 degrees F.

RATIONALE:

The Qualification Thermal Cycle test temperature range of the Airlock SPCE BCA will be from 30 degrees F to 150 degrees F. However, functional tests during this test will not be performed at these extremes. This is due to components within the BCA are commercial parts whose temperature range do not allow proper operation outside the temperature range of 45 degrees F to 120 degrees F. The operational performance temperature limits for the BCA are 65 degrees F and 80 degrees F; therefore, the qualification (and acceptance) test temperature limits are driven by minimum workmanship screening levels. The qualification thermal cycle test is intended to qualify the design (with margin) for the acceptance thermal cycle test. During acceptance thermal cycle testing for workmanship screening, sufficient monitoring of the component is necessary to detect any intermittence or failures. Full operational performance at workmanship test levels is not necessary. During qualification, full functional tests shall be conducted on the BCA twice during the first cycle (prior to and after exposure to the minimum and maximum temperature extremes) and twice during the last cycle (also prior to and after exposure to the minimum and maximum temperature extremes). The BCA will be powered and monitored sufficiently throughout the full qualification thermal cycle test, including the 30 degrees F and 150 degrees F temperature extremes, to detect any intermittence or failures. Thus, margin above and below the operational temperature extremes will be demonstrated.

GFE–16:

ITEM:

SPCE as follows:

Airlock SPCE BCA Part Number SEG39128212–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance

Paragraph 5.1.3.4, Supplementary Requirements. Functional tests shall be conducted during the first and last operating thermal cycles after the dwell at the maximum and minimum predicted operating temperatures and after return of the component to ambient.

EXCEPTION:

Functional tests during the Acceptance Thermal Cycle Test for the Airlock SPCE BCA shall be conducted at 50 degrees F and 110 degrees F.

RATIONALE:

The Acceptance Thermal Cycle test temperature range of the Airlock SPCE BCA will be from 40 degrees F to 140 degrees F. However, functional tests during this test will not be performed at these extremes. This is due to components within the BCA are commercial parts whose temperature range do not allow proper operation outside the temperature range of 45 degrees F to 120 degrees F. The operational performance temperature limits for the BCA are 65 degrees F and 80 degrees F; therefore, the acceptance test temperature limits are driven by minimum workmanship screening levels. During acceptance thermal cycle testing for workmanship screening, sufficient monitoring of the component is necessary to detect any intermittence or failures. Full operational performance at workmanship test levels is not necessary. During acceptance, full functional tests shall be conducted on the BCA twice during the first cycle (prior to and after exposure to the minimum and maximum temperature extremes) and twice during the last cycle (also prior to and after exposure to the minimum and maximum temperature extremes). The BCA will be powered and monitored sufficiently throughout the full qualification thermal cycle test, including the 40 degrees F and 140 degrees F temperature extremes, to detect any intermittence or failures. Thus, margin above and below the operational temperature extremes will be demonstrated.

GFE–17:

ITEM:

Early Communication (ECOMM) components as follows:
Command and Telemetry Processor (CTP) Part Number SEG39130534–301
Radio Frequency Power Distribution Box (RFPDB) Part Number SEG39130725–301
Antenna Part Numbers SEG39130674–301 and SEG39130674–303
Transceiver Part Number SEG190–136110–004

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification

Paragraph 4.2.3.3C, Duration. The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total.

EXCEPTION:

The duration of the Qualification Thermal Cycle Tests for the CTP, RFPDB, Antenna, and Transceiver shall be three cycles.

RATIONALE:

The ECOMM is a Criticality 3 system, the loss of which will not put crew or vehicle at additional risk.

The designs of the CTP and RFPDB use highly reliable, pre–screened, military–standard grade parts with a temperature range that can meet the environment. The temperature range planned for the test includes the 20–degree margin on both the high and low end of the operational temperature range. Antenna and transceiver use Commercial–Off–The–Shelf parts; however, redundancy for the antenna and in–flight backup for the transceiver are available.

The qualification test plan for the ECOMM system was part of the ECOMM Critical Design Review and received no comments from the ISS pertaining to number of cycles.

The ECOMM has a short mission lifecycle (2A through completion of 5A Lab activation). The reduced number of cycles optimizes GFE Project test time and cost with respect to achieving the intent of the SSP 41172 requirements while meeting the intent of a criticality 3 system.

GFE–18:

ITEM:

Early Communication (ECOMM) components as follows:
Command and Telemetry Processor (CTP) Part Number SEG39130534–301
Radio Frequency Power Distribution Box (RFPDB) Part Number SEG39130725–301
Antenna Part Numbers SEG39130674–301 and SEG39130674–303
Transceiver Part Number SEG190–136110–004

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance

Paragraph 5.1.3.3C, Duration. The minimum number of temperature cycles shall be eight.

EXCEPTION:

The duration of the Acceptance Thermal Cycle Tests for the CTP and RFPDB shall be 1.5 cycles.

The duration of the Acceptance Thermal Cycle Tests for the Antenna and Transceiver shall be three cycles.

RATIONALE:

The ECOMM is a Criticality 3 system, the loss of which will not put crew or vehicle at additional risk.

The designs of the CTP and RFPDB use highly reliable, pre–screened, military–standard grade parts with a temperature range that can meet the environment. The temperature range planned for the test includes the 20–degree margin on both the high and low end of the operational temperature range. Antenna and transceiver use Commercial–Off–The–Shelf parts; however, redundancy for the antenna and in–flight backup for the transceiver are available.

The qualification test plan for the ECOMM system was part of the ECOMM Critical Design Review and received no comments from the ISS pertaining to number of cycles.

The ECOMM has a short mission lifecycle (2A through completion of 5A Lab activation). The reduced number of cycles optimizes GFE Project test time and cost with respect to achieving the intent of the SSP 41172 requirements while meeting the intent of a criticality 3 system.

GFE–19:

ITEM:

Early Communication (ECOMM) components as follows:
Command and Telemetry Processor (CTP) Part Number SEG39130534–301
Radio Frequency Power Distribution Box (RFPDB) Part Number SEG39130725–301
Antenna Part Numbers SEG39130674–301 and SEG39130674–303
Transceiver Part Number SEG190–136110–004

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification

Paragraph 4.2.5.3, Test Levels and Duration. The test duration in each of the three orthogonal axes shall be three times the expected flight exposure time to the maximum predicted level and spectrum or three times the component random vibration acceptance test time if that is greater, but not less than three minutes per axis.

EXCEPTION:

The duration of the Qualification Random Vibration Tests for the CTP, RFPDB, Antenna, and Transceiver shall be two minutes.

RATIONALE:

The ECOMM is a Criticality 3 system, the loss of which will not put crew or vehicle at additional risk.

System is launched unpowered in the Shuttle middeck, where it will experience a very benign vibration environment. Once on–orbit, ECOMM also will not be expected to operate in a high vibration environment.

For each ORU axis, the “Qualification for Acceptance Test” (QAVT) duration of two minutes is combined with the “Acceptance–for–Flight Vibration Test” (AFVT) duration of one minute to meet the intent of SSP 41172.

GFE–20:

ITEM:

Early Communication (ECOMM) components as follows:
Command and Telemetry Processor (CTP) Part Number SEG39130534–301
Radio Frequency Power Distribution Box (RFPDB) Part Number SEG39130725–301
Antenna Part Numbers SEG39130674–301 and SEG39130674–303
Transceiver Part Number SEG190–136110–004

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification

Paragraph 4.2.5.3, Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:

B. Acceptances test levels and spectrum plus test tolerances.

EXCEPTION:

The qualification random vibration level applied to each ORU axis shall be through a combination of ECOMM QAVT (0.067 g2/Hz) and QAVT (0.04 g2/Hz) levels.

RATIONALE:

The ECOMM is a Criticality 3 system, the loss of which will not put crew or vehicle at additional risk.

The Shuttle–to–Payloads launch vibration environment in the Middeck area is very benign (0.03 g2/Hz). Shuttle vibration levels translated to inside the Middeck Stowage Bag/packing foam is substantially less. SSP 41172 levels applied to ECOMM represent unnecessary environmental extremes not to be seen by the flight ORUs.

The ECOMM has a short mission lifecycle (2A through completion of 5A Lab activation). The reduced vibration level optimizes GFE Project test time and cost with respect to achieving the intent of the SSP 41172 requirements while meeting the intent of a criticality 3 system.

GFE–21:

ITEM:

Early Communication (ECOMM) components as follows:
Command and Telemetry Processor (CTP) Part Number SEG39130534–301
Radio Frequency Power Distribution Box (RFPDB) Part Number SEG39130725–301
Antenna Part Numbers SEG39130674–301 and SEG39130674–303
Transceiver Part Number SEG190–136110–004

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

Qualification Random Vibration Tests without Power–On and monitoring is permitted for the CTP, RFPDB, Antenna, and Transceiver.

RATIONALE:

The ECOMM is a Criticality 3 system, the loss of which will not put crew or vehicle at additional risk.

System is launched unpowered in the Shuttle middeck, where it will experience a very benign vibration environment. Once on–orbit, ECOMM also will not be expected to operate in a high vibration environment.

Functional tests will be performed before and after vibration in each axis. This testing, combined with the (powered) thermal cycling testing, provides adequate verification of workmanship/latent defects.

ECOMM GSE cabling was not designed to be attached during vibration testing. This could add significant unwarranted stress on the flight unit connector shells/pins.

GFE–22:

ITEM:

Early Communication (ECOMM) components as follows:
Command and Telemetry Processor (CTP) Part Number SEG39130534–301
Radio Frequency Power Distribution Box (RFPDB) Part Number SEG39130725–301
Antenna Part Numbers SEG39130674–301 and SEG39130674–303
Transceiver Part Number SEG190–136110–004

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

Acceptance Random Vibration Tests without Power–On and monitoring is permitted for the CTP, RFPDB, Antenna, and Transceiver.

RATIONALE:

The ECOMM is a Criticality 3 system, the loss of which will not put crew or vehicle at additional risk.

System is launched unpowered in the Shuttle middeck, where it will experience a very benign vibration environment. Once on–orbit, ECOMM also will not be expected to operate in a high vibration environment.

Functional tests will be performed before and after vibration in each axis. This testing, combined with the (powered) thermal cycling testing, provides adequate verification of workmanship/latent defects.

ECOMM GSE cabling was not designed to be attached during vibration testing. This could add significant unwarranted stress on the flight unit connector shells/pins.

GFE–23:

ITEM:

Early Communication (ECOMM) components as follows:
Command and Telemetry Processor (CTP) Part Number SEG39130534–301
Radio Frequency Power Distribution Box (RFPDB) Part Number SEG39130725–301
Antenna Part Numbers SEG39130674–301 and SEG39130674–303
Transceiver Part Number SEG190–136110–004

SSP 41172 REQUIREMENT:

Paragraph 5.1.8, Burn–In Test, Component Acceptance

Paragraph 5.1.8.3C, Test Levels and Duration. For constant temperature burn–in (either at ambient or accelerated via elevated temperature), the total operating time shall be equivalent to 300 hours at ambient temperature.

EXCEPTION:

The Acceptance Burn–In test on the CTP, RFPDB, Antenna, and Transceiver will be 200 hours.

RATIONALE:

The ECOMM is a Criticality 3 system, the loss of which will not put crew or vehicle at additional risk.

The ECOMM has a short mission lifecycle (2A through completion of 5A Lab activation). The reduced duration of the Burn–In test optimizes GFE Project test time and cost with respect to achieving the intent of the SSP 41172 requirements while meeting the intent of a criticality 3 system.

GFE–24:

ITEM:

Respiratory Support Pack (RSP) Part Number SEG42103650–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.

Paragraph 4.2.3.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 4.2.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Qualification Thermal Cycling tests will not be performed on the Respiratory Support Pack.

RATIONALE:

The RSP is comprised of a Commercial–Off–The–Shelf (COTS) ventilator, the Autovent 2000, various COTS respiratory therapy supplies, and a softpack constructed primarily from Lexan, Nomex, and Velcro. The COTS hardware was modified sparingly to comply with requirements for space flight. These changes include materials changes, material processing changes (such as annealing), and structural reinforcement of components. None of these changes affected the operation of this device. The Autovent 2000 is pneumatically driven and contains no electrical or electronic components and no soldering joints. Its timing is controlled by a series of flow restrictors, volume chambers, and a spool valve which controls the on/off state of the device. The spool valve switches the On/off State when enough GOx pressure accumulates to overcome a small spring force keeping the valve in that state.

The RSP contains no electrical or electronic components. Hot/cold tests are performed by the vendor for 8 hours per day for 3 weeks. Wyle Laboratories also conducts 8 thermal cycles on unit. Thus, a full Qualification Thermal Cycle test will not screen any unforeseen materials or manufacturing defects.

Due to a lack of an appropriate category, the RSP is classified and tested as a Fluid/Pressure system in accordance with SSP 41172. However, the RSP differs from a classic, sealed fluid/pressure system as it is not permanently pressurized and it is designed to leak during proper operation. The RSP has much commonality with a moving mechanical assembly, and thermal cycling testing is not required for this type of hardware.

GFE–25:

ITEM:

Respiratory Support Pack (RSP) Part Number SEG42103650–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Thermal Cycling tests will not be performed on the Respiratory Support Pack.

RATIONALE:

The RSP is comprised of a Commercial–Off–The–Shelf (COTS) ventilator, the Autovent 2000, various COTS respiratory therapy supplies, and a softpack constructed primarily from Lexan, Nomex, and Velcro. The COTS hardware was modified sparingly to comply with requirements for space flight. These changes include materials changes, material processing changes (such as annealing), and structural reinforcement of components. None of these changes affected the operation of this device. The Autovent 2000 is pneumatically driven and contains no electrical or electronic components and no soldering joints. Its timing is controlled by a series of flow restrictors, volume chambers, and a spool valve which controls the on/off state of the device. The spool valve switches the On/off State when enough GOx pressure accumulates to overcome a small spring force keeping the valve in that state.

The RSP contains no electrical or electronic components. Thus, a full Acceptance Thermal Cycle test will not screen any unforeseen materials or manufacturing defects.

Due to a lack of an appropriate category, the RSP is classified and tested as a Fluid/Pressure system in accordance with SSP 41172. However, the RSP differs from a classic, sealed fluid/pressure system as it is not permanently pressurized and it is designed to leak during proper operation. The RSP has much commonality with a moving mechanical assembly, and thermal cycling testing is not required for this type of hardware.

GFE–26:

ITEM:

Respiratory Support Pack (RSP) Part Number SEG42103650–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.

Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Qualification Random Vibration test on the Respiratory Support Pack is replaced by the manufacturer's sine vibration, random vibration, and shock tests.

RATIONALE:

Manufacturer's–performed testing includes:

Random Vibration Testing from 20 Hertz to 500 Hertz at 0.02g^2/Hertz;

Sine Vibration Testing over 10 Hertz to 500 Hertz at 1 g; and

Shock testing using 100g's for 6 ms 1/2 sine waveform.

Additionally, Vendor performed off limits testing during initial design consisting of tumbling the unit down flights of stairs.

The RSP has a long–use history in helicopters, military vehicles, and ambulances with no failures attributed to vibration. The only moving parts in the RSP pressure system are the spool valve, regulator piston, the timer and volume knobs, a lever assembly and piston in the patient valve assembly, and the Symmetric Quick Disconnect. Additionally, all screws are locked down with either lock washers or Locktite. Any Material and Manufacturing associated defects will be detected during pressure testing of the hardware.

GFE–27:

ITEM:

Respiratory Support Pack (RSP) Part Number SEG42103650–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.

Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Random Vibration tests will not be performed on the Respiratory Support Pack.

RATIONALE:

The RSP is comprised of a Commercial–Off–The–Shelf (COTS) ventilator, the Autovent 2000, various COTS respiratory therapy supplies, and a softpack constructed primarily from Lexan, Nomex, and Velcro. The COTS hardware was modified sparingly to comply with requirements for space flight. These changes include materials changes, material processing changes (such as annealing), and structural reinforcement of components. None of these changes affected the operation of this device. The Autovent 2000 is pneumatically driven and contains no electrical or electronic components and no soldering joints. Its timing is controlled by a series of flow restrictors, volume chambers, and a spool valve which controls the on/off state of the device. The spool valve switches the On/off State when enough GOx pressure accumulates to overcome a small spring force keeping the valve in that state.

The RSP has a long–use history in helicopters, military vehicles, and ambulances with no failures attributed to vibration. The only moving parts in the RSP pressure system are the spool valve, regulator piston, the timer and volume knobs, a lever assembly and piston in the patient valve assembly, and the Symmetric Quick Disconnect. Additionally, all screws are locked down with either lock washers or Locktite. Any material and manufacturing associated defects will be detected during pressure testing of the hardware.

GFE–28:

ITEM:

Respiratory Support Pack (RSP) Part Number SEG42103650–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.11, Leakage Test, Component Qualification.

Paragraph 4.2.11.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The Qualification Leakage Test will not be performed on the Respiratory Support Pack assembly except for mechanical fittings which will be assembled and tested in accordance with Air Force T.O 00–25–223, Integrated Pressure Systems and Components (Portable and Installed), section 1–528 through section 1–573, as applicable.

RATIONALE:

The RSP is comprised of a Commercial–Off–The–Shelf (COTS) ventilator, the Autovent 2000, various COTS respiratory therapy supplies, and a softpack constructed primarily from Lexan, Nomex, and Velcro. The COTS hardware was modified sparingly to comply with requirements for space flight. These changes include materials changes, material processing changes (such as annealing), and structural reinforcement of components. None of these changes affected the operation of this device. The Autovent 2000 is pneumatically driven and contains no electrical or electronic components and no soldering joints. Its timing is controlled by a series of flow restrictors, volume chambers, and a spool valve which controls the on/off state of the device. The spool valve switches the On/off State when enough GOx pressure accumulates to overcome a small spring force keeping the valve in that state.

Components of the Respiratory Support Pack will be Qualification Leakage Tested by the following test method at the White Sands Test Facility:

(1) Component pressurized to test level;

(2) Pressurized gas source and seal input line is removed;

(3) Component is accepted if the pressure in the article does not fall more than 5–10 psi in 5 minutes.

As indicated, individual components used by the vendor during unit manufacture are leak tested prior to RSP Autovent assembly. The RSP system is designed to leak to facilitate rapid transition between the on and off states. It must be able to vent the pressure in certain gas passageways in order to rapidly switch states; therefore, it must “leak” during operation. The spool valve “rides” on a gas bearing during operation resulting in a small amount of leakage. The hardware is not pressurized while stowed and must be deployed and manually connected to a pressurized GOx source. The entire pressurized operational profile is 72 hours or less. Finally, the hardware will be refurbished every two years or after every use, whichever occurs first. Thus, the additional cost to conduct a RSP assembly leak test in accordance with SSP 41172 will produce little additional knowledge.

GFE–29:

ITEM:

Respiratory Support Pack (RSP) Part Number SEG42103650–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.7, Leakage Test, Component Acceptance.

Paragraph 5.1.7.3, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The Acceptance Leakage Test will not be performed on the Respiratory Support Pack assembly except for mechanical fittings which will be assembled and tested in accordance with Air Force T.O 00–25–223, Integrated Pressure Systems and Components (Portable and Installed), section 1–528 through section 1–573, as applicable.

RATIONALE:

The RSP is comprised of a Commercial–Off–The–Shelf (COTS) ventilator, the Autovent 2000, various COTS respiratory therapy supplies, and a softpack constructed primarily from Lexan, Nomex, and Velcro. The COTS hardware was modified sparingly to comply with requirements for space flight. These changes include materials changes, material processing changes (such as annealing), and structural reinforcement of components. None of these changes affected the operation of this device. The Autovent 2000 is pneumatically driven and contains no electrical or electronic components and no soldering joints. Its timing is controlled by a series of flow restrictors, volume chambers, and a spool valve which controls the on/off state of the device. The spool valve switches the On/off State when enough GOx pressure accumulates to overcome a small spring force keeping the valve in that state.

Components of the Respiratory Support Pack will be Acceptance Leakage Tested by the following test method at the White Sands Test Facility:

(1) Component pressurized to test level;

(2) Pressurized gas source and seal input line is removed;

(3) Component is accepted if the pressure in the article does not fall more than 5–10 psi in 5 minutes.

As indicated, individual components used by the vendor during unit manufacture are leak tested prior to RSP Autovent assembly. The RSP system is designed to leak to facilitate rapid transition between the on and off states. It must be able to vent the pressure in certain gas passageways in order to rapidly switch states; therefore, it must “leak” during operation. The spool valve “rides” on a gas bearing during operation resulting in a small amount of leakage. The hardware is not pressurized while stowed and must be deployed and manually connected to a pressurized GOx source. The entire pressurized operational profile is 72 hours or less. Finally, the hardware will be refurbished every two years or after every use, whichever occurs first. Thus, the additional cost to conduct a BSP assembly leak test in accordance with SSP 41172 will produce little additional knowledge.

GFE–30:

ITEM:

Compound Specific Analyzer-Combustion Products Part Number SED46115801–301
Sampling Pump Part Number SED46115803–301

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests: When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification). This requires that the protoflight hardware be electrically energized and monitored during protoflight thermal cycling tests and functional tests be conducted at the temperature levels as indicated in paragraph 6.1.1B.

Paragraph 6.1.1.B. For the thermal cycling test, the temperature cycles shall be conducted at 10 degrees F (5.6 degrees C) beyond the maximum and minimum predicted temperatures.

EXCEPTION:

The Compound Specific Analyzer-Combustion Products and Sampling Pump will be unpowered outside of maximum and minimum operational temperatures during protoflight thermal cycling testing.

The Compound Specific Analyzer-Combustion Products and Sampling Pump functional tests will be performed at the maximum and minimum operational temperatures during protoflight thermal cycling testing.

RATIONALE:

The CSA–CP only has 4 wires and very few connectors with solder joints. Visual inspection of electronics is performed at the vendor prior to assembly.

The Operational limit of the Compound Specific Analyzer-Combustion Products is from 32 degrees F to 104 degrees F. Operation outside these limits is not recommended by the vendor and could possibly alter electrochemical sensor performance. The CSA–CP is only used inside the crew cabin at a temperature of 60 degrees F to 80 degrees F. The performed protoflight thermal cycling test had a ramp rate of approximately 100 degrees F per hour which provides an effective stress screen. Additionally, a spare CSA–CP is maintained on–orbit at all times. Based on vendor data, the mean time between failures is approximately 7 years.

Thousands of Sampling Pumps are currently in industrial use, demonstrating the workmanship of the instrument design. As this hardware is protoflight, damage due to testing outside of vendor recommended limits from 32 degrees F to 104 degrees F would impact hardware delivery to Program.

GFE–31:

ITEM:

Russian Depressurization Pump Cable and Junction Box Part Number SEG33110830–801

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.

Paragraph 6.1.1.B, Thermal Cycling Test. The minimum number of cycles shall be eight.

EXCEPTION:

The Protoflight Thermal Cycling Test on the Russian Depressurization Pump Cable and Junction Box shall be three cycles.

RATIONALE:

The Russian Depressurization Pump Cable and Junction Box does not contain any electronics (only a printed circuit board with copper conductors). Three thermal cycles during the protoflight thermal cycling test are sufficient to screen for workmanship.

GFE–32:

ITEM:

Moisture Removal Kit (MRK) Part Number SEG11100311

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.

Paragraph 4.2.3.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 4.2.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Qualification Thermal Cycling tests will not be performed on the Moisture Removal Kit.

RATIONALE:

A visual inspection of all soldering joints and added modifications to the internal MRK fan to verify no loose parts or connections will be performed on the MRK assembly.

The internal MRK fan and desiccant bag are one–time use items as qualified by test. The operational requirement is that the MRK fan has a single twenty–four hour operating period with the Node closed and the Orbiter still docked. The temperature testing for the fan was from 60 degrees F to 90 degrees F over the twenty–four hour operating period. A full thermal cycle is beyond the MRK operating environment. The manufacturer has sold approximately 50,000 units with less than 1 percent returned. Additionally, a post–modification PDA is performed and a PIA just prior to manifest for a flight.

Finally, redundancy is provided as four fans and eight desiccant assemblies are provided on–orbit while test shows only three are required to meet the requirement.

GFE–33:

ITEM:

Moisture Removal Kit (MRK) Part Number SEG11100311

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Thermal Cycling tests will not be performed on the Moisture Removal Kit.

RATIONALE:

A visual inspection of all soldering joints and added modifications to the internal MRK fan to verify no loose parts or connections will be performed on the MRK assembly.

The internal MRK fan and desiccant bag are one–time use items as qualified by test. The operational requirement is that the MRK fan has a single twenty–four hour operating period with the Node closed and the Orbiter still docked. The temperature testing for the fan was from 60 degrees F to 90 degrees F over the twenty–four hour operating period. A full thermal cycle is beyond the MRK operating environment. The manufacturer has sold approximately 50,000 units with less than 1 percent returned. Additionally, a post–modification PDA is performed and a PIA just prior to manifest for a flight.

Finally, redundancy is provided as four fans and eight desiccant assemblies are provided on–orbit while test shows only three are required to meet the requirement.

GFE–34:

ITEM:

Moisture Removal Kit (MRK) Part Number SEG11100311

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.

Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Random Vibration test will not be performed on the Moisture Removal Kit.

RATIONALE:

A visual inspection of all soldering joints and added modifications to the internal MRK fan to verify no loose parts or connections will be performed on the MRK assembly.

A Qualification Random Vibration test at the specified levels with the unit open and operating would probably cause failure of the hinged Top Cover Assembly at the hinges. The MRK assembly will be mounted in foam in an Orbiter storage area for transport to the ISS; therefore, it will not see the high vibration levels of an attached payload. In its intended operational use, the MRK fan is connected to an ISS Node seat track by an 18–inch flexible bracket hanging free from the wall seat track in space. In addition, a load analysis was performed and found that a standard bump would not cause any performance concerns. Thus, the need to perform Random Vibration testing as specified is alleviated.

The manufacturer has sold approximately 50,000 units with less than 1 percent returned. Finally, a post–modification PDA is performed and a PIA just prior to manifest for a flight.

GFE–35:

ITEM:

Moisture Removal Kit (MRK) Part Number SEG11100311

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.

Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Random Vibration test will not be performed on the Moisture Removal Kit.

RATIONALE:

A visual inspection of all soldering joints and added modifications to the internal MRK fan to verify no loose parts or connections will be performed on the MRK assembly.

An Acceptance Random Vibration test at the specified levels with the unit open and operating would probably cause failure of the hinged Top Cover Assembly at the hinges. The MRK assembly will be mounted in foam in an Orbiter storage area for transport to the ISS; therefore, it will not see the high vibration levels of an attached payload. In its intended operational use, the MRK fan is connected to an ISS Node seat track by an 18–inch flexible bracket hanging free from the wall seat track in space. In addition, a load analysis was performed and found that a standard bump would not cause any performance concerns. Thus, the need to perform Random Vibration testing as specified is alleviated.

The manufacturer has sold approximately 50,000 units with less than 1 percent returned. Finally, a post–modification PDA is performed and a PIA just prior to manifest for a flight.

GFE–36:

ITEM:

Moisture Removal Kit (MRK) Part Number SEG11100311

SSP 41172 REQUIREMENT:

Paragraph 5.1.8, Burn In Test, Component Acceptance.

Paragraph 5.1.8.3, Test Levels and Duration. ALL REQUIREMENTS.

Paragraph 5.1.8.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Burn–In test will not be performed on the Moisture Removal Kit.

RATIONALE:

The manufacturers stated life of the MRK is 300 hours. Acceptance Burn–In testing as specified would consume stated life.

The manufacturer has sold approximately 50,000 units with less than 1 percent returned.

The total MRK fan life operation is one use of twenty–four hours.

Each fan is pre–modification tested to confirm adequate flow rate for use after modification. A post–modification PDA is performed and a PIA just prior to manifest for a flight. This usually results in 1–2 hours of operation.

Finally, one qualification flight modified Class I fan was operated for 100 hours on four separate sets of batteries of 25 hours each. Three unmodified MRK fans have been operated in excess of 80 hours and one have been operated in excess of 150 hours (estimated) for personal cooling.

GFE–37:

ITEM:

Manual Pressure Equalization Valve (MPEV) Internal Sampling Adapter (ISA) Part Number 97M55830–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle. The minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible. See Figure 4–1. For electrical/electronic equipment where the minimum sweep (paragraph 5.1.3.3–2) does not encompass by the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F.

EXCEPTION:

The Qualification Thermal Cycle Test for the Internal Sampling Adapter will be performed over a temperature range of 32 degrees F to 131 degrees F (99 degrees F delta).

RATIONALE:

The purpose of qualification thermal cycle testing is to qualify the hardware design for the acceptance thermal cycle test. Typically, this requires imposing a 20–degree F temperature margin above and below the acceptance test minimum and maximum temperature extremes. For the ISA, imposing a 20–degree F qualification margin to the acceptance thermal cycle test minimum and maximum extremes would expose the unit to potentially damaging temperatures. Therefore, a 10–degree F margin will be applied for qualification thermal cycle testing. Test temperature tolerances will be controlled to 2.5 degrees F for both the qualification and acceptance thermal cycle tests so that the flight units are not exposed to qualification thermal environments.

The rate of temperature transition shall be 8 degrees F per minute above 70 degrees F, and 3 degrees F per minute below 70 degrees F. These transition rates exceed the minimum of 1–degree F per minute as specified by SSP 41172. Therefore, during the Qualification Thermal Cycle Test of the ISA, the 140 degrees F sweep is being replaced by a higher temperature transition rate of change. This will result in an acceptable qualification thermal cycle test for the ISA and reduce any risk associated with not achieving full qualification margin.

GFE–38:

ITEM:

Manual Pressure Equalization Valve (MPEV) Internal Sampling Adapter (ISA) Part Number 97M55830–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification

Paragraph 4.2.3.4, Supplementary Requirements. Functional tests shall be conducted, as a minimum, at the maximum predicted temperature plus 20 degrees F (11.1 degrees C) and at the minimum predicted temperature minus 20 degrees F (11.1 degrees C) during the first and last thermal cycles and after return of the component to ambient.

EXCEPTION:

Functional tests during the Qualification Thermal Cycle Test for the Internal Sampling Adapter shall be conducted at ambient temperature during the first and last thermal cycle.

RATIONALE:

Functional tests of the component are required at the minimum and maximum operational acceptance temperatures plus margin. For the ISA, the output is temperature sensitive and will not pass a functional test at these temperatures. Functional tests will only be performed at ambient temperature during the transition between the minimum and maximum temperature extremes during the first and last thermal cycles. The ISA will be powered and monitored throughout the full operational temperature range as required. This will be sufficient for detecting workmanship defects. This will result in an acceptable qualification thermal cycle test for detecting workmanship defects of the ISA

GFE–39:

ITEM:

Manual Pressure Equalization Valve (MPEV) Internal Sampling Adapter (ISA) Part Number 97M55830–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance

Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle. For components identified in Table 5–1, with note 4, there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The Acceptance Thermal Cycle Test for the Internal Sampling Adapter will be performed over an operational temperature range of 42 degrees F to 121 degrees F (79 degrees F delta).

RATIONALE:

The purpose of acceptance thermal cycle testing is to environmentally screen electrical and electronic components for latent workmanship defects. The critical parameters influencing the effectiveness of a thermal cycle stress screen are temperature range, number of temperature cycles, and rate of temperature change during hot/cold transitions. Of these, the rate of temperature change is generally considered to be the more effective screen with higher rates of change being most effective. In addition, the component is required to be powered up and monitored during the test (except for specified power off periods) to detect intermittence. SSP 41172 sets minimum thermal cycle screening parameters as 8 cycles of at least 100 degrees F temperature range with transitions at no less than 1 degree F per minute.

The ISA cannot be assured of operating without damage outside a temperature range of 32 degrees F to 131 degrees F. As it is not the purpose of environmental acceptance tests to damage otherwise acceptable hardware, the Acceptance Thermal Cycle Test for the ISA will be performed over an operational temperature range of 42 degrees F to 121 degrees F (79 degrees F delta). The rate of temperature transition shall be 8 degrees F per minute above 70 degrees F, and 3 degrees F per minute below 70 degrees F. These transition rates exceed the minimum of 1–degree F per minute as specified by SSP 41172. Therefore, during the Acceptance Thermal Cycle Test of the ISA, the 100 degrees F sweep is being replaced by a higher temperature transition rate of change. This will result in an acceptable acceptance thermal cycle test for the ISA.

GFE–40:

ITEM:

Manual Pressure Equalization Valve (MPEV) Internal Sampling Adapter (ISA) Part Number 97M55830–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance

Paragraph 5.1.3.4, Supplementary Requirements. Functional tests shall be conducted during the first and last operating thermal cycles after the dwell at the maximum and minimum predicted operating temperatures and after return of the component to ambient.

EXCEPTION:

Functional tests during the Acceptance Thermal Cycle Test for the Internal Sampling Adapter shall be conducted at ambient temperature during the first and last thermal cycle.

RATIONALE:

Functional tests of the component are required at the minimum and maximum operational acceptance temperatures. For the ISA, the output is temperature sensitive and will not pass a functional test at the operational maximum and minimum acceptance temperatures. Functional tests will only be performed at ambient temperature during the transition between the minimum and maximum temperature extremes during the first and last thermal cycles. The ISA will be powered and monitored throughout the full operational temperature range as required. This will be sufficient for detecting workmanship defects. This will result in an acceptable acceptance thermal cycle test for detecting workmanship defects of the ISA.

GFE–41:

ITEM:

Portable Computer System (PCS) Laptop Assembly as follows:
Computer (760XD) Part Number SDZ39129262–301
PGSC Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301
Floppy Drive Case Part Number SDZ39131205–301
Ethernet Card Part Number SDZ39129269–301
Flash Memory Card Part Number SDZ39131200–301
RS–422 Card Part Number SDZ39129284–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification

Paragraph 4.2.2.5, Depress/Repress Vacuum Requirements. ALL REQUIREMENTS.

EXCEPTION:

The PCS Laptop Assembly will perform operational qualification depress/repress test in the range of 9.5 psi to 14.2 psi.

RATIONALE:

The PCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760XD, RS–422 Card, Ethernet Card, Flash Memory Card, External Floppy Drive, and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 96–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, “shake” testing, as well as functional burn–ins. The decision to perform a 96–hour burn in for PCS was based on the previous experience of the a 48–hour burn–in for the Shuttle PGSC project and the Early Portable Computer System project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle’s IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR from September 97 to June 98 and experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. The hardware used by the PCS is common across several systems (e.g. Station Support Computer, CHeCs, Express Rack, HRF, etc.). In the event of a PCS failure, a laptop could be taken from a less critical system and used as a replacement for the PCS. On Flight 4A, there are currently six laptops manifested. Only two are required for the PCS function. The number of laptops on–board increases as payload users are added. For example at Flight 7A, fourteen laptops are manifested and only three are required for PCS operations.

In addition, it is known from both operational and evaluation testing that the COTS hardware will be damaged when depressed to less than 1 psi. Further, there are no PCS requirements for the hardware to operate beyond the range of 10.1 to 14.7 psi.

GFE–42:

ITEM:

Portable Computer System (PCS) Laptop Assembly as follows:
Computer (760XD) Part Number SDZ39129262–301
PGSC Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301
Floppy Drive Case Part Number SDZ39131205–301
Ethernet Card Part Number SDZ39129269–301
Flash Memory Card Part Number SDZ39131200–301
RS–422 Card Part Number SDZ39129284–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification

Paragraph 4.2.3.3, Thermal Cycling Test, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The PCS Laptop Assembly will perform an operational test of a single cycle to manufacturer's specifications of 50 degrees F to 95 degrees F (10 degrees C to 35 degrees C).

RATIONALE:

The PCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760XD, RS–422 Card, Ethernet Card, Flash Memory Card, External Floppy Drive, and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 96–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, "shake" testing, as well as functional burn–ins. The decision to perform a 96–hour burn in for PCS was based on the previous experience of the a 48–hour burn–in for the Shuttle PGSC project and the Early Portable Computer System project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle's IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR from September 97 to June 98 and experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. The hardware used by the PCS is common across several systems (e.g. Station Support Computer, CHeCs, Express Rack, HRF, etc.). In the event of a PCS failure, a laptop could be taken from a less critical system and used as a replacement for the PCS. On Flight 4A, there are currently six laptops manifested. Only two are required for the PCS function. The number of laptops on–board increases as payload users are added. For example at Flight 7A, fourteen laptops are manifested and only three are required for PCS operations.

In addition, testing beyond manufacturer's specifications will damage this COTS equipment. Further, there are no PCS requirements for the hardware to operate beyond manufacturer's specifications.

GFE–43:

ITEM:

Portable Computer System (PCS) Laptop Assembly as follows:
Computer (760XD) Part Number SDZ39129262–301
PGSC Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301
Floppy Drive Case Part Number SDZ39131205–301
Ethernet Card Part Number SDZ39129269–301
Flash Memory Card Part Number SDZ39131200–301
RS–422 Card Part Number SDZ39129284–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification

(1) Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Qualification Random Vibration tests will not be performed on the PCS Laptop Assembly.

RATIONALE:

The PCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760XD, RS–422 Card, Ethernet Card, Flash Memory Card, External Floppy Drive, and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 96–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, “shake” testing, as well as functional burn–ins. The decision to perform a 96–hour burn in for PCS was based on the previous experience of the a 48–hour burn–in for the Shuttle PGSC project and the Early Portable Computer System project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle’s IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR from September 97 to June 98 and experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. The hardware used by the PCS is common across several systems (e.g. Station Support Computer, CHeCs, Express Rack, HRF, etc.). In the event of a PCS failure, a laptop could be taken from a less critical system and used as a replacement for the PCS. On Flight 4A, there are currently six laptops manifested. Only two are required for the PCS function. The number of laptops on–board increases as payload users are added. For example at Flight 7A, fourteen laptops are manifested and only three are required for PCS operations.

In addition, the COTS equipment (hard drive, floppy drive, and CDROM) will be damaged if operating during vibration testing. Further, the hardware is not exposed to significant on–orbit vibration (it is not mounted). Operational flight experience has shown that the hardware (not operating) is capable of withstanding launch and landing vibration environments (through 4 EPCS Shuttle flights and over 150 similarly–launched models (PGSCs)).

GFE–44:

ITEM:

Portable Computer System (PCS) Laptop Assembly as follows:
Computer (760XD) Part Number SDZ39129262–301
PGSC Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301
Floppy Drive Case Part Number SDZ39131205–301
Ethernet Card Part Number SDZ39129269–301
Flash Memory Card Part Number SDZ39131200–301
RS–422 Card Part Number SDZ39129284–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance

(1) Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Thermal Cycling tests will not be performed on the PCS Laptop Assembly.

RATIONALE:

The PCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760XD, RS–422 Card, Ethernet Card, Flash Memory Card, External Floppy Drive, and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 96–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, “shake” testing, as well as functional burn–ins. The decision to perform a 96–hour burn in for PCS was based on the previous experience of the a 48–hour burn–in for the Shuttle PGSC project and the Early Portable Computer System project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle’s IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR from September 97 to June 98 and experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. The hardware used by the PCS is common across several systems (e.g. Station Support Computer, CHeCs, Express Rack, HRF, etc.). In the event of a PCS failure, a laptop could be taken from a less critical system and used as a replacement for the PCS. On Flight 4A, there are currently six laptops manifested. Only two are required for the PCS function. The number of laptops on–board increases as payload users are added. For example at Flight 7A, fourteen laptops are manifested and only three are required for PCS operations.

In addition, testing beyond manufacturer’s specifications will damage this COTS equipment. Further, there are no PCS requirements for the hardware to operate beyond manufacturer’s specifications.

GFE–45:

ITEM:

Portable Computer System (PCS) Laptop Assembly as follows:
Computer (760XD) Part Number SDZ39129262–301
PGSC Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301
Floppy Drive Case Part Number SDZ39131205–301
Ethernet Card Part Number SDZ39129269–301
Flash Memory Card Part Number SDZ39131200–301
RS–422 Card Part Number SDZ39129284–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance

(1) Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Random Vibration tests will not be performed on the PCS Laptop Assembly.

RATIONALE:

The PCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760XD, RS–422 Card, Ethernet Card, Flash Memory Card, External Floppy Drive, and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 96–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, “shake” testing, as well as functional burn–ins. The decision to perform a 96–hour burn in for PCS was based on the previous experience of the a 48–hour burn–in for the Shuttle PGSC project and the Early Portable Computer System project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle’s IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR from September 97 to June 98 and experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. The hardware used by the PCS is common across several systems (e.g. Station Support Computer, CHeCs, Express Rack, HRF, etc.). In the event of a PCS failure, a laptop could be taken from a less critical system and used as a replacement for the PCS. On Flight 4A, there are currently six laptops manifested. Only two are required for the PCS function. The number of laptops on–board increases as payload users are added. For example at Flight 7A, fourteen laptops are manifested and only three are required for PCS operations.

In addition, the COTS equipment (hard drive, floppy drive, and CDROM) will be damaged if operating during vibration testing. Further, the hardware is not exposed to significant on–orbit vibration (it is not mounted). Operational flight experience has shown that the hardware (not operating) is capable of withstanding launch and landing vibration environments (through 4 EPCS Shuttle flights and over 150 similarly–launched models (PGSCs)). Finally, the JSC manufacturing and quality control processes has a proven track record of manufacturing flight units without a single in–flight failure that could be attributed to a defect that vibration testing would have detected.

GFE–46:

ITEM:

Portable Computer System (PCS) Laptop Assembly as follows:
Computer (760XD) Part Number SDZ39129262–301
PGSC Power Supply Part Number SED39126010–301
1553 Card Part Number SDG39129273–301
Floppy Drive Part Number SEG39129288–301
Floppy Drive Case Part Number SDZ39131205–301
Ethernet Card Part Number SDZ39129269–301
Flash Memory Card Part Number SDZ39131200–301
RS–422 Card Part Number SDZ39129284–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.8, Burn–In Test, Component Acceptance

Paragraph 5.1.8.3C, Test Levels and Duration. For constant temperature burn–in (either at ambient or accelerated via elevated temperature), the total operating time shall be equivalent to 300 hours at ambient temperature.

EXCEPTION:

The Acceptance Burn–In test on the PCS Laptop Assembly will be 96 hours.

RATIONALE:

The PCS hardware is comprised of modified commercial hardware (IBM ThinkPad 760XD, RS–422 Card, Ethernet Card, Flash Memory Card, External Floppy Drive, and 1553 card) and existing shuttle designed hardware (28V dc power supply). In addition to the 96–hour GFE acceptance burn–in, the Commercial–Off–The–Shelf (COTS) manufacturer performs component–level testing, “shake” testing, as well as functional burn–ins. The decision to perform a 96–hour burn in for PCS was based on the previous experience of the a 48–hour burn–in for the Shuttle PGSC project and the Early Portable Computer System project which also utilizes IBM ThinkPads, commercial PCMCIA cards, and the 28 V dc power supply.

The shuttle 28 V dc power supply has operated for 9400 hours with no in flight failures. Twenty 28 V dc power supplies have been built and flown on the shuttle. The shuttle's IBM ThinkPad 755C has operated 6000 hours with no in flight failures and the IBM ThinkPad 750 operated 60,000 hours with only two recorded in flight failures (clock chip and a system board failure). A total of fourteen IBM ThinkPad 750s were modified for space flight and ten of these were flown. A total of forty nine IBM ThinkPad 755Cs were modified for space flight and forty five have flown to date. The EPCS (IBM ThinkPad 760ED and 28VDC power supply) has been operating on the MIR from September 97 to June 98 and experienced no failures. Fourteen IBM ThinkPad 760EDs have been modified for space flight and four have flown to date. The hardware used by the PCS is common across several systems (e.g. Station Support Computer, CHeCs, Express Rack, HRF, etc.). In the event of a PCS failure, a laptop could be taken from a less critical system and used as a replacement for the PCS. On Flight 4A, there are currently six laptops manifested. Only two are required for the PCS function. The number of laptops on–board increases as payload users are added. For example at Flight 7A, fourteen laptops are manifested and only three are required for PCS operations.

In addition to the COTS manufacturer burn–in tests and the 96–hour acceptance burn–in test, JSC performs additional functional tests upon receipt of the units, both prior to and after manufacturing into flight units. Finally, the JSC manufacturing and quality control processes has a proven track record of manufacturing flight units without a single in–flight failure that could be attributed to a defect that extended burn–in testing would have detected.

GFE–47:

ITEM:

Volatile Organic Analyzer (VOA) Part Number 0611–0001, Serial Numbers 001 and 002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests: When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification). This requires that the protoflight hardware be electrically energized and monitored during protoflight random vibration tests.

EXCEPTION:

Protoflight Random Vibration Tests without Power–On and monitoring is permitted for the Volatile Organic Analyzer.

RATIONALE:

The VOA requires 4 hours to complete a full functional test. During Protoflight Random Vibration Testing, abbreviated functional tests will be performed between each axis. An abbreviated functional test that exercises all electromechanical components requires 3.5 minutes to complete. This functional consists of the software pulsing electrical components; therefore, an intermittent failure would likely be impossible to detect during the Protoflight Random Vibration Test.

Information available via the VOA display is limited to the mode of operation. Engineering data required to determine an electrical intermittence is not accessible without violating the hardware configuration.

GFE–48:

ITEM:

Volatile Organic Analyzer (VOA) Part Number 0611–0001, Serial Numbers 001 and 002

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.

Paragraph 6.1.1.B, Thermal Cycling Test. The minimum number of cycles shall be eight.

Paragraph 6.1.1, Assembly/Components Protoflight Tests: When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification). This requires that the protoflight hardware be electrically energized and monitored during the minimum and maximum temperature dwells during the protoflight thermal cycling tests as indicated in paragraph 4.2.3.2.

EXCEPTION:

The Volatile Organic Analyzer will remain unpowered during the minimum and maximum temperature dwells of the Protoflight Thermal Cycling Test.

RATIONALE:

A Protoflight Thermal Cycle test of 8 cycles from 30 degrees F to 130 degrees F will be performed on the VOA. All cycles will include a full 100 degrees F temperature sweep at a rate not less than 1 degrees F per minute.

The VOA operates with active cooling provided in the range of 63 degrees F to 69 degrees F. During ground testing, active cooling is accomplished at ambient temperatures with dual Ground Support Equipment fans. Operating the VOA at ambient temperatures above 85 degrees F will blow thermal fuses protecting the secondary power supplies.

The VOA contains 14 heaters operating continuously in the range from 120 degrees F to 620 degrees F. Functional testing cannot be initiated below 60 degrees F since the heaters will not be capable of reaching their specified limit to complete the test.

Abbreviated functional testing will be performed during each cycle of the test at 60 degrees F and 85 degrees F which represents the operational limits of the VOA without active cooling. A full functional test will be performed prior to the first cycle and following the final cycle at ambient temperature.

GFE–49:

ITEM:

Circuit Isolation Device (CID) Part Number 54059M90A

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification

Paragraph 4.2.2.3C, Test Levels and Duration. A minimum of three temperature cycles shall be used.

EXCEPTION:

The duration of the Qualification Thermal Vacuum Test for the Circuit Isolation Device shall be one cycle.

RATIONALE:

The CIDs consist of a robust Military Quality switch that has no electronics within the switch. Flight connectors and wire are furnished from the Space Station program and have already undergone testing. Only one resistor is in the entire design and it acts as a bleed resistor to dissipate static charges. The switch has undergone a vigorous testing program to meet Military requirements. Additionally, the switch have been further tested at Glenn Research Center beyond its operational environment in destructive and repetitive thermal vacuum testing of at least 100 cycles prior to developing a qualification unit.

GFE–50:

ITEM:

Circuit Isolation Device (CID) Part Number 54059M90A

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification

Paragraph 4.2.3.3C, Test Levels and Duration. The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total.

EXCEPTION:

The duration of the Qualification Thermal Cycle Test for the Circuit Isolation Device shall be six cycles. The first cycle will be performed as part of the qualification thermal vacuum test.

RATIONALE:

The CIDs consist of a robust Military Quality switch that has no electronics within the switch. Flight connectors and wire are furnished from the Space Station program and have already undergone testing. Only one resistor is in the entire design and it acts as a bleed resistor to dissipate static charges. The switch has undergone a vigorous testing program to meet Military requirements. Additionally, the switch have been further tested at Glenn Research Center beyond its operational environment in destructive and repetitive thermal vacuum testing of at least 100 cycles prior to developing a qualification unit.

GFE–51:

ITEM:

Circuit Isolation Device (CID) Part Number 54059M90A

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance

Paragraph 5.1.3.3C, Test Levels and Duration. The minimum number of temperature cycles shall be eight.

EXCEPTION:

The duration of the Acceptance Thermal Cycle Test for the Circuit Isolation Device shall be two cycles. The first cycle will be performed as part of the acceptance thermal vacuum test.

RATIONALE:

The CIDs consist of a robust Military Quality switch that has no electronics within the switch. Flight connectors and wire are furnished from the Space Station program and have already undergone testing. Only one resistor is in the entire design and it acts as a bleed resistor to dissipate static charges. The switch has undergone a vigorous testing program to meet Military requirements. Additionally, the switch have been further tested at Glenn Research Center beyond its operational environment in destructive and repetitive thermal vacuum testing of at least 100 cycles prior to developing a qualification unit.

GFE–52:

ITEM:

Manual Pressure Equalization Valve (MPEV) Internal Sampling Adapter (ISA) Part Number 97M55830–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification

Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

Qualification Random Vibration Tests without Power–On and monitoring is permitted for the Manual Pressure Equalization Valve Internal Sampling Adapter.

RATIONALE:

The ISA is launched packed in foam to protect it against potentially detrimental launch vibration loads. The ISA is not operated during these launch environments.

The ISA was subjected to a qualification random vibration test in its launch configuration at a level of 8.6 grms. The internal electronic pressure module component was not powered or monitored during this test. However, functionality of the pressure module was verified following qualification random vibration testing.

The manufacturer of the internal electronic pressure module has reported less than a one–percent failure rate.

The ISA is noncritical ISS hardware. There are both operational workarounds available in the event of ISA failure as well as spare pressure modules for replacement on orbit. The ISA will be periodically checked for health on–orbit and will be returned for yearly calibration.

GFE–53:

ITEM:

Manual Pressure Equalization Valve (MPEV) Internal Sampling Adapter (ISA) Part Number 97M55830–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.10, Pressure Test, Component Qualification

Paragraph 4.2.10.3, Test Levels. ALL REQUIREMENTS.

Paragraph 4.2.10.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A Negative Qualification Proof Pressure Test will not be performed on the Manual Pressure Equalization Valve Internal Sampling Adapter.

RATIONALE:

The ISA external pressure will be ISS cabin pressure (nominally, 14.2 to 14.9 psia). The Variable Relief Valve will prevent cabin pressure from exceeding 15.2 psia under failure conditions. The in–use ISA internal pressure will be the pressure of the volume being sampled. In the case of vestibule depressurization, the ISA internal pressure will reach vacuum. Therefore, the maximum negative delta pressure, external to internal, of the ISA will be 14.9 psia under nominal conditions and 15.2 psia under failure conditions (if an overpressure anomaly occurs during the vestibule venting operation).

The ISA is composed of stainless steel fittings, a stainless steel valve, and the internal pressure module. These are mounted on a stainless steel block. The stainless steel parts are rated for higher pressures than the ISA will see and are not susceptible to damage from either positive or negative delta pressures imposed by normal ISA operation. The only ISA component that could be damaged by exposure to delta pressure is the internal pressure module. The pressure module is contained in a plastic housing which is open to ambient pressure; thus, the only component that is actually exposed to delta pressure is the pressure sensor itself. The sensor range is 0 to 30 psia; therefore, no damage to the diaphragm should occur at ISA delta pressures.

According to the pressure module vendor, Crystal Engineering Corporation, the only potential susceptible area is an o–ring seal between the chamber that is referenced to the measured pressure and the housing volume that is open to ambient pressure. The o–ring groove is a symmetrical cross–section; therefore, the effect of negative delta pressure on the seal should be the same as the effect of positive delta pressure. A positive proof pressure test was performed at 22.7 psid. The leak rate following the positive proof test was 2.6 x 10E–06 sccs. Thus, the successful positive proof pressure test answers concerns about exposure to negative proof pressure as well.

In addition, a Safety Assessment was performed for the ISA and approved by the NASA Safety Review Panel. Structural failure was assessed as being a noncredible hazard for this hardware. As no credible safety hazard exists, the hardware is noncritical, and operational workarounds are available, the risk associated with not conducting the negative proof pressure test is negligible.

GFE–54:

ITEM:

Manual Pressure Equalization Valve (MPEV) Internal Sampling Adapter (ISA) Part Number 97M55830–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.

Paragraph 5.1.4.3, Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:

B. A workmanship screening level and spectrum shown in Figure 5–2 or to screening levels and spectrums approved by the Prime contractor.

EXCEPTION:

The Acceptance Random Vibration Tests on the Manual Pressure Equalization Valve Internal Sampling Adapter in its launch configuration will be performed at an overall screening level of 4.3 grms.

RATIONALE:

The ISA is launched packed in foam to protect it against potentially detrimental launch vibration loads. The ISA is not operated during these launch environments.

Subjecting an unprotected ISA to the test environment specified by SSP 41172 could result in a failure that the foam packing is designed to prevent. Also, there are not any physical attach points on the ISA for hard–mounting the unit to a vibration test fixture. An additional fixture/adapter would need to be fabricated in order to perform the test in this manner.

Additional workmanship screening is provided during the Acceptance Thermal Cycle Test. The test for the ISA is performed over an operational temperature range of 42 degrees F to 121 degrees F. The rate of temperature transition is 8 degrees F per minute above 70 degrees F, and 3 degrees F per minute below 70 degrees F. These transition rates exceed the minimum of 1–degree F per minute as specified by SSP 41172.

The manufacturer of the internal electronic pressure module has reported less than a one–percent failure rate.

The ISA is noncritical ISS hardware. There are both operational workarounds available in the event of ISA failure as well as spare pressure modules for replacement on orbit. The ISA will be periodically checked for health on–orbit and will be returned for yearly calibration.

GFE–55:

ITEM:

Manual Pressure Equalization Valve (MPEV) Internal Sampling Adapter (ISA) Part Number 97M55830–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance

Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be electrically energized and monitored during the test.

EXCEPTION:

Acceptance Random Vibration Tests without Power–On and monitoring is permitted for the Manual Pressure Equalization Valve Internal Sampling Adapter.

RATIONALE:

The ISA is launched packed in foam to protect it against potentially detrimental launch vibration loads. The ISA is not operated during these launch environments.

The ISA is subjected to an acceptance random vibration test in its launch configuration at a level of 4.3 grms. The internal electronic pressure module component is not powered or monitored during this test. However, functionality of the pressure module is verified following acceptance random vibration testing.

Additional workmanship screening is provided during the Acceptance Thermal Cycle Test. The test for the ISA is performed over an operational temperature range of 42 degrees F to 121 degrees F. The rate of temperature transition is 8 degrees F per minute above 70 degrees F, and 3 degrees F per minute below 70 degrees F. These transition rates exceed the minimum of 1–degree F per minute as specified by SSP 41172.

The manufacturer of the internal electronic pressure module has reported less than a one–percent failure rate.

The ISA is noncritical ISS hardware. There are both operational workarounds available in the event of ISA failure as well as spare pressure modules for replacement on orbit. The ISA will be periodically checked for health on–orbit and will be returned for yearly calibration.

GFE–56:

ITEM:

Battery Charger Subassembly Part Number SEG39128256–303, Serial Number 1006

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The Qualification Thermal Cycle Test for the Airlock SPCE BCA shall consist of 24 thermal cycles from 40 degrees F to 150 degrees F and one thermal cycle from 30 degrees F to 150 degrees F.

RATIONALE:

The Acceptance Thermal Cycle test temperature range of the Airlock SPCE BCA will be from 40 degrees F to 140 degrees F. SSCN 1348 initially reduced the qualification margin required to 10 degrees F beyond the maximum and minimum acceptance temperature extremes from 30 degrees F to 150 degrees F. The as–run test will provide the 10–degree F margin during one cold thermal cycle and 25 hot thermal cycles.

Components within the battery charger subassembly are commercial parts whose temperature range cannot withstand the minimum required qualification temperature sweep of 140 degrees F without potential damage to the component. Due to unexplained anomalies experienced on the battery charger subassembly qualification unit (Serial Number 1002), repeated intrusions into the unit for repair that have led to the replacement of damaged components, and the accumulated test stress that has been logged on the battery charger subassembly qualification unit, qualification testing will be performed on the backup battery charger subassembly flight unit (Serial Number 1006) at the temperatures indicated.

GFE–57:

ITEM:

Battery Charger Subassembly Part Number SEG39128256–303, Serial Number 1006

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification

Paragraph 4.2.5.3, Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:

B. Acceptances test levels and spectrum plus test tolerances.

EXCEPTION:

The Qualification Random Vibration Test for the Battery Charger Subassembly will be performed at a composite screening level of 6.1 grms for a duration of one minute per axis.

RATIONALE:

The Qualification Random Vibration Test for the Battery Charger Subassembly will be performed at the same levels as during the Acceptance Random Vibration Test.

Due to unexplained anomalies experienced on the battery charger subassembly qualification unit (Serial Number 1002), repeated intrusions into the unit for repair that have led to the replacement of damaged components, and the accumulated test stress that has been logged on the battery charger subassembly qualification unit, qualification testing will be performed on the backup battery charger subassembly flight unit (Serial Number 1006) at the levels indicated. This is needed to preserve the flight status of this backup unit. These levels do encompass predicted maximum flight loads (4.3 grms composite) plus margin.

GFE–58:

ITEM:

Battery Stowage Assembly Part Number SEG39128213–301, Serial Number 1001

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification

Paragraph 4.2.5.3, Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:

B. Acceptances test levels and spectrum plus test tolerances.

EXCEPTION:

The Qualification Random Vibration Test for the Battery Stowage Assembly will be performed at a composite screening level of 6.1 grms for a duration of three minutes per axis.

RATIONALE:

The Qualification Random Vibration Test for the Battery Stowage Assembly will be performed at the same levels but for a duration three times that during the Acceptance Random Vibration Test.

The Battery Stowage Assembly contains few electronic components. These do experience a full Qualification Thermal Cycle Test of 24 thermal cycles with a 140 degrees F temperature sweep as specified in SSP 41172. As the qualification random vibration levels do encompass maximum flight loads (4.3 grms composite) plus margin, the additional risk associated with testing at the reduced level is minimal.

GFE–59:

ITEM:

Extravehicular Mobility Unit Don/Doff Assembly (EDDA) Part Number SEG39128210–301, Serial Number 1001

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification

Paragraph 4.2.5.3, Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:

B. Acceptances test levels and spectrum plus test tolerances.

EXCEPTION:

The Qualification Random Vibration Test for the Extravehicular Mobility Unit Don/Doff Assembly will be performed at a composite screening level of 6.1 grms for a duration of three minutes per axis.

RATIONALE:

The Qualification Random Vibration Test for the EDDA will be performed at the same levels but for a duration three times that during the Acceptance Random Vibration Test.

The EDDA contains no electronic components. The EDDA was utilized during a series of 1–g and 0–g donning loads evaluations (reference JSC 39238, dated January 25, 1999) with no anomalies reported. As the qualification random vibration levels do encompass maximum flight loads (4.3 grms composite) plus margin, the additional risk associated with testing at the reduced level is minimal.

GFE–60:

ITEM:

Control Electronics Unit (CEU) Part Number 829834–551, Serial Number 00003

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle. The minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible. See Figure 4–1. For electrical/electronic equipment where the minimum sweep (para. 5.1.3.3–2) does not encompass by the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F.

EXCEPTION:

The Qualification Thermal Cycle Test for the Control Electronics Unit shall be performed operating from 13 degrees F to 110 degrees F.

RATIONALE:

A 20–degree F qualification margin is provided on the minimum and maximum operating acceptance thermal cycle temperatures. However, this does not provide the minimum sweep of 140 degrees F as specified in SSP 41172.

The ramp rate for thermal testing of the CEU was 3 degrees F per minute. The higher ramp rate does provide a more strenuous test condition.

The Qualification CEU completed 32 thermal cycles (eight cycles were at acceptance test procedure levels).

The thermal system is single–fault tolerant which reduces the risk of the CEU experiencing worst–case conditions.

There are two CEUs in the on–orbit configuration. Each CEU contains three Video Graphics Cards that provides redundancy.

GFE–61:

ITEM:

Display and Control Panel Part Number 829872–551, Serial Number 00003

SSP 41172 REQUIREMENT:

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle. The minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible. See Figure 4–1. For electrical/electronic equipment where the minimum sweep (para. 5.1.3.3–2) does not encompass by the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F.

EXCEPTION:

The Qualification Thermal Cycle Test for the Display and Control Panel shall consist of 24 thermal cycles from 5 degrees F to 105 degrees F. The Display and Control Panel shall be operating from 45 degrees F to 105 degrees F.

RATIONALE:

The ramp rate for thermal testing of the Display and Control Panel was greater than 1 degree F per minute as specified by SSP 41172.

The Qualification Display and Control Panel completed 32 thermal cycles (eight cycles were at acceptance test procedure levels).

The thermal system is single–fault tolerant which reduces the risk of the Display and Control Panel experiencing worst–case conditions.

There are two Display and Control Panels in the on–orbit configuration.

GFE–62:

ITEM:

Control Electronics Unit (CEU) Part Number 829834–551, Serial Numbers 00001 and 00002

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance

Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle. For components identified in Table 5–1, with note 4, there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The Acceptance Thermal Cycle Test for the Control Electronics Unit shall consist of eight thermal cycles from –10 degrees F to 90 degrees F. The Control Electronics Unit shall be operating from 33 degrees F to 90 degrees F.

RATIONALE:

CEU has 3 temperature–limiting components on its internal Video Graphics Board:

A. D–Latch, Part Number 54AC574FMQB – Upper Temperature Limit of 113 degrees. F due to junction hot spot

B. NTSC Decoder, BT819AKPF – Lower Temperature Limit of 32 degrees F due to timing errors

C. Raytheon TMC2081 - Lower Temperature Limit of 32 degrees F due to timing errors

Flight CEUs completed Acceptance Thermal Cycle Tests both prior to Huntsville shipment in August 1997 and prior to KSC shipment in April 1999. These flight CEUs also completed additional workmanship screens including two Acceptance Random Vibration Tests.

The USL–specified coldplate temperature is within 33 degrees F to 90 degrees F, identical to the operating Acceptance Thermal Cycle Test environment.

The nominal operating environment is predicted to be from 60 degrees F to 80 degrees F and the Thermal System is single–fault tolerant.

There are two CEUs in the on–orbit configuration. Each CEU contains three Video Graphics Cards that provides redundancy.

GFE–63:

ITEM:

Display and Control Panel Part Number 829872–551, Serial Numbers 00001 and 00002

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance

Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle. For components identified in Table 5–1, with note 4, there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The Acceptance Thermal Cycle Test for the Display and Control Panel shall consist of eight thermal cycles from –15 degrees F to 85 degrees F. The Display and Control Panel shall be operating from –15 degrees F to 85 degrees F.

RATIONALE:

The USL–specified Display and Control Panel operating temperature range is predicted to be from 65 degrees F to 85 degrees F.

Flight Display and Control Panels completed Acceptance Thermal Cycle Tests both prior to Huntsville shipment in August 1997 and prior to KSC shipment in April 1999. These flight Display and Control Panels also completed additional workmanship screens including two Acceptance Random Vibration Tests. As integrated as part of the Robotic Workstation, these will undergo a projected 700 hours of burn–in prior to launch.

GFE–64:

ITEM:

Display and Control Panel Fan Part Number 830957–551

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification

Paragraph 4.2.5.3, Test Levels and Duration. The test duration in each of the three orthogonal axes shall be three times the expected flight exposure time to the maximum predicted level and spectrum or three times the component random vibration acceptance test time if that is greater, but not less than three minutes per axis.

EXCEPTION:

The Qualification Random Vibration Test for the Display and Control Panel Fan will be performed for a duration of one minute per axis.

RATIONALE:

The Display and Control Panel with Fans is launched packed in foam in the Mini–Pressurized Logistics Module. Robotic Workstation Testing indicates at least a 60 percent reduction in the random vibration environment experienced by the component when minimal foam thickness is used for packing.

Vendor data specifies 200,000 hours of operational reliability.

Display and Control Panel Fan is criticality three hardware.

GFE–65:

ITEM:

Display and Control Panel Fan Part Number 830957–551

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance

(1) Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.

(2) Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

Acceptance Random Vibration tests will not be performed on the Display and Control Panel Fan.

RATIONALE:

The Display and Control Panel with Fans is launched packed in foam in the Mini–Pressurized Logistics Module. Robotic Workstation Testing indicates at least a 60 percent reduction in the random vibration environment experienced by the component when minimal foam thickness is used for packing.

A Qualification Display and Control Panel Fan does pass a Qualification Random Vibration Test for a duration of one minute per axis.

Vendor data specifies 200,000 hours of operational reliability.

Display and Control Panel Fan is criticality three hardware.

GFE–66:

ITEM:

Video Monitor Part Number 829880–501

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification

Paragraph 4.2.5.3, Test Levels and Duration. The test duration in each of the three orthogonal axes shall be three times the expected flight exposure time to the maximum predicted level and spectrum or three times the component random vibration acceptance test time if that is greater, but not less than three minutes per axis.

EXCEPTION:

The Qualification Random Vibration Test for the Video Monitor will be performed for a duration of one minute per axis.

RATIONALE:

The Video Monitor is launched packed in foam in the Mini–Pressurized Logistics Module. Robotic Workstation testing indicates at least a 60 percent reduction in the random vibration environment experienced by the component when minimal foam thickness is used for packing.

Vendor data specifies 20,000 hours of operational reliability.

Video Monitor is a Military Qualified component.

The integrated Robotic Workstation contains three video monitors each.

GFE–67:

ITEM:

Robotic Workstation Part Number 829875–551

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification

Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle. The minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible. See Figure 4–1. For electrical/electronic equipment where the minimum sweep (para. 5.1.3.3–2) does not encompass by the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F.

EXCEPTION:

The Qualification Thermal Cycle Test for the Robotic Workstation shall not be performed for the nonoperating tempetature range from –25 degrees F to 150 degrees F.

RATIONALE:

The nonoperating temperature range is derived from ground transportation thermal environments. EMS thermal analysis indicates that the Robotic Workstation will meet the nonoperating temperature range specified.

Additionally, Robotic Workstation packaging instruction states specific transportation requirements to mitigated any risk. These include an air–ride van with humidity and environment controls and ORUs packed in foam.

GFE–68:

ITEM:

Robotic Workstation Part Number 829875–551

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Qualification

Paragraph 5.1.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits during the hot portion of the cycle and at the minimum acceptance limits during the cold portion of the cycle.

EXCEPTION:

The Acceptance Thermal Cycle Test for the Robotic Workstation shall not be performed for the nonoperating tempetature range from –25 degrees F to 150 degrees F.

RATIONALE:

The nonoperating temperature range is derived from ground transportation thermal environments. EMS thermal analysis indicates that the Robotic Workstation will meet the nonoperating temperature range specified.

Additionally, Robotic Workstation packaging instruction states specific transportation requirements to mitigated any risk. These include an air–ride van with humidity and environment controls and ORUs packed in foam.

GFE–69:

ITEM:

Centerline Berthing Camera System:
Camera and LED Assembly with Mounting Bracket Part Number SEG33112576–301

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.

Paragraph 6.1.1.B, Thermal Cycling Test. The minimum number of cycles shall be eight.

EXCEPTION:

The Protoflight Thermal Cycling Test on the Camera and LED Assembly with Mounting Bracket shall be three cycles.

RATIONALE:

The Centerline Berthing Camera System is comprised of Orbiter and DTO hardware including the LED Control Unit, Video Interface Unit, and Camera and LED Assembly with Mounting Bracket.

The Camera Assembly has an extensive flight history in payload bay environments including Shuttle missions STS–62, STS–64, STS–82, STS–95, and STS–85. Temperature ranges for these missions were more extreme than those in which the Centerline Berthing Camera System will experience. The Camera Assembly has flown approximately 25 shuttle missions without an anomaly or failure. Additionally, the Camera Assembly manufacturer has determined a Mean Time Between Failure of 38,400 hours. Thus, when viewed with its past flight history, three thermal cycles during the protoflight thermal cycling test are sufficient to screen for workmanship.

GFE–70:

ITEM:

Centerline Berthing Camera System:
Camera and LED Assembly with Mounting Bracket Part Number SEG33112576–301

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests: "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…." This requires that the protoflight hardware undergo electromagnetic compatibility testing as indicated in paragraph 4.2.12.

EXCEPTION:

The Camera and LED Assembly with Mounting Bracket shall be certified for radiated emissions and susceptibility only during the Protoflight Electromagnetic Compatibility Test. Testing for conducted emissions and susceptibility shall be omitted.

The Ku–band Power Supply shall not be included in the Protoflight Electromagnetic Compatibility Test.

RATIONALE:

The Centerline Berthing Camera System is comprised of Orbiter and DTO hardware including the LED Control Unit, Video Interface Unit, and Camera and LED Assembly with Mounting Bracket.

The Ku–band Power Supply has been certified for use on the ISS. The Ku–band Power Supply isolates the Centerline Berthing Camera System avionics for conducted emissions and susceptibility from the ISS power.

GFE–71:

ITEM:

Centerline Berthing Camera System:
Camera and LED Assembly with Mounting Bracket Part Number SEG33112576–301

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests: "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…." This requires that the protoflight hardware undergo depress/repress vacuum testing as indicated in paragraph 4.2.2.5.

EXCEPTION:

The Camera and LED Assembly with Mounting Bracket shall not undergo a Depress/Repress Vacuum Test.

RATIONALE:

The Centerline Berthing Camera System is comprised of Orbiter and DTO hardware including the LED Control Unit, Video Interface Unit, and Camera and LED Assembly with Mounting Bracket.

The Camera Assembly has flown approximately 25 shuttle missions without an anomaly or failure and has an extensive flight history in payload bay environments including Shuttle missions STS–62, STS–64, STS–82, STS–95, and STS–85. Thus, the basic element of the Centerline Berthing Camera System has been both previously tested in thermal vacuum and experienced actual flight vacuum environments.

The LED Control Unit will individually undergo its own depress/repress vacuum test.

When viewed with its past flight history, the omission of a depress/repress vacuum test on the Camera and LED Assembly with Mounting Bracket is permitted.

GFE–72:

ITEM:

Centerline Berthing Camera System:
LED Control Unit Part Number SEG33112643–301

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests: "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…." This requires that the protoflight hardware undergo depress/repress vacuum testing as indicated in paragraph 4.2.2.5 with the component powered as indicated in paragraph 4.2.2.5.1.

EXCEPTION:

The LED Control Unit will not be powered during its Depress/Repress Vacuum Test.

RATIONALE:

The LED Control Unit is designed with Military Specification parts which are certified for use. During both the Thermal Cycle and Burn–In testing performed, the LED Control Unit has demonstrated robustness to the thermal environment that would be induced by the reduced pressure.

GFE–73:

ITEM:

Centerline Berthing Camera System:
Camera and LED Assembly with Mounting Bracket Part Number SEG33112576–301

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests: "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…." This requires that the protoflight hardware undergo burn–in testing of 300 hours or equivalent as determined by the Time Acceleration Factor equation in paragraph 5.1.8.3.

EXCEPTION:

The fully–assembled Centerline Berthing Camera System will undergo approximately 28 hours of operating time during testing.

RATIONALE:

The Centerline Berthing Camera System is comprised of Orbiter and DTO hardware including the LED Control Unit, Video Interface Unit, and Camera and LED Assembly with Mounting Bracket.

The Camera Assembly has flown approximately 25 shuttle missions without an anomaly or failure and they have an extensive flight history in payload bay environments including Shuttle missions STS–62, STS–64, STS–82, STS–95, and STS–85. Additionally, the Camera Assembly manufacturer has determined a Mean Time Between Failure of 38,400 hours. Thus, when viewed with its past flight history, the performed operating time in lieu of compliance to the burn–in requirement is permitted.

GFE–74:

ITEM:

Centerline Berthing Camera System:
LED Control Unit Part Number SEG33112643–301

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests: "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…." This requires that the protoflight hardware undergo burn–in testing of 300 hours or equivalent as determined by the Time Acceleration Factor equation in paragraph 5.1.8.3.

EXCEPTION:

The LED Control Unit will undergo burn–in testing of 244.5 hours.

RATIONALE:

The LED Control Unit is designed with Military Specification parts which are certified for use. The LED Control Unit do undergo vibration, depress/repress vacuum, and thermal cycling testing. The thermal ramp rate during thermal cycle testing is approximately 3 degree F per minute.

This LED Control Unit design has been previously flown with no in–flight anomalies. Additionally, the build of LED Control Units for the ISS have been extensively tested including functional tests at board and subassembly levels prior to assembly. These functional tests were not included in the calculation of the LED Control Unit burn–in test time. Thus, when viewed with its past flight history, the reduction of burn–in testing is acceptable.

GFE–75:

ITEM:

Power and Video Grapple Fixure Part Number 51818–003

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3A, Test Levels and Duration. The pressure shall be reduced from atmospheric to below 0.0001 Torr (0.0133 Pa)

EXCEPTION:

Qualification thermal vacuum testing will be permitted at ambient pressure for the Power and Video Grapple Fixture.

RATIONALE:

Thermal cycle testing for functional performance was conducted on complete assembly (Qualification: 10 cycles, Acceptance: 2 cycles).

Power and Video Grapple Fixture high–use components (Grapple Shaft and Latching End Effector connectors) were previously qualified individually for Power and Data Grapple Fixture and are qualified by similarity (SPAR–SS–R.18999 and CR 8976–F–143).

The selection of coatings and processes eliminates the need for vacuum testing on low–use components (contingency use only). This applies to the Power and Video Grapple Fixture Grapple Shaft EVA release mechanism threaded components and sliding parts that are lubricated with a Lubeco 905 solid film lubricant. This lubricant has been widely used on Shuttle Remote Manipulator System and Space Station Remote Manipulator System mechanisms and is vacuum rated. With the light usage and low Hertzian stresses on each of the load–bearing components of the mechanism, the lubricant will not be compromised. As a further precaution against micro–welding and galling, the sliding surface of the spear has been Titanium nitrided and the stationary surface of the busing which interfaces with the release spear has been nitrided.

GFE–76:

ITEM:

Power and Video Grapple Fixure   Part Number 51818–003

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3A. Pressure. The pressure shall be reduced from atmospheric to below 0.0001 Torr (0.0133Pa).

EXCEPTION:

Acceptance thermal vacuum testing will be permitted at ambient pressure for the Power and Video Grapple Fixture.

RATIONALE:

Thermal cycle testing for functional performance was conducted on complete assembly (Qualification: 10 cycles, Acceptance: 2 cycles).

Power and Video Grapple Fixture high–use components (Grapple Shaft and Latching End Effector connectors) were previously qualified individually for Power and Data Grapple Fixture and are qualified by similarity (SPAR–SS–R.18999 and CR 8976–F–143).

The selection of coatings and processes eliminates the need for vacuum testing on low–use components (contingency use only). This applies to the Power and Video Grapple Fixture Grapple Shaft EVA release mechanism threaded components and sliding parts that are lubricated with a Lubeco 905 solid film lubricant. This lubricant has been widely used on Shuttle Remote Manipulator System and Space Station Remote Manipulator System mechanisms and is vacuum rated. With the light usage and low Hertzian stresses on each of the load–bearing components of the mechanism, the lubricant will not be compromised. As a further precaution against micro–welding and galling, the sliding surface of the spear has been Titanium nitrided and the stationary surface of the busing which interfaces with the release spear has been nitrided.

GFE–77:

ITEM:

Power and Video Grapple Fixure   Part Number 51818–003

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

Qualification random vibration testing without power–on and monitoring is permitted for the Power and Video Grapple Fixture.

RATIONALE:

The Power and Video Grapple Fixture is a simple mechanical interface between the Space Station Remote Manipulator System and payloads/pallets with simple electrical parts only: connectors, wires, and pins.

The Power and Video Grapple Fixture is launched with the connectors in an unmated condition. Securing the connectors as they are when mated to the Latching End Effector will support connectors during the vibration test and render the tests non–representative of the flight configuration. Providing only mating connectors with cable harness will add unsupported mass to Power and Video Grapple Fixture connectors. This will subject the compliance mechanisms of the Power and Video Grapple Fixture connectors to greater stresses than normal in expected launch configuration. This may cause damage not directly attributable to the flight article configuration.

GFE–78:

ITEM:

Power and Video Grapple Fixure Part Number 51818–003

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.4, Supplementary Requirements. Electrical and electronic components shall be powered on and monitored for failures or intermittences during the random vibration test.

EXCEPTION:

Acceptance random vibration testing without power on and monitoring is permitted for the Power and Video Grapple Fixture.

RATIONALE:

The Power and Video Grapple Fixture is a simple mechanical interface between the Space Station Remote Manipulator System and payloads/pallets with simple electrical parts only: connectors, wires, and pins.

The Power and Video Grapple Fixture is launched with the connectors in an unmated condition. Securing the connectors as they are when mated to the Latching End Effector will support connectors during the vibration test and render the tests non–representative of the flight configuration. Providing only mating connectors with cable harness will add unsupported mass to Power and Video Grapple Fixture connectors. This will subject the compliance mechanisms of the Power and Video Grapple Fixture connectors to greater stresses than normal in expected launch configuration. This may cause damage not directly attributable to the flight article configuration.

GFE–79:

ITEM:

Duct Smoke Detector Configuration Item M37070F Serial Number 038

SSP 41172 REQUIREMENT:

Paragraph 5.1.4 Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3 Test Levels and Duration. Component random vibration test levels and spectrums shall be the envelope of the following:
Paragraph 5.1.4.3A Test Levels and Duration. The maximum predicted flight level and spectrum minus 6 dB, but not less than a level derived from a 135 dB overall acoustic environment.

EXCEPTION:

The Acceptance Random Vibration Test of the Duct Smoke Detector Serial Number 038 was the minimum workmanship screening level of 6.1 grms for 120 seconds per axis in place of the maximum predicted flight level and spectrum minus 6 dB.

RATIONALE:

The Test and Verification Control Panel reviewed the comparison of the X–,Y–, and Z–axis spectra with the appropriate qualification and acceptance levels and agreed that the deltas were extremely small. The Test and Verification Control Panel was provided a picture of the Duct Smoke Detector and, based on its small size and weight, determined that the 6.1 grms spectrum and the 120 second duration were sufficient to adequately screen the Duct Smoke Detector.

GFE–80:

ITEM:

Plasma Contactor Unit Hollow Cathode Assembly Part Number 62416J

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.5.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Plasma Contactor Unit Hollow Cathode Assembly shall not undergo a qualification random vibration test at the component level.

RATIONALE:

The qualification unit of the Plasma Contactor Unit Hollow Cathode Assembly was subjected to the following testing during its design certification:

Component–level Testing
The qualification Hollow Cathode Assembly has undergone ignition testing. This test is performed under vacuum conditions and consists of 10 on/off/cool cycles. During this test, the temperature of the Hollow Cathode Assembly varied from ambient to 2192 degrees F (1200 degrees C). The Hollow Cathode Assembly was visually examined and performance data was reviewed following the ignition testing.

The qualification Hollow Cathode Assembly heater completed confidence testing. This testing consisted of 150 on/off operational cycles. The Hollow Cathode Assembly heaters were visually examined and performance data reviewed following the confidence testing.

High–fidelity developmental Hollow Cathode Assembly units identical to flight Hollow Cathode Assemblies are in life testing and ignition testing. These units have exceeded the 18,000 hours life requirement. One unit in ignition testing exceeded 42,000 ignitions, far surpassing the requirement of 6,000 ignitions with reliability greater than 99 percent.

Lower–fidelity developmental Hollow Cathode Assembly units with the same cathode and heater, but different wiring configuration and mounting flange, successfully completed vibration testing. The vibration testing consisted of sinusoidal vibration of 0.5 sine in 3 axes at 1 octave per minute and random vibration of 16.5 grms in 3 axes for 1 minute per axis.

ORU Level Testing
The qualification Hollow Cathode Assembly unit was vibration tested in the Plasma Contactor Unit in accordance with SSP 41172. Electrical functional testing and the clamp voltage test followed the vibration test. The Hollow Cathode Assembly operated as specified during the electrical functional test. There was an unexplained anomaly during the clamp voltage test as the Hollow Cathode Assembly failed to ignite after 12 successful ignitions. This was due to a current short to the single point ground, drawing current away from the Hollow Cathode Assembly heater. Thus, there was insufficient current at the heater to ignite the HCA. However, the Hollow Cathode Assembly and its heater were eliminated as causes for the short. Therefore, the conclusion was the Hollow Cathode Assembly is fully functional.

GFE–81:

ITEM:

Plasma Contactor Unit Hollow Cathode Assembly Part Number 62416J

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Plasma Contactor Unit Hollow Cathode Assembly shall not undergo an acceptance random vibration test at the component level.

RATIONALE:

The flight PCU Hollow Cathode Assemblies are subjected to the following testing during its workmanship screening:

Component–level Testing
All flight Hollow Cathode Assemblies undergoes ignition testing. This test is performed under vacuum conditions and consists of 10 on/off/cool cycles. During this test, the temperature of the Hollow Cathode Assembly varied from ambient to 2192 degrees F (1200 degrees C). All Hollow Cathode Assemblies are visually examined and performance data reviewed following the ignition testing and prior to Hollow Cathode Assembly acceptance as flight–certified units.

All flight Hollow Cathode Assembly heaters complete confidence testing. This testing consists of 150 on/off operational cycles. The Hollow Cathode Assembly heaters are visually examined and performance data reviewed following the confidence testing and prior to Hollow Cathode Assembly acceptance as flight–certified units.

High–fidelity developmental Hollow Cathode Assembly units identical to flight Hollow Cathode Assemblies are in life testing and ignition testing. These units have exceeded the 18,000 hours life requirement. One unit in ignition testing exceeded 42,000 ignitions, far surpassing the requirement of 6,000 ignitions with reliability greater than 99 percent.

Lower–fidelity developmental Hollow Cathode Assembly units with the same cathode and heater, but different wiring configuration and mounting flange, successfully completed vibration testing. The vibration testing consisted of sinusoidal vibration of 0.5 sine in 3 axes at 1 octave per minute and random vibration of 16.5 grms in 3 axes for 1 minute per axis.

ORU Level Testing
Flight Hollow Cathode Assemblies are vibration tested in the Plasma Contactor Unit in accordance with SSP 41172. Electrical functional testing and the clamp voltage test follow the vibration test. The first Hollow Cathode Assembly successfully operated as required.

GFE–82:

ITEM:

Mini–Pressurized Logistic Module Heater Battery Configuration Item M42080Q

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Thermal Vacuum Test on MPLM Heater Batteries will not be performed.

RATIONALE:

Alenia requests deletion of the acceptance thermal vacuum testing of the Ag–Zn batteries because of design and life characteristics.

Silver–Zinc batteries are limited life items that are stored dry until required for use. Three significant characteristics of the MPLM Heater Battery are a Dry (unactivated) life of 10 years, a Wet (activated) live of 12 months, and 10 Charge/Discharge cycles. Performing an acceptance thermal vacuum test after filling the cell with electrolyte would result in losing at least one Charge/Discharge cycle and some of the wet life. However, a qualification thermal vacuum test was performed on the cell design with no anomalies. All Ag–Zn battery cells are tested at lot and 100 percent level during final production. These Ag–Zn battery cells do have the same design characteristics of previously space–flown items. Finally, there are no battery parts or subassemblies considered to be sensitive to vacuum.

GFE–83:

ITEM:

Portable Fan Assembly Part Number 96M68020–1 Serial Numbers 001, 002, and 003

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The CO2 Removal Kit Portable Fan Assembly will not undergo an acceptance random vibration test.

RATIONALE:

The Portable Fan Assembly is launched packed in foam to protect it against potentially detrimental launch vibration loads. The Portable Fan Assembly is not operated during launch. The Portable Fan Assembly can only be hard–mounted to the shaker table via its hex stud. Analysis has indicated that testing in this manner will result in structural damage. Attaching the Portable Fan Assembly in some other way (strapping, clamps, etc.) would create unrealistic stress concentrations in contact areas that would likely lead to premature failure.

The vendor–supplied MIL–901 fan has flown on seven Spacehab missions with no failures. The same fan is used for military applications including the United Defense Crusader Vehicle, Bradley A3 Armored Vehicle, and the Hercules Helicopter.

GFE–84:

ITEM:

CO2 Removal Kit Portable Fan Assembly Part Number 96M52562

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same…" This requires that the component undergo random vibration testing in accordance with paragraph 5.1.4.

EXCEPTION:

The CO2 Removal Kit Portable Fan Assembly will not undergo a protoflight random vibration test.

RATIONALE:

The protoflight Portable Fan Assembly to be used in the CO2 Removal Kit is the Portable Fan Assembly Qualification Unit developed under ITA MT–30 Environmental Control and Life Support System Supporting Development.

The Portable Fan Assembly is launched packed in foam to protect it against potentially detrimental launch vibration loads. The Portable Fan Assembly is not operated during launch. The Portable Fan Assembly can only be hard–mounted to the shaker table via its hex stud. Analysis has indicated that testing in this manner will result in structural damage. Attaching the Portable Fan Assembly in some other way (strapping, clamps, etc.) would create unrealistic stress concentrations in contact areas that would likely lead to premature failure.

Additional workmanship screening is provided by the thermal cycling test and a burn–in test. The qualification thermal cycling test performed under ITA MT–30 Environmental Control and Life Support System Supporting Development included 24 operating cycles between –10 degrees F and 145 degrees F (service environment will be nominally 65 degree F to 80 degrees F) with a ramp rate of a minimum of 3.5 degrees F per minute. Portable Fan Assembly functional checks were performed at thermal extremes and before and after the thermal cycling test. A 300–hour Burn–In Test was also performed on this unit, and functional performance tests were performed before and after the Burn–In Test.

The vendor–supplied MIL–901 fan has flown on seven Spacehab missions with no failures. The same fan is used for military applications including the United Defense Crusader Vehicle, Bradley A3 Armored Vehicle, and the Hercules Helicopter.

GFE–85:

ITEM:

CO2 Removal Kit Portable Fan Assembly Part Number 96M52562

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification) with the following exceptions.”
Paragraph 6.1.1B. The minimum number of cycles shall be eight.

EXCEPTION:

The CO2 Removal Kit Portable Fan Assembly experienced 24 thermal cycles under ITA MT–30 Environmental Control and Life Support System Supporting Development Qualification Thermal Cycle testing.

RATIONALE:

The protoflight Portable Fan Assembly to be used in the CO2 Removal Kit is the Portable Fan Assembly Qualification Unit developed under ITA MT–30 Environmental Control and Life Support System Supporting Development. As such, this unit has experienced 24 thermal cycles during testing.

Thermal cycling is not life or fatigue limiting and should not reduce service life. Twenty–four thermal cycles provides more confidence in hardware operability. The thermal cycling test was followed by a 300–hour Burn–In Test with the Portable Fan Assembly shown to be operating properly at the conclusion of the test.

GFE–86:

ITEM:

Robotic Workstation Video Monitor Part Number 832281–501

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Robotic Workstation Video Monitors will not undergo an acceptance random vibration test.

RATIONALE:

Analysis has indicated that the risk of damage to this Commercial–Off–The–Self hardware during acceptance vibration testing exceeds the hardware confidence and reliability provided by performance of the test. However, other environmental testing performed on the Robotic Workstation Video Monitors including an acceptance thermal cycle test of eight cycles with a minimum 100–degree F sweep and acceptance burn–in testing of at least 300 hours mitigates infant mortality risks and provides adequate workmanship screening.

The Robotic Workstation Video Monitors are launched soft–stowed to protect it against potentially detrimental launch vibration loads. Six video monitors are manifested (three per Robotic Workstation). All six video monitors are reconfigurable and interchangeable, and redundant views are available to the video monitors.

Field experience data provided by the vendor IEC indicate a 3.5 percent failure rate relative to the advertised unit life of 20,000 operational hours. The majority of these ”failures” were for pixel defects that actually were within specification, not fatal failures of the equipment. This history provides additional confidence in the workmanship screening processes and reliability of the Robotic Workstation Video Monitors.

GFE–87:

ITEM:

Mini–Pressurized Logistics Module Configuration Item ISSA08A

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3C, Test Levels and Duration. The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total.

EXCEPTION:

Qualification Thermal Cycle testing of the Mini–Pressurized Logistics Module and its components shall be two times the number of thermal cycles as used for acceptance testing but not less than 8 cycles total.

RATIONALE:

The purpose for conducting qualification thermal cycle testing to three times the number of cycles used for acceptance is to demonstrate the capability to performed two additional acceptance thermal cycle tests on a flight unit if needed. Since SSP 41172 requires a minimum of eight cycles for acceptance, the minimum number for qualification becomes 24. For MPLM hardware, the minimum number of acceptance thermal cycles is four (see exception GFE–88). Then, this requirement would indicate 12 qualification thermal cycles. This would demonstrate capability for two additional acceptance thermal cycle tests consistent with SSP 41172 intent. Performing eight qualification thermal cycles still demonstrates a capability to perform a full acceptance thermal cycle retest if required. The technical impact and risk of only qualifying to two times the number of acceptance thermal cycles is low.

GFE–88:

ITEM:

Mini–Pressurized Logistics Module Configuration Item ISSA08A

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.3C, Test Levels and Duration. The minimum number of temperature cycles shall be eight.

EXCEPTION:

The minimum number of temperature cycles during acceptance thermal cycle testing of the Mini–Pressurized Logistics Module and its components shall be four cycles.

RATIONALE:

The purpose of acceptance thermal cycle testing is to environmentally stress screen electrical and electronic components for latent workmanship defects. The effectiveness of a thermal cycle screen is a function of the temperature difference between the hot and cold extremes (delta T), the rate of temperature transition between the extremes (dT/dt), and the number of cycles. Of the three, the number of cycles is generally considered the least important to screen effectiveness with rate of temperature transition usually considered the most important. While there are fewer thermal cycles for MPLM hardware, the minimum rate of transition is greater (MPLM requires a transition rate of no less than 1 degree C per minute while SSP 41172 requires a minimum transition rate of 1 degree F per minute). The vast majority of workmanship defects will be precipitated in the first few cycles, and the likelihood of missing additional defects by only conducting four cycles instead of eight is considered minimal. Retesting hardware to full compliance would be neither cost nor schedule effective considering the minimal amount of additional confidence gained.

GFE–89:

ITEM:

Robot Micro Conical Tool Holster Part Number SEG33111852–301

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.4, Supplementary Requirements. Functional tests shall be conducted, as a minimum, at the maximum operating temperature plus 20 degrees F (11.1 degrees C) and at the minimum operating temperature minus 20 degrees F (11.1 degrees C) during the first and last operating cycle after the dwell and after return of the component to ambient temperature.

EXCEPTION:

Functional testing will not be performed on the Robot Micro Conical Tool Holster during Qualification Thermal Vacuum testing. Verification will be by similarity, based on thermal cycle and thermal vacuum testing of similar mechanisms and dry–film coatings used in the Robot Micro Conical Tool.

RATIONALE:

The Robot Micro Conical Tool Holster launch restraint arm fixture prevents functional testing when combined with the Robot Micro Conical Tool during Qualification Thermal Vacuum testing. Functional testing of the Robot Micro Conical Tool Holster would need to be performed during an additional independent Qualification Thermal Vacuum test. Via approval of this exception, the project will not have to incur an additional cost impact and schedule delay to perform such testing.

The Qualification Robot Micro Conical Tool and the Robot Micro Conical Tool Holster will be exposed to three qualification level thermal cycles with functional tests at the maximum and minimum temperatures in addition to acceptance thermal testing performed on the flight units. This will provide verification of the Robot Micro Conical Tool Holster mechanism functionality at thermal extremes.

The Holster's use of dry–film coatings on the launch restraint mechanisms will be verified by similarity. The identical coatings are used on various mechanisms inside the Robot Micro Conical Tool with similar loading profiles that will be exposed to cycle life testing (greater than three cycles) over three thermal cycles in vacuum with functional tests performed at minimum, intermediate, and maximum temperatures.

The Robot Micro Conical Tool Holster is scheduled for one planned release during flight operations.

Thus, the removal of functional testing during the qualification thermal vacuum testing of the Robot Micro Conical Tool Holster will reduce cost and schedule delays in the test program without impacting validation of requirements.

GFE–90:

ITEMS:

Robot Micro Conical Tool Part Number SEG33111851–301

Robot Micro Conical Tool Holster Part Number SEG33111852–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.2, Thermal Vacuum Test, Component Acceptance.
Paragraph 5.1.2.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.2.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An Acceptance Thermal Cycle Test on the Robot Micro Conical Tool and the Robot Micro Conical Tool Holster will be performed in lieu of an Acceptance Thermal Vacuum Test.

RATIONALE:

Qualification Thermal Vacuum testing will be performed to verify no vacuum sensitive materials are present and Qualification and Acceptance Thermal Cycle testing will be performed to verify component performance.

The materials and processes used for both the qualification and flight Robot Micro Conical Tools and the Robot Micro Conical Tool Holsters are common and controlled. The materials for qualification and flight units are purchased as one lot, and fabrication and processes are performed on qualification and flight units at the same time. Additionally, the thermal chamber used during thermal cycle testing allows greater flexibility to take quantitative measurements while performing functional tests.

Therefore, the removal of acceptance thermal vacuum testing of the Robot Micro Conical Tool and the Robot Micro Conical Tool Holster will reduce cost and schedule delays in the test program without impacting validation of requirements.

GFE–91:

ITEMS:

Water Processor Assembly Part Number SV825500–1 Components as follows:
Waste Water ORU Part Number SV825412–1
Water Storage ORU Part Number SV825502–1
Water Delivery ORU Part Number SV825449–1
Separator Filter ORU Part Number SV825438–1
Particulate Filter ORU Part Number SV825442–1
Multifiltration Bed ORU Part Number SV825452–1
pH Adjuster ORU Part Number SV826778–1
Ion Exchanger Bed ORU Part Number SV825493–1
Microbial Check Valve ORU Part Number SV825499–1

Oxygen Generation Assembly Part Number SV825600–1 Components as follows:
Deionizing Bed (Inlet and Water Loop) ORU Part Number SV825569–1
Pump ORU Part Number SV825565–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycle Test, Component Qualification.
Paragraph 4.2.3.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

A qualification thermal cycling test will not be performed for the indicated components of the Water Processor Assembly or Oxygen Generation Assembly including accumulators, pump “head”, manual sample valve, and expendable ORUs.

RATIONALE:

The purpose of qualification thermal cycle testing is to demonstrate the ability of components to operate over the design temperature range and to survive the thermal cycling screening test imposed during acceptance testing. The subject components of the Water Processor Assembly and Oxygen Generation Assembly will be shown to be compliant with the above criteria without this testing for the following reasons:

Water Processor Assembly Accumulator:
Accumulator has a benign operating temperature range of 63 degrees F to 113 degrees F. Design contains no tight tolerances that would be affected by this operating thermal environment.

Accumulator with quantity sensor:
The quantity sensor potentiometer is a EEE part that goes through thermal cycling for screening. Accumulator has a benign operating temperature range of 63 degrees F to 113 degrees F. Design contains no tight tolerances that would be affected by this operating thermal environment.

Pump "head":
The pump "head" is constructed of ceramic and stainless steel materials that are not subject to thermal expansion effects on tolerances in the benign operating temperature range of 63 degrees F to 136 degrees F.

Manual sample valve:
Valve exposed to a benign operating temperature range of 63 degrees F to 113 degrees F.

Expendable ORUs:
These ORUs are non–complex mechanical items which are either packed chemical beds or contain paper elements with associated quick disconnects and plumbing. These are also exposed to a benign operating temperature range of 63 degrees F to 113 degrees F.

An analysis will be performed on the above listed devices to ensure that tolerances between dissimilar metals are adequate to prevent interferences or binding over the temperature range.

GFE–92:

ITEMS:

Water Processor Assembly Part Number SV825500–1, including all associated ORUs as follows:
Waste Water ORU Part Number SV825412–1
Water Storage ORU Part Number SV825502–1
Water Delivery ORU Part Number SV825449–1
Pump Separator ORU Part Number SV825426–1
Separator Filter ORU Part Number SV825438–1
Catalytic Reactor ORU Part Number SV825455–1
Gas Liquid Separator ORU Part Number SV825487–1
Reactor Health Sensor ORU Part Number SV826302–1
Sensor ORU Part Number SV825447–1
Multifiltration Bed ORU Part Number SV825452–1
Ion Exchange Bed ORU Part Number SV825493–1
Microbial Check Valve ORU Part Number SV825499–1
Particulate Filter ORU Part Number SV825442–1
pH Adjuster ORU Part Number SV826777–1

Oxygen Generation Assembly Part Number SV825600–1, including all associated ORUs and components as follows:
Deionizing Bed (Inlet and Water Loop) ORU Part Number SV825569–1
Pump ORU Part Number SV825565–1
Oxygen Outlet ORU Part Number SV825582–1
Water ORU Part Number SV827690–1
Hydrogen Sensor ORU Part Number SV826167–1
Hydrogen ORU Part Number SV827305–1
Process Controller ORU Part Number SV826025–1
Heat Exchanger Part Number SV825579–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An acceptance thermal cycling test will not be performed for the indicated components of the Water Processor Assembly and Oxygen Generation Assembly including solenoid valves, pressure regulators, quick disconnects, check valves, thermal expansion devices, manual sample valve, accumulator, accumulator with quantity sensor, and pump "head".

An acceptance thermal cycle testing will not be conducted on Water Processor Assembly/Oxygen Generation Assembly expendable ORUs and the following specific ORUs: waste water, pump separator, sensor, reactor health sensor, water storage, water delivery, hydrogen, oxygen/water, oxygen phase separator, and pump ORUs.

RATIONALE:

The purpose of acceptance thermal cycle testing is to environmentally screen components for latent workmanship defects. The subject components of the Water Processor Assembly and Oxygen Generation Assembly will be shown to be compliant with the above criteria without this testing for the following reasons:

NOTE: Unless otherwise noted, the following components will be exposed to a benign on–orbit operating temperature range of 63 degrees F to 113 degrees F.

Solenoid valves:
The solenoid valves are simple electrical devices. The only solder connection between the coil wire and lead wire is fully inspectable in accordance with NHB 5300.4 (3A–2). The valve position indicator contained within the valve is a previously–tested EEE part. The valve will undergo performance testing at the maximum operating temperature. Workmanship will be verified during run–in/pressure/leakage testing at the maximum operating temperature for devices above 125 degrees F.

Regulators:
The pressure regulators are constructed of stainless steel and Inconel and are not subject to thermal expansion effects on tolerances in the operating temperature range. Regulators will undergo performance testing at the maximum operating temperature. Workmanship will be verified during run–in/pressure/leakage testing at maximum operating temperature.

Quick Disconnects (hot and ambient):
Quick Disconnects are proof/leak tested at the maximum operating temperature. The crew will not disconnect Quick Disconnects while hot. Mate/demate cycles and flow/delta pressure testing at ambient temperature will verify workmanship.

Check valves (hot and ambient):
Check valves are constructed of stainless steel and Inconel and are not subject to thermal expansion effects on tolerances in the benign operating temperature range of 63 degrees F to 150 degrees F. Workmanship will be verified during run–in and subsequent crack/reseat and leakage testing at the maximum operating temperature.

Thermal expansion device:
The Thermal expansion device consists of a small bellows welded to a Quick Disconnect and is used to protect water–solid ORUs from thermal expansion pressure increases during transport. The device is constructed of Inconel. Workmanship will be verified during run–in/pressure/leakage testing at the maximum operating temperature.

Water Processor Assembly Accumulator:
Accumulator has a benign operating temperature range of 63 degrees F to 113 degrees F. Design contains no tight tolerances that would be affected by this operating thermal environment.

Accumulator with quantity sensor:
The quantity sensor potentiometer is a EEE part that goes through thermal cycling for screening. Accumulator has a benign operating temperature range of 63 degrees F to 113 degrees F. Design contains no tight tolerances that would be affected by this operating thermal environment.

Pump “head”:
The pump “head” is constructed of ceramic and stainless steel materials that are not subject to thermal expansion effects on tolerances in the benign operating temperature range of 63 degrees F to 136 degrees F.

Manual sample valve:
Valve exposed to a benign operating temperature range of 63 degrees F to 113 degrees F.

Expendable ORUs:
These ORUs are non–complex mechanical items which are either packed chemical beds or contain paper elements with associated quick disconnects and plumbing. These are also exposed to a benign operating temperature range of 63 degrees F to 113 degrees F.

Waste water, pump separator, sensor, reactor health sensor, water storage, water delivery, hydrogen, oxygen/water, oxygen phase separator, and pump ORUs:
These ORUs have a benign operating temperature range of 63 degrees F to 113 degrees F. Any internal components undergo acceptance thermal cycle testing as defined in SSP 41172 except for exceptions defined above. Justification will be provided as part of the verification report to ensure that assembly of the previously–tested components does not introduce risk for thermal effects at the ORU level.

GFE–93:

ITEMS:

Water Processor Assembly Part Number SV825500–1, including all associated ORUs as follows:
Waste Water ORU Part Number SV825412–1
Water Storage ORU Part Number SV825502–1
Water Delivery ORU Part Number SV825449–1
Pump Separator ORU Part Number SV825426–1
Separator Filter ORU Part Number SV825438–1
Catalytic Reactor ORU Part Number SV825455–1
Gas Liquid Separator ORU Part Number SV825487–1
Reactor Health Sensor ORU Part Number SV826302–1
Sensor ORU Part Number SV825447–1
Multifiltration Bed ORU Part Number SV825452–1
Ion Exchange Bed ORU Part Number SV825493–1
Microbial Check Valve ORU Part Number SV825499–1
Particulate Filter ORU Part Number SV825442–1
pH Adjuster ORU Part Number SV826777–1

Oxygen Generation Assembly Part Number SV825600–1, including all associated ORUs and components as follows:
Deionizing Bed (Inlet and Water Loop) ORU Part Number SV825569–1
Pump ORU Part Number SV825565–1
Oxygen Outlet ORU Part Number SV825582–1
Water ORU Part Number SV827690–1
Hydrogen Sensor ORU Part Number SV826167–1
Hydrogen ORU Part Number SV827305–1
Process Controller ORU Part Number SV826025–1
Heat Exchanger Part Number SV825579–1

SSP 41172 REQUIREMENT:

Paragraph 5.1.4, Random Vibration Test, Component Acceptance.
Paragraph 5.1.4.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.4.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

An acceptance random vibration test will not be performed for the indicated electrical components of the Water Processor Assembly and Oxygen Generation Assembly including solenoid valves, temperature sensors, heaters, and accumulators with quantity sensors.

An acceptance random vibration test will not be performed on the indicated all–mechanical components of the Water Processor Assembly and Oxygen Generation Assembly including check valves, relief valves, accumulator, thermal expansion device, manual sample valve, regulators, quick disconnects, and pump “head”.

An acceptance random vibration test will not be performed on the indicated Water Processor Assembly and Oxygen Generation Assembly expendable ORUs including Water Processor Assembly separator filter, particulate filter, multifiltration beds, pH adjuster module, ion exchange bed, microbial check valve, and startup filter kit, and the Oxygen Generation Assembly ion exchange beds.

RATIONALE:

Electrical or electronic components, except those discussed below, will be subjected to random vibration tests at acceptance levels. However, the following electrical components of the Water Processor Assembly and Oxygen Generation Assembly will be shown to be compliant with the criteria of adequate workmanship screening without this testing for the following reasons:

Solenoid valves:
The solenoid valves are simple electrical devices. The only solder connection between the coil wire and lead wire is fully inspectable per NHB. The valve position indicator contained within the valve is a previously–tested EEE part. Workmanship will also be verified during run–in, pressure, and leakage testing.

Temperature sensors:
The temperature sensors are simple electrical devices. The temperature sensor probes are a one–piece construction. The solder joint is fully inspectable per NHB.

Heaters:
The heaters are simple blanket–type electrical devices.

Accumulators:
The accumulator contains quantity sensors that are simple electrical devices (potentiometers) which are EEE parts that are previously tested.

These electrical components and all mechanical components listed in the exception will be protoflight random vibration tested after assembly into ORUs at the maximum predicted flight level for a duration of 1 minute in each axis.

The following Water Processor Assembly and Oxygen Generation Assembly expendable ORUs will be shown to be compliant with the criteria of adequate workmanship screening without this testing as follows:

Water Processor Assembly separator filter, particulate filter, multifiltration beds, pH adjuster module, ion exchange bed, microbial check valve, and startup filter kit, and the Oxygen Generation Assembly ion exchange bed ORUs:
These ORUs are non–complex mechanical items which are either packed chemical beds or contain paper elements with associated quick disconnects and plumbing. Chemical beds are vibrated as part of packing process.

GFE–94:

ITEM:

Space Vision System Artificial Vision Unit Part Number 000954–04

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests.
Paragraph 6.1.1B. For electrical/electronic components, the minimum operational temperature sweep shall be 100 degrees F (55.6 degrees C).

EXCEPTION:

The minimum Protoflight Thermal Cycle temperature sweep on the Artificial Vision Unit shall be 72 degrees F (32 to 104 degrees F).

RATIONALE:

The Artificial Vision Unit operates in the on–orbit crew environment of 62 to 82 degrees F. The Artificial Vision Unit Protoflight Thermal Testing did meet or exceed the following SSP 41172 requirements:

Ramp Rate – The thermal transition rate during thermal testing was 3.6 degrees F (2 degrees C) per minute. This exceeds the 1–degree F per minute minimum requirement.

Burn–In – Each Artificial Vision Unit has been subjected to 300 hours of burn–in on the entire unit.

Cold–Soak – Each Artificial Vision Unit was subjected to minimum 1–hour thermal equilibrium at the cold temperature.

Cycles – Each Artificial Vision Unit was subjected to 8 thermal cycles.

The Artificial Vision Unit has been tested to the maximum temperature values of its limiting components with no thermal anomalies.

The Artificial Vision Unit was designed with a large number of commercial components, the majority of which have a specified operating temperature range of 32 degrees F to 158 degrees F (0 degrees C to 70 degrees C). Examples of these components include the Brooktree NTSC Video Decoder Integrated Circuits BT812KPF located on the Video Input Circuit Card, for which there is no military or extended temperature range equivalent.

The internal temperature of the Space Vision System is on average about 20 degrees F (11 degrees C) warmer than the ambient air or cold plate cooling temperature while the Space Vision System is operating. A small number of components are known to operate as much as 45 degrees F (~25 degrees C) above the ambient air or cold plate cooling temperature. These temperatures are documented in Neptec report NDG001954–01, Test Results for the AVU Thermal Characterization Test. Therefore, while the test chamber sweep is 72 degrees F, the local ambient temperature of which certain components are exposed exceeds 100 degrees F.

The general reliability philosophy for the Artificial Vision Unit is that the unit is designed as an IVA Orbit Replaceable Unit which can be replaced on–orbit and repaired on the ground. A back–up unit is available in the event of failure.

GFE–95:

ITEM:

Space Vision System Artificial Vision Unit Part Number 000954–04

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When the assembly/components qualification tests are conducted on an assembly intended for subsequent flight, the test shall be the same (as defined in paragraph 4.2 for component qualification)…" This requires that electrical and electronic components be powered on and monitored for failures or intermittences during the protoflight random vibration test in accordance with paragraph 4.2.5.4.

EXCEPTION:

The Artificial Vision Unit will not be powered on and monitored for failures or intermittences during the protoflight random vibration test.

RATIONALE:

The Space Vision System Artificial Vision Unit cannot be powered on during vibration testing because the design incorporates some Commercial–Off–The–Shelf (COTS) components that are not rated for the vibration environment. The Space Vision System was designed in accordance with the requirements of Contract End Item Specification NDG001030. NDG001030, paragraph 3.7.6, specified that the Removable Hard Disk is COTS equipment to be stowed in a Shuttle Middeck locker (packed in energy–absorbing foam for launch and landing) and inserted into the Space Vision System for on–orbit operations. As COTS equipment, neither the Hard Disk Drive nor its connector was designed to be subjected to the random vibration levels of the protoflight test while operating. However, the drive is similar to those used for the Payload and General Support Computer in use.

The Space Vision System Artificial Vision Unit operation depends on operation of the Hard Disk Drive. The Artificial Vision Unit is only operated during the on–orbit environment.

Vibration testing has been performed on the Artificial Vision Unit using a procedure that includes a functional check of the unit before and after each axis of vibration, but with the unit powered off and the Hard Disk Drive removed during the actual vibration. That testing was successful.

The same test method is used for the Orbiter Space Vision Unit of the Space Vision System. No on–orbit vibration–related anomalies have been reported in its nine flights to date. The Orbiter Space Vision Unit qualification unit successfully completed Qualification random vibration testing for a 100–mission life. Moreover, each Artificial Vision Unit has been subjected to 300 hours of burn–in testing.

The general reliability philosophy for the Artificial Vision Unit is that the unit is designed as an IVA Orbit Replaceable Unit which can be replaced on–orbit and repaired on the ground. A back–up unit is available in the event of failure.

In addition, an engineering test was performed on the Space Vision System Qualification Unit, configured as an Orbiter Space Vision Unit on September 14, 2000, with the unit powered on. The Hard Disk Drive was isolated from the vibration by mounting it separately from the unit and making electrical connection through an extender cable. The extender cable included a mating connector to connect to the unit. The mating connector could not be isolated from the vibration input. The unit experienced a failure after approximately 10 seconds of vibration. The failure mode was consistent with a communication failure between the Hard Disk Drive and the Orbiter Space Vision Unit. The most probable cause was the Hard Disk Drive connector.

GFE–96:

ITEM:

Oxygen Recharge Compressor Assembly Part Number SEG29100906

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.
Paragraph 4.2.3.3B, Test Levels and Duration. For electrical/electronic equipment where the minimum sweep (paragraph 5.1.3.3–2) does not encompass by the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F.

EXCEPTION:

The thermal sweep during qualification thermal cycle testing of Oxygen Recharge Compressor Assembly was 100 degrees F under operating conditions.

Under operating and nonoperating conditions, the margin relative to the acceptance temperature limits was 10 degrees F.

RATIONALE:

During acceptance thermal cycle testing, the Oxygen Recharge Compressor Assembly experienced thermal temperatures from – 15 degrees F to 130 degrees F during non–operating conditions and from 30 degrees F to 110 degrees F during operating conditions. However, during qualification thermal cycle testing, the Oxygen Recharge Compressor Assembly experienced thermal temperatures from – 25 degrees F to 140 degrees F under non–operating conditions and from 20 degrees F to 120 degrees F under operating conditions. Thus, this provides less than the 140 degrees F sweep under operating conditions and less than the 20 degrees F thermal margin required by SSP 41172.

The Oxygen Recharge Compressor Assembly operates in the crew environment of 63 degrees F to 82 degrees F. It has been tested to the maximum temperature values of its limiting components, which are the cycle counter and O2 pressure switch components, whose limitations are –25 degrees F to 140 degrees F under non–operating conditions and –25 degrees F to 120 degrees F under operating conditions. No thermal anomalies were indicated during the testing, and the performance of the Oxygen Recharge Compressor Assembly at the temperature extremes was well within the requirements. Additionally, the thermal ramp rate during the thermal cycle sweep was greater than 1 degree F per minute as measured on the annunciator box as required by SSP 41172. Finally, an equivalent of 300 hours of burn–in has been completed on all electrical components as required by SSP 41172.

As such, no additional qualification thermal testing is required.

GFE–97:

ITEM:

Oxygen Recharge Compressor Assembly Part Number SEG29100906

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.

Paragraph 5.1.3.3B, Test Levels and Duration. For components identified in Table 5–1, with note 4, there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

EXCEPTION:

The thermal sweep during acceptance thermal cycle testing of the Oxygen Recharge Compressor Assembly was 80 degrees F (30 degrees F to 110 degrees F) under operating conditions.

RATIONALE:

The Oxygen Recharge Compressor Assembly operates in the crew environment of 63 degrees F to 82 degrees F. It has been tested to the maximum temperature values of its limiting components, which are the cycle counter and O2 pressure switch components, whose limitations are –25 degrees F to 140 degrees F under nonoperating conditions and –25 degrees F to 120 degrees F under operating conditions. No thermal anomalies were indicated during the testing, and the performance of the Oxygen Recharge Compressor Assembly at the temperature extremes was well within the requirements. Additionally, the thermal ramp rate during the thermal cycle sweep was greater than 1 degree F per minute as measured on the annunciator box as required by SSP 41172. Finally, an equivalent of 300 hours of burn–in has been completed on all electrical components as required by SSP 41172.

As such, no additional acceptance thermal testing is required.

GFE–98:

ITEM:

Oxygen Recharge Compressor Assembly Part Number SEG29100906

SSP 41172 REQUIREMENT:

Paragraph 4.2.10, Pressure Test, Component Qualification.
Paragraph 4.2.10.3, Test Levels. ALL REQUIREMENTS.
Paragraph 4.2.10.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The fully–assembled Oxygen Recharge Compressor Assembly will not undergo a Qualification Proof Pressure Test.

RATIONALE:

Proof pressure testing of all internal pressurized components and subassemblies of the Oxygen Recharge Compressor Assembly has been performed to 1.5 times the maximum design pressure in accordance with SSP 30559. All welds also underwent proof pressure testing at the subassembly level to 1.5 times the maximum design pressure. Additionally, leak testing at the maximum design pressure was performed on the fully–assembled Oxygen Recharge Compressor Assembly.

With knowledge that various Oxygen Recharge Compressor Assembly sensor readings would become inaccurate after exposure to proof pressure testing at 1.5 times the maximum design pressure, and the additional knowledge that the qualification unit will be refurbished for flight, to reduce the quantity of refurbishment, the fully–assembled Oxygen Recharge Compressor Assembly will not undergo a Proof Pressure Test at 1.5 times the maximum design pressure. The insight from the individual pressurized components, subassemblies, and welds proof pressure tests, and the leak testing of the fully–assembled Oxygen Recharge Compressor Assembly allow the proof pressure certification of the design.

GFE–99:

ITEM:

Oxygen Recharge Compressor Assembly Part Number SEG29100906

SSP 41172 REQUIREMENT:

Paragraph 4.2.10, Pressure Test, Component Qualification;
Paragraph 4.2.10.3, Test Levels. ALL REQUIREMENTS.
Paragraph 4.2.10.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Oxygen Recharge Compressor Assembly will not undergo a Qualification Ultimate Pressure Test.

RATIONALE:

SSP 41172 requires that ultimate pressure testing not be performed on actual flight articles as described in paragraph 4.2.10.2. As the qualification unit will be refurbished for flight, the Oxygen Recharge Compressor Assembly will not undergo an Ultimate Pressure Test. However, Oxygen Recharge Compressor Assembly Stress Analysis HDID–SAS–99–0024 has verified a positive margin of safety relative to the ultimate pressure requirements in SSP 30559, section 3. The Test and Verification Control Panel including the cognizant NASA Manufacturing, Materials, and Process Technology Division representative concurs with the analysis for the ultimate pressure certification of the design.

GFE–100:

ITEM:

Oxygen Recharge Compressor Assembly Part Number SEG29100906

SSP 41172 REQUIREMENT:

Paragraph 4.2.5, Random Vibration Test, Component Qualification.
Paragraph 4.2.5.3, Test Levels and Duration. The test duration in each of the three orthogonal axes shall be three times the expected flight exposure time to the maximum predicted level and spectrum or three times the component random vibration acceptance test time if that is greater, but not less than three minutes per axis.

EXCEPTION:

The duration of the Qualification Random Vibration Test of the Oxygen Recharge Compressor Assembly will be 5 minutes with a notched spectrum and one minute with an unnotched spectrum.

RATIONALE:

The qualification random vibration test duration of the Oxygen Recharge Compressor Assembly will be five minutes with a notched spectrum and one minute with an unnotched spectrum as indicated. However, this will not provide three times the component random vibration acceptance test time as required by SSP 41172.

The Oxygen Recharge Compressor Assembly is transported to the Airlock via the Orbiter and will be soft stowed in the mid–deck or the Mini–Pressurized Logistics Module during launch and landing. Then, the vibration test herein is used to verify workmanship, as the expected random vibration flight environment is minimal. As the Oxygen Recharge Compressor Assembly does undergo acceptance random vibration testing to an adequate workmanship screening level as required by SSP 41172, is powered and monitored during all vibration tests, and is a limited–life item that will be replaced with another unit when it reaches the end of its cycle life, additional random vibration testing during its qualification program is not warranted.

As the qualification unit will be refurbished for flight, it will undergo an acceptance random vibration test to the acceptance random vibration notched spectrum performed on other Oxygen Recharge Compressor Assemblies prior to any flight.

GFE–101:

ITEM:

Power Supply Assembly Part Numbers SEG39128211–303 and SEG39128211–305

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification
Paragraph 4.2.2.3B., Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

Paragraph 4.2.3 Thermal Cycling Test, Component Qualification
Paragraph 4.2.3.3B., Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees (11.1 degrees C) (minimum design temperature) during the cold portion of the cycle.

EXCEPTION:

The Power Supply Assembly experienced an operational temperature environment of 10 degrees F to 150 degrees F during Qualification Thermal Vacuum and Thermal Cycle testing. This provides 0 degrees F margin over the minimum operational thermal vacuum and thermal cycle temperatures experienced during the Power Supply Assembly acceptance thermal testing.

RATIONALE:

The full 24 cycles of Qualification Thermal testing included three cycles of operational thermal vacuum testing (10 degrees F to 150 degrees F), three cycles of nonoperational thermal vacuum testing (–10 degrees F to 150 degrees F), and 18 cycles of the standard Qualification Thermal Cycle Testing (10 degrees F to 150 degrees F). Since Acceptance Thermal Cycle and Thermal Vacuum Testing utilize a range of 10 degrees F to 110 degrees F, qualification did not provide the 20 degree margin on the low side as required by SSP 41172.

The Power Supply Assembly was exposed to three temperature cycles from –10 degrees F to 150 degrees F during Nonoperational Thermal Vacuum testing as part of the qualification test program. This demonstrates a 20 degree margin on the low side, although this was a nonoperational test. Functional testing was successfully performed at the completion of the nonoperational thermal vacuum test. Functional testing was also successfully performed at the temperature extremes during operational thermal vacuum testing. The standard acceptance test provides a 30 degree margin on the low side over the specified on–orbit range of 40 degrees F to 109 degrees F (operating and non–operating) as specified in its Interface Control Document.

The Power Supply Assembly is located in the Avionics Rack within the Equipment Lock. It is mounted to a coldplate that is maintained at a temperature not greater than 73 degrees F during normal operations. The Equipment Lock is at atmospheric pressure for most operations, and at 10.2 psia during pre–EVA “campout”. The nominal expected on–orbit temperature range in this area of the Airlock is from 62 degrees F to 83 degrees F.

A contingency scenario does exist whereby the Equipment Lock must be depressurized to allow an EVA to occur. This would be in the event of a Crewlock hatch failure. In such a scenario, the Power Supply Assembly would be required to operate in a vacuum. The coldplate would continue to maintain the Power Supply Assembly temperature. Even so, the Equipment Lock temperature is not predicted to go below the specified 40 degrees F.

The Power Supply Assembly utilizes a liquid crystal display. The display washes out (not readable) above 110 degrees F and begins to darken as the temperature approaches 10 degrees F.

Power Supply Assembly Serial Number 1002 is the primary flight article and is installed in the ISS Airlock. Power Supply Assembly Serial Number 1003 is the on–orbit spare and can be used to replace Serial Number 1002 on–orbit, if needed.

Additionally, burn–in testing (300 hours equivalent) was conducted on the flight units (Serial Numbers 1002 and 1003) with no anomalies.

Finally, the Power Supply Assembly utilizes EEE parts (rated for –125 degrees C to 125 degrees C) and an extended temperature liquid crystal display.

GFE–102:

ITEM:

Umbilical Interface Assembly Part Number SEG39128214–303

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The Umbilical Interface Assembly experienced temperatures of 0 degrees F to 140 degrees F (electrical functional at 0 degrees F and 140 degrees F; fluid functional at 40 degrees F and 140 degrees F) during Qualification Thermal Cycle testing. This does not provide a 20 degrees F margin at minimum temperature nor envelope the maximum temperature during Acceptance Thermal Cycle testing of the Umbilical Interface Assembly of 10 degrees F to 150 degrees F (non–operating at the extremes; operating at 40 degrees F to 140 degrees F).

RATIONALE:

The original acceptance test range was planned to be 40 degrees F to 140 degrees F to preclude freezing of water and to provide the 100 degree F sweep required by SSP 41172. The range of 10 degrees F to 150 degrees F was added to show compliance to the full non–operational temperature range.

Both electrical and fluids functional testing are performed at 40 degrees F and 140 degrees F. Since the unit contains water, any temperature at or below 32 degrees F on–orbit constitutes a freeze hazard. Therefore, the unit is first purged and dried before dropping the temperature below 32 degrees F.

Actual on–orbit temperatures are predicted to be 50 degrees F to 110 degrees F as specified in JSC 33237B, paragraph 4.4. The non–operational range of the Acceptance Test, therefore, provides a 40 degree margin at both minimum and maximum temperatures over the expected on–orbit temperatures. The operational test range provides margins of 10 degrees F and 30 degrees F, respectively.

The combined non–operating test range of 0 degrees F to 150 degrees F (0 degrees F from the qualification test, 150 degrees F from the acceptance test) provides a 50 degree margin at the minimum temperature and a 40 degree margin at the maximum temperature over the actual expected on–orbit non–operating temperatures. The 40 degrees F to 140 degrees F functional test range provides a 10 degree F margin on the low side and a 30 degree F margin on the high side over the actual expected on–orbit operating temperatures.

The 40 degrees F to 140 degrees F functional test range provides a 0 degree margin at the minimum temperature and a 31 degree margin at the maximum temperature over the on–orbit operating temperatures specified in the Project Technical Requirements Specification and Contract End Item specification.

The flight Umbilical Interface Assemblies have successfully completed their respective thermal acceptance tests.

Umbilical Interface Assembly Serial Number 1003 is the primary flight article and is installed in the ISS Airlock. Umbilical Interface Assembly Serial Number 1002 is the on–orbit spare and can be used to replace Serial Number 1003 on–orbit, if needed.

Additionally, Burn–in testing (300 hours equivalent) was conducted on the flight units (Serial Numbers 1002 and 1003) with no anomalies.

Finally, the Umbilical Interface Assembly utilizes EEE parts (rated for –125 degrees C to 125 degrees C). The higher acceptance test temperature does not affect these mechanical or fluid components.

GFE–103:

ITEM:

Metal Oxide Controller Part Number SV821750–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.2.5, Depress/Repress Vacuum Requirements. Internal components shall be subjected to a depressurization and repressurization test in accordance with either 4.2.2.5.1 or 4.2.2.5.2. A thermal vacuum qualification test in accordance with 4.2.2.1 through 4.2.2.4 may be substituted for this depressurization/repressurization qualification test.

EXCEPTION:

A Qualification Depress/Repress Test was not performed on the Metal Oxide Controller. Hamilton Sundstrand Space Systems International certified compliance by analysis.

RATIONALE:

The Metal Oxide Regenerator Assembly was certified by analysis to withstand a depressurization rate not to exceed 11 psia/minute, and a subsequent repressurization rate not to exceed 1.7 psia/second.

There are cavities within both the controller and the fan that could have been isolated and, therefore, susceptible to over–pressurization. These volumes are protected by vent holes. For the controller, the internal cavities that are not part of the air flow path are vented out the back of the controller through a series of ten 0.062–inch diameter holes. There is a volume that is not part of the air flow path in the fan assembly between the fan acoustic cover and the fan housing. A 0.166–inch diameter vent hole in the fan outlet support bracket ties the outlet fan flow to this volume to protect it from bulging or collapsing during depressurization and repressurization.

NASA performed an analysis of combinations of worst case conditions through ISS peak usage times. The analysis showed that the availability of the Metal Oxide Regenerator Assemblies is 99.68 percent for 12 EVAs per year and 98.54 percent for 40 EVAs per year. These numbers do not account for the spare ORUs available in inventory. In summary, there are three flight regenerator assemblies, of which two will be on–orbit at the same time. In the event of a failure of one of these units, the second unit will be available to regenerate canisters. Also, there are an additional four flight controller ORUs to support the flight regenerators. Finally, there are ten METOX canisters planned to be on orbit at any given time, two of which are held as contingency for emergency EVA capability. In the unlikely event that both regenerator assemblies were inoperable, there would still be CO2 removal capability for several EVAs.

GFE–104:

ITEM:

Metal Oxide Controller Part Number SV821750–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.
Paragraph 4.2.3.3B, Test Levels and Duration. For electrical/electronic equipment where the minimum sweep (paragraph 5.1.3.3–2) does not encompass by the acceptance limits + and – the margin, the minimum sweep for qualification shall be 140 degrees F.

Paragraph 4.2.3.3C, Test Levels and Duration. The duration shall be three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total.

EXCEPTION:

The thermal sweep during qualification thermal cycle testing of the Metal Oxide Controller was 72 degrees F (18 degrees F to 90 degrees F).

The Metal Oxide Controller experienced 7 1/2 thermal cycles during qualification thermal cycle testing.

RATIONALE:

The Metal Oxide Controller is an ORU that is designed to be removable from the Metal Oxide Regenerator Assembly. The regenerator (specifically the air inlet to the controller) interfaces with the adjacent rack Temperature and Humidity Control system that provides cooling air. The controller operates in the range of 38 degrees F to 70 degrees F and is controlled by the Temperature and Humidity Control System. The controller is not allowed to operate without cooling air. The Controller Assembly is similar to other controller assemblies that have been designed for ISS, specifically, the Pump and Flow Control System and Cabin Temperature and Humidity Control System.

The controller was thermally conditioned based on the inlet process air that flows through the controller and based on the Temperature and Humidity Control System that provides cooling air.

The controller is made up of four printed circuit boards.

Fan control – Certified as part of the Avionics Air Assembly and is comprised of predominantly Class S parts.

Driver Board for power supply and heater – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

Display Board – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

Logic Board – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

All electronic components are inspected in accordance with NHB 5300.4. All cavity devices are PIND tested. Of all boards there are 13 components that did not meet grade B and they have been approved on NASPARs (EEE parts approval).

The controller was thermally cycled during qualification testing from 18 degrees F to 90 degrees F. This is 20 degrees F above and below the Temperature and Humidity Control control range of 38 degrees F to 70 degrees F. The controller was cycled over this temperature range 7 1/2 times. On each cycle, the unit was powered on at 30 degrees F while undergoing thermal cycling to its maximum qualification temperature and powered off at 30 degrees F while undergoing thermal cycling to its minimum qualification temperature.

Additionally, Hamilton Sundstrand Space Systems International performed random vibration testing in accordance with SP–T–0023. Qualification test levels were performed at the Regenerator assembly level because the mounted configuration at the regenerator level results in higher loads (4.7 grms at controller). The controller was powered and monitored during qualification vibration testing.

Finally, NASA performed an analysis of combinations of worst case conditions through ISS peak usage times. The analysis showed that the availability of the Metal Oxide Regenerator Assemblies is 99.68 percent for 12 EVAs per year and 98.54 percent for 40 EVAs per year. These numbers do not account for the spare ORUs available in inventory. In summary, there are three flight regenerator assemblies, of which two will be on–orbit at the same time. In the event of a failure of one of these units, the second unit will be available to regenerate canisters. There are an additional 4 flight controller ORUs to support the flight regenerators and there are also ten METOX canisters planned to be on orbit at any given time, two of which are held as contingency for emergency EVA capability. In the unlikely event that both regenerator assemblies were inoperable, there would still be CO2 removal capability for several EVAs.

GFE–105:

ITEM:

Metal Oxide Controller Part Number SV821750–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.
Paragraph 4.2.3.4, Supplementary Requirements. Functional tests shall be conducted, as a minimum, at the maximum operating temperature plus 20 degrees F (11.1 degrees C) and at the minimum operating temperature minus 20 degrees F (11.1 degrees C) during the first and last operating cycle after the dwell and after return of the component to ambient.

EXCEPTION:

The Metal Oxide Controller was powered and operated at 30 degrees F (8 degrees F below the minimum acceptance operating temperature) during the minimum temperature sweeps of the Qualification Thermal Cycle test.

RATIONALE:

The Metal Oxide Controller is an ORU that is designed to be removable from the Metal Oxide Regenerator Assembly. The regenerator (specifically the air inlet to the controller) interfaces with the adjacent rack Temperature and Humidity Control system that provides cooling air. The controller operates in the range of 38 degrees F to 70 degrees F and is controlled by the Temperature and Humidity Control System. The controller is not allowed to operate without cooling air. The Controller Assembly is similar to other controller assemblies that have been designed for ISS, specifically, the Pump and Flow Control System and Cabin Temperature and Humidity Control System.

The controller was thermally conditioned based on the inlet process air that flows through the controller and based on the Temperature and Humidity Control System that provides cooling air.

The controller is made up of four printed circuit boards.

Fan control – Certified as part of the Avionics Air Assembly and is comprised of predominantly Class S parts.

Driver Board for power supply and heater – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

Display Board – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

Logic Board – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

All electronic components are inspected in accordance with NHB 5300.4. All cavity devices are PIND tested. Of all boards there are 13 components that did not meet grade B and they have been approved on NASPARs (EEE parts approval).

The controller was thermally cycled during qualification testing from 18 degrees F to 90 degrees F. This is 20 degrees F above and below the Temperature and Humidity Control control range of 38 degrees F to 70 degrees F. The controller was cycled over this temperature range 7 1/2 times. On each cycle, the unit was powered up at 30 degrees F while undergoing thermal cycling to its maximum qualification temperature and powered off at 30 degrees F while undergoing thermal cycling to its minimum qualification temperature.

Additionally, Hamilton Sundstrand Space Systems International performed random vibration testing in accordance with SP–T–0023. Qualification test levels were performed at the Regenerator assembly level because the mounted configuration at the regenerator level results in higher loads (4.7 grms at controller). The controller was powered and monitored during qualification vibration testing.

Finally, NASA performed an analysis of combinations of worst case conditions through ISS peak usage times. The analysis showed that the availability of the Metal Oxide Regenerator Assemblies is 99.68 percent for 12 EVAs per year and 98.54 percent for 40 EVAs per year. These numbers do not account for the spare ORUs available in inventory. In summary, there are three flight regenerator assemblies, of which two will be on–orbit at the same time. In the event of a failure of one of these units, the second unit will be available to regenerate canisters. There are an additional 4 flight controller ORUs to support the flight regenerators and there are also ten METOX canisters planned to be on orbit at any given time, two of which are held as contingency for emergency EVA capability. In the unlikely event that both regenerator assemblies were inoperable, there would still be CO_2 removal capability for several EVAs.

GFE–106:

ITEM:

Metal Oxide Controller Part Number SV821750–2, Serial Numbers 0002 through 0008

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.
Paragraph 5.1.3.3B, Test Levels and Duration. For components identified in Table 5–1, with note 4, there shall be at least 100 degrees F (55.6 degrees C) sweep between the minimum and maximum test temperatures, and the minimum test temperature shall be below 30 degrees F (–1.1 degrees C) where possible.

Paragraph 5.1.3.3C, Test Levels and Duration. The minimum number of temperature cycles shall be eight.

EXCEPTION:

The thermal sweep during acceptance thermal cycle testing of the Metal Oxide Controller was 32 degrees F (38 degrees F to 70 degrees F).

The Metal Oxide Controller experienced 5 thermal cycles during acceptance thermal cycle testing.

RATIONALE:

The Metal Oxide Controller is an ORU that is designed to be removable from the Metal Oxide Regenerator Assembly. The regenerator (specifically the air inlet to the controller) interfaces with the adjacent rack Temperature and Humidity Control system that provides cooling air. The controller operates in the range of 38 degrees F to 70 degrees F and is controlled by the Temperature and Humidity Control System. The controller is not allowed to operate without cooling air. The Controller Assembly is similar to other controller assemblies that have been designed for ISS, specifically, the Pump and Flow Control System and Cabin Temperature and Humidity Control System.

The controller was thermally conditioned based on the inlet process air that flows through the controller and based on the Temperature and Humidity Control System that provides cooling air.

The controller is made up of four printed circuit boards.

Fan control – Certified as part of the Avionics Air Assembly and is comprised of predominantly Class S parts.

Driver Board for power supply and heater – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

Display Board – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

Logic Board – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

All electronic components are inspected in accordance with NHB 5300.4. All cavity devices are PIND tested. Of all boards there are 13 components that did not meet grade B and they have been approved on NASPARs (EEE parts approval).

The controller was thermally cycled during acceptance testing from 38 degrees F to 70 degrees F. This temperature range was established based on the controller's internal heat exchanger's control range. The controller was cycled over this temperature range 5 times. On each cycle, the unit was powered on at the minimum temperature of 38 degrees F after dwell, thermally cycled to the maximum temperature of 70 degrees F, then thermally cycled to the minimum temperature of 38 degrees F before powered off.

For additional workmanship screening, Hamilton Sundstrand Space Systems International performed random vibration testing in accordance with SP–T–0023, Paragraph 3.4.1, Acceptance Vibration Test (levels and duration). Acceptance test levels were performed at the controller level (6.83 grms). The controller was powered and monitored during acceptance vibration testing.

The hardware does experience approximately 157 hours of burn–in during acceptance testing. In addition to the burn–in during acceptance testing, subsequent testing was also performed at Marshall Space Flight Center. After acceptance testing and subsequent airlock testing at Marshall Space Flight Center, the hardware has not experienced less than 200 hours of burn–in.

Hardware development testing and qualification testing is summarized in Analysis Memorandum 98–007, dated August 19, 1998, METOX Regenerator Development Test Results. Additional electronic mapping was performed at the Space Power Electronics Lab and EMI testing was performed at Hamilton Sundstrand Space Systems International. The total burn–in time associated with this testing was approximately 270 hours.

Finally, NASA performed an analysis of combinations of worst case conditions through ISS peak usage times. The analysis showed that the availability of the Metal Oxide Regenerator Assemblies is 99.68 percent for 12 EVAs per year and 98.54 percent for 40 EVAs per year. These numbers do not account for the spare ORUs available in inventory. In summary, there are three flight regenerator assemblies, of which two will be on–orbit at the same time. In the event of a failure of one of these units, the second unit will be available to regenerate canisters. There are an additional 4 flight controller ORUs to support the flight regenerators and there are also ten METOX canisters planned to be on orbit at any given time, two of which are held as contingency for emergency EVA capability. In the unlikely event that both regenerator assemblies were inoperable, there would still be CO2 removal capability for several EVAs.

GFE–107:

ITEM:

Metal Oxide Controller Part Number SV821750–2, Serial Numbers 0002 through 0008

SSP 41172 REQUIREMENT:

Paragraph 5.1.8, Burn–In Test, Component Acceptance.
Paragraph 5.1.8.3C, Test Levels and Duration. For constant temperature burn–in (either at ambient or accelerated via elevated temperature), the total operating time shall be equivalent to 300 hours at ambient temperature.

EXCEPTION:

During acceptance testing, the total burn–in time is approximately 157 hours for the Metal Oxide Controller. After acceptance testing and subsequent airlock testing at Marshall Space Flight Center, the hardware has not experienced less than 200 hours of burn–in.

RATIONALE:

The Metal Oxide Controller is an ORU that is designed to be removable from the Metal Oxide Regenerator Assembly. The regenerator (specifically the air inlet to the controller) interfaces with the adjacent rack Temperature and Humidity Control system that provides cooling air. The controller is not allowed to operate without cooling air. The Controller Assembly is similar to other controller assemblies that have been designed for ISS, specifically, the Pump and Flow Control System and Cabin Temperature and Humidity Control System.

The controller was thermally conditioned based on the inlet process air that flows through the controller and based on the Temperature and Humidity Control System that provides cooling air.

The controller is made up of four printed circuit boards.

Fan control – Certified as part of the Avionics Air Assembly and is comprised of predominantly Class S parts.

Driver Board for power supply and heater – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

Display Board – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

Logic Board – Certified as part of the METOX Regenerator Assembly and is comprised of predominantly grade B components.

All electronic components are inspected in accordance with NHB 5300.4. All cavity devices are PIND tested. Of all boards there are 13 components that did not meet grade B and they have been approved on NASPARs (EEE parts approval).

During acceptance testing, the controllers are powered on during the three separate functional tests, and thermal cycling and random vibration testing. Each of the three functional tests take approximately 12 hours for a total of 36 hours. The Controller was powered during acceptance thermal cycle testing for 5 cycles. This encompassed the minimum required of 1.5 thermal cycles in accordance with SP–T–0023 and a 48–hour burn–in as defined in SV822000–2, Specification/Assembly Drawing, Metal Oxide Regenerator, Paragraph 4.6, for a total powered time of approximately 120 hours during thermal testing. During acceptance random vibration testing, the controller is powered on and monitored. This test takes approximately 1 hour.

In summary, the hardware sees approximately 157 hours of burn–in during acceptance testing. In addition to the burn–in during acceptance testing subsequent testing was also performed at Marshall Space Flight Center. After acceptance testing and subsequent airlock testing at Marshall Space Flight Center, the hardware has not experienced less than 200 hours of burn–in.

Hardware development testing and qualification testing is summarized in Analysis Memorandum 98–007, dated August 19, 1998, METOX Regenerator Development Test Results. Additional electronic mapping was performed at the Space Power Electronics Lab and EMI testing was performed at Hamilton Sundstrand Space Systems International. The total burn–in time associated with this testing was approximately 270 hours.

For additional workmanship screening, Hamilton Sundstrand Space Systems International performed random vibration testing in accordance with SP–T–0023, Paragraph 3.4.1, Acceptance Vibration Test (levels and duration). Acceptance test levels were performed at the controller level. The controller was powered and monitored during acceptance vibration testing.

Finally, NASA performed an analysis of combinations of worst case conditions through ISS peak usage times. The analysis showed that the availability of the Metal Oxide Regenerator Assemblies is 99.68 percent for 12 EVAs per year and 98.54 percent for 40 EVAs per year. These numbers do not account for the spare ORUs available in inventory. In summary, there are three flight regenerator assemblies, of which two will be on–orbit at the same time. In the event of a failure of one of these units, the second unit will be available to regenerate canisters. There are an additional 4 flight controller ORUs to support the flight regenerators and there are also ten METOX canisters planned to be on orbit at any given time, two of which are held as contingency for emergency EVA capability. In the unlikely event that both regenerator assemblies were inoperable, there would still be CO2 removal capability for several EVAs.

GFE–108:

ITEMS:

Prebreathe Hose Assembly Kit Part Number SJG 33112241–301 Serial Numbers Unavailable (Quantity 2) including the following:
Prebreathe Hose Assembly Containment Bag Part Number SEZ 33112234–301
Hose Assembly – Oxygen Part Number SEG 33112744–301 Serial Numbers 1003, 1004, 1005, 1006, 1007, 1008, 1009, 1010
Quick Don Mask Assembly Part Number SEG 33105020–301 Serial Numbers 1012, 1013
Relief Valve Tee Assembly Part Number SEG 33112233–303 Serial Numbers 1008, 1009
Tee Assembly Part Number SEG 33112233–301 Serial Numbers 1003, 1004
Special Tee Assembly Part Number SEG 33112233–305 Serial Numbers 1013, 1014

Prebreathe Hose Spares Kit Part Number SJG 33112747–301 Serial Number Unavailable (Quantity 1) including the following:
Prebreathe Hose Assembly Containment Bag Part Number SEZ 33112234–303
Hose Assembly – Oxygen Part Number SEG 33112744–301 Serial Number 1002
Extension Hose Assembly Part Number SEG 33105101–301 Serial Number 1001
Quick Don Mask Assembly Part Number SEG 33105020–301 Serial Number 1016
Relief Valve Tee Assembly Part Number SEG 33112233–303 Serial Number 1010
Tee Assembly Part Number SEG 33112233–301 Serial Number 1005
Special Tee Assembly Part Number SEG 33112233–305 Serial Number 1015

SSP 41172 REQUIREMENT:

Paragraph 5.1.9, Oxygen Compatibility Test, Component Acceptance.
Paragraph 5.1.9.2, Test Description. Each component shall be subjected to 10 oxygen pressurization cycles from ambient pressure (10 to 15 psia) to Maximum Design Pressure within 100 milliseconds.

EXCEPTION:

The Prebreathe Hose Assembly in the 120–foot configuration shall reach Maximum Design Pressure in 1 minute or less. The Prebreathe Hose Assembly in the 30–foot configuration shall reach Maximum Design Pressure in 20 seconds or less. This is a one–time exception for the three flight units delivered for Flight 7A as indicated above. Should the hardware provider be tasked to provide additional units, or to refurbish and retest these units, all testing shall be conducted in full compliance with the requirements of SSP 41172.

RATIONALE:

Hardware has been tested at or above the operational conditions that will be experienced on–orbit. Pressure cycles were initiated by connection of the Prebreathe Hose Assembly Quick Disconnect to the supply as will occur on–orbit. Both the worst–case configuration (shortest flow path) and the nominal configuration were tested. The worst–case configuration consisted of 30 feet of hose with the Quick Don Mask (this resulted in the hardware reaching Maximum Design Pressure 3 times faster than the nominal configuration). This configuration should not be seen on orbit. The nominal configuration consists of all 120 feet of hose, 3 tee assemblies, and the Quick Don Mask. All of the Prebreathe Hose Assembly components are identical to or very similar to Portable Breathing Apparatus components.

Due to late detection of the requirement oversight, testing was performed at the Johnson Space Center instead of the White Sands Test Facility. Based on the length of the Prebreathe Hose Assembly hose (approximately 120 feet) and the limitations of the test equipment, it was not possible to meet the 100–millisecond requirement. All future oxygen testing for environment acceptance workmanship screening shall be performed at White Sands Test Facility and will meet the requirement specified in 5.1.9.2.

GFE–109:

ITEM:

Audio Terminal Unit Part Number 3000001–301

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycle Test, Component Acceptance.
Paragraph 5.1.3.3B, Test Levels and Duration. ALL REQUIREMENTS.

EXCEPTION:

The Cupola Audio Terminal Unit experiences an acceptance thermal cycle test over its operating temperature range from 33 degrees F to 90 degrees F.

RATIONALE:

The Audio Terminal Unit performance in temperature range up to 96 degrees F is acceptable. Unit will meet performance requirements as well as Mean Time Between Failures and touch temperature requirements.

The Audio Terminal Unit experienced a qualification thermal cycle over an operating temperature range from 13 degrees F to 110 degrees F. This qualifies the design for an operating temperature range from 33 degrees F to 90 degrees F during its acceptance thermal cycle test.

However, in the ESA Cupola, the Audio Terminal Unit Coldplate temperature range is from 72 degrees F to 96 degrees F, as the supply temperature of the Node 3 High Temperature Internal Thermal Control System loop to the Cupola is in this range of 72 degrees F to 96 degrees F, non–selectable.

Harris, the Audio Terminal Unit manufacturer, performed a thermal analysis of the Audio Terminal Unit to 99 degrees F. The analysis indicated:

> The fiber optic output power level with a 99 degree F coldplate can be expected to be 0.2 dB less output power than the performance with a 90 degree F coldplate;
>
> The audio fiber optic link between Node 3 and the Cupola is not a marginal link. The signal passes from the Node 3 Audio Bus Coupler to the Cupola Audio Terminal Unit and vice versa, so there is only one element bulkhead interface involved; and
>
> Final link budget to be established via SSCN 002632.

The fiber optic loss of 0.2 dB was deemed acceptable.

The impact upon reliability (failure rate and Mean Time Between Failures) for Audio Terminal Units at a maximum coldplate temperature of 99 degrees F were:

The predicted failure rate at a maximum coldplate temperature of 99 degrees F is 13.79 failures per million. This is within the allotted failure rate for Audio Terminal Units of 25.85 failures per million with grade 2 parts. All EEE parts derating were included in these calculations.

Harris analysis confirmed the speaker, keypad, and microphone will operate at higher temperature by consulting with manufacturer. No performance problems were identified for operating to a maximum temperature of 117 degrees F.

An assessment of the Audio Terminal Unit touch temperatures (including the keypad, speaker, and microphone) as adjusted for a coldplate temperature at 96 degrees F was developed. Harris used knowledge acquired during the Audio Terminal Unit qualification to measure hot spots. These hot spots guided analysis. The analysis validated:

The maximum chassis temperature of 120 degrees F would not be exceeded. The chassis temperature (at its hot spot) would be 113.8 degrees F.

The maximum front panel touch temperature of 113 degrees F would not be exceeded. The front panel touch temperature would be 111.6 degrees F.

Thus, the as–tested Audio Terminal Units for use in the Cupola are deemed acceptable.

GFE–110:

ITEM:

Oxygen Recharge Compressor Assembly Part Number SEG29100906

SSP 41172 REQUIREMENT:

Paragraph 5.1.6, Pressure Test, Component Qualification.
Paragraph 5.1.6.3, Test Levels. ALL REQUIREMENTS.

EXCEPTION:

The fully–assembled Oxygen Recharge Compressor Assembly will not undergo an Acceptance Proof Pressure Test.

RATIONALE:

Proof pressure testing of all internal pressurized components and subassemblies of the Oxygen Recharge Compressor Assembly has been performed to 1.5 times the maximum design pressure in accordance with SSP 30559. All welds also underwent proof pressure testing at the subassembly level to 1.5 times the maximum design pressure. Additionally, leak testing at the maximum design pressure was performed on the fully–assembled Oxygen Recharge Compressor Assembly.

With knowledge that various Oxygen Recharge Compressor Assembly sensor readings would become inaccurate after exposure to proof pressure testing at 1.5 times the maximum design pressure, the fully–assembled Oxygen Recharge Compressor Assembly will not undergo a Proof Pressure Test at 1.5 times the maximum design pressure. The insight from the individual pressurized components, subassemblies, and welds proof pressure tests, and the leak testing of the fully–assembled Oxygen Recharge Compressor Assembly allow the proof pressure certification of the design.

GFE–111:

ITEM:

Floating Potential Measurement Unit (FPMU) Part Number 39–0001

SSP 41172 REQUIREMENT:

Paragraph 4.2.3, Thermal Cycling Test, Component Qualification.
Paragraph 4.2.3.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 4.2.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Floating Potential Measurement Unit will not experience a Qualification Thermal Cycle test.

RATIONALE:

Qualification thermal cycle testing under ambient pressure conditions could result in condensation on the FPMU probe electronics that may damage the FPMU. To mitigate this, the FPMU will undergo twelve qualification thermal vacuum cycles at the MSFC thermal test chamber. The MSFC test chamber is capable of a transition rate of 1 degree F per minute; however, no quantitative assessment of the achievable thermal ramp rate of the FPMU may be made. To improve the thermal ramp rate, the FPMU will be attached to the chamber via an aluminum isolator instead of the flight–designed titanium isolator as documented and approved by a configuration variance.

Additional workmanship screening is provided on the qualification FPMU via certified processes for all solders either by hand by NASA qualified technicians or by Surface Mount Technology, powered–on qualification random vibration testing meeting the requirements of SSP 41172, and electronics bake out of both the individual electronics board and the partially–assembled electronics box. Thus, the risk associated with this exception is minimal.

GFE–112:

ITEM:

Floating Potential Measurement Unit (FPMU) Part Number 39–0001

SSP 41172 REQUIREMENT:

Paragraph 5.1.3, Thermal Cycling Test, Component Acceptance.
Paragraph 5.1.3.3, Test Levels and Duration. ALL REQUIREMENTS.
Paragraph 5.1.3.4, Supplementary Requirements. ALL REQUIREMENTS.

EXCEPTION:

The Floating Potential Measurement Unit will not experience an Acceptance Thermal Cycle test.

RATIONALE:

Acceptance thermal cycle testing under ambient pressure conditions could result in condensation on the FPMU probe electronics that may damage the FPMU. To mitigate, the FPMU will undergo four acceptance thermal vacuum cycles at the Space Dynamics Laboratory thermal test chamber. The test chamber is capable of a transition rate of 3.6 degrees F (2 degrees C) per minute on cool down and 5.4 degrees F (3 degrees C) per minute on warm up; thus, the FPMU will experience a thermal ramp rate greater than 1 degree F per minute during the acceptance thermal vacuum test. To improve the thermal ramp rate, the FPMU will be attached to the chamber via an aluminum isolator instead of the flight–designed titanium isolator as documented and approved by a configuration variance.

Additional workmanship screening is provided on the flight FPMUs via certified processes for all solders either by hand by NASA qualified technicians or by Surface Mount Technology, powered–on acceptance random vibration testing meeting the requirements of SSP 41172, and electronics bake out of both the individual electronics board and the partially–assembled electronics box. Also, acceptance burn–in testing for a minimum of 60 hours at ambient thermal temperatures and 67 hours at 120 degrees F (49 degrees C) is performed to meet the SSP 41172–equivalent of 300 hours of burn–in. A minimum of twenty on/off cycles will be performed as part of the acceptance burn–in testing with ten cycles performed during the elevated portion of the burn–in. In view of this, the risk associated with this exception is minimal.

GFE–113:

ITEM:

Floating Potential Measurement Unit (FPMU) Part Number 39–0001

SSP 41172 REQUIREMENT:

Paragraph 4.2.2, Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.3B, Test Levels and Duration. The component shall be at the maximum acceptance limits plus a margin of 20 degrees F (11.1 degrees C) (maximum design temperature) during the hot portion of the cycle and at the minimum acceptance test temperature minus a margin of 20 degrees F (11.1 degrees C) (minimum design limits temperature) during the cold portion of the cycle.

EXCEPTION:

The minimum temperature during qualification thermal vacuum testing under non–operating conditions for the Floating Potential Measurement Unit shall be – 85 degrees F.

RATIONALE:

As limited by the FPMU electronics, the minimum temperature the FPMU is certified to experience without risk of hardware damage is – 85 degrees F (– 65 degrees C). Therefore, during four of the twelve qualification thermal vacuum cycles where non–operating minimums are tested, this condition limits the minimum temperature. However, during one of the four acceptance thermal vacuum cycles, the flight FPMUs are tested to a minimum extreme of – 76 degrees F (– 60 degrees C) to maximize the time allowed for EVA deployment under on–orbit vacuum conditions. Thus, under this non–operating minimum temperature condition, only 9 degrees F (5 degrees C) margin is proven by test.

GFE–114:

ITEM:

Floating Potential Measurement Unit (FPMU) Part Number 39–0001

SSP 41172 REQUIREMENT:

Paragraph 5.1.1**,** Functional Test, Component Acceptance.
Paragraph 5.1.1.2, Test Description. ALL REQUIREMENTS.

EXCEPTION:

Functional acceptance testing will be performed by electronics functional testing and ground calibration only. Acceptance functional testing utilizing a plasma chamber will not be performed.

RATIONALE:

The plasma chambers available for testing produce a plasma that is on the edge of the operating envelope of the probe and is not comparable to the ionospheric plasma environment. Design will be proven by qualification testing performed both in a plasma chamber and with electronics functional testing and calibration. Acceptance testing will only be performed by electronics functional testing and calibration which will demonstrate the probe performance in the operational range of interest. The Environments team has indicated that this is the best approach for acceptance testing of the hardware.

GFE–115:

ITEMS:

Oxygen Generator Assembly (OGA), Part Number SV825600–1
Pump ORU, Part Number SV825565–1
Water Assembly ORU, Part Number SV827690–1
Frame and Tank N2 Purge Assembly, Part Number SV828110–1
Deionizing Bed ORU, Part Number SV825569–1
Process Controller OG ORU– SV826025–1
Sensor, Hydrogen ORU, Part Number SV826167–1
Oxygen Outlet Assembly ORU, Part Number SV825582–1
Hydrogen Dome Assembly ORU, Part Number SV827305–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.2. Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.5.2 Depress/Repress Vacuum Requirements, Automated Power Down, ALL REQUIREMENTS.

EXCEPTION:

A Qualification Depress/Repress Test will not be performed on the items indicated. Verification of this requirement shall be completed by analysis and test at the ORU or component level rather than at the Assembly level (as initially agreed to with the T&V Control Panel).

RATIONALE:

Verification of the hardware’s structural ability to withstand the pressure differential of depress/repress is provided by the structural analysis and the proof/leak test conducted on each ORU. The pressure differential required for the MDP analysis significantly exceeds the depress/repress pressure differential. Materials are accepted based on the materials compatibility analysis with a vacuum environment as documented in the Materials Identification and Usage List (MIUL). Furthermore, if a depress/repress event occurs, the OGA is automatically powered down to a state that protects components from the vacuum environment.

The components at risk to the depress/repress environment are those external to the proof pressure barrier, including thermal covers, acoustic covers, thermal insulation, and electronics. These components will be verified by similarity analysis when feasible and tested as required. In a worst–case scenario, damage to an acoustic cover will not impair the functionality of the OGA but will only result in increased acoustic levels (possibly exceeding the acoustic requirement). Damage to thermal covers and insulation will result in increased power consumption, but should not cause loss of OGA function. If the damage is significant, temperature sensors will detect the anomalous performance and shutdown the OGA to a safe state (ORU replacement will be required). Electronic boxes are vented to limit the pressure differential; adequate venting will be verified by similarity to previous designs. Conformal coating for electrical components are baked for 48 hours at 180 degrees F and at 10E–04 torr during the manufacturing process, thus proving the survivability of this coating when exposed to space vacuum.

The Water Assembly ORU, Nitrogen Purge Assembly ORU, and Deionizing Bed ORU will be accepted by analysis. These ORUs do not employ any components external to the proof pressure barrier; thus, they can be accepted based on the proof/leak test conducted at the ORU level and the materials analysis for vacuum exposure.

The Pump ORU will be accepted by analysis. This ORU does contain an acoustic cover external to the proof pressure barrier; an analysis based on a test of the first flight ORU will verify the acoustic cover is not damaged during exposure to a depress/repress event.

The Process Controller for the OGA will be accepted by analysis and by similarity to the Electrical Interface Box (EIB) built and tested by Hamilton Sunstrand for the Temperature and Humidity Controller System. The Process Controller covers contain vent holes, as does the EIB, to allow each controller compartment to depressurize/repressurize during depress/repress. The holes are sized so that the maximum delta pressure across the controller housing and cover is 1.5 psid. At a delta pressure of 1.5 psid and with a factor of safety of 1.5, the minimum margin of safety is 0.03.

The Hydrogen Sensor ORU will be accepted by analysis. This ORU does contain electronic components external to the proof pressure barrier; an analysis will verify the gaps in the electronic box can adequately vent during depress/repress to prevent an excessive pressure drop.

The Oxygen Outlet ORU will be accepted by similarity analysis. This ORU contains components external to the proof pressure barrier, including insulated lines and thermal insulation. The insulated lines will be accepted by similarity to those tested on the Water Processor Assembly (WPA) Reactor ORU (reference GFE–116 for WPA). The thermal wrap will be accepted based on similarity to the SpaceHab Heat Exchanger, Part Number SV823150, also built by Hamilton Sundstrand.

The Hydrogen Dome Assembly ORU will be accepted by similarity analysis. This ORU contains components external to the proof pressure barrier, including insulated lines and acoustic wrap. The insulated lines will be accepted by similarity to those tested on the WPA Reactor ORU (reference GFE–116 for WPA). The acoustic wrap is Bisco wrap and will be accepted based on similarity to the ISS IMV application and the WPA Pump Separator ORU, which are both tested.

The rack resident hardware located in the OGA Rack will be accepted based on analysis. These components can be accepted based on the proof/leak test conducted at the component level and the materials analysis for vacuum exposure.

GFE–116:

ITEMS:

Water Processor Assembly (WPA), Part Number SV825500–1:
Waste Water ORU, Part Number SV825412–1
Pump Separator ORU, Part Number SV825426–1
Separator Filter ORU, Part Number SV825438–1
Particulate Filter ORU, Part Number SV825442–1
Multifiltration Bed ORU, Part Number SV825452–1
Sensor ORU, Part Number SV825447–1
Catalytic Reactor ORU, Part Number SV825455–1
Gas Separator ORU, Part Number SV825487–1
Oxygen Filter ORU, Part Number SV828118–1
Reactor Health Sensor ORU, Part Number SV826302–1
pH Adjuster ORU, Part Number SV826777–1
Ion Exchange Bed ORU, Part Number SV825493–1
Water Storage ORU, Part Number SV825502–1
Water Delivery ORU, Part Number SV825449–1
Process Controller ORU, Part Number SV826000–1
Microbial Check Valve ORU, Part Number SV825499–1
Startup Filter ORU, Part Number SV825425–1

SSP 41172 REQUIREMENT:

Paragraph 4.2.2. Thermal Vacuum Test, Component Qualification.
Paragraph 4.2.2.5.2 Depress/Repress Vacuum Requirements, Automated Power Down. ALL REQUIREMENTS.

EXCEPTION:

A Qualification Depress/Repress Test will not be performed on the items indicated. Verification of this requirement shall be completed by analysis and test at the ORU or component level rather than at the Assembly level (as initially agreed to with the T&V Control Panel).

RATIONALE:

Verification of the hardware's structural ability to withstand the pressure differential of depress/repress is provided by the structural analysis and the proof/leak test conducted on each ORU. The pressure differential required for the MDP analysis significantly exceeds the depress/repress pressure differential. Materials are accepted based on the materials compatibility analysis with a vacuum environment as documented in the Materials Identification and Usage List (MIUL). Furthermore, if a depress/repress event occurs, the WPA is automatically powered down to a state that protects components from the vacuum environment.

The components at risk to the depress/repress environment are those external to the proof pressure barrier, including thermal covers, acoustic covers, thermal insulation, and electronics. These components will be verified by similarity analysis when feasible and tested as required. In a worst–case scenario, damage to an acoustic cover will not impair the functionality of the WPA but will only result in increased acoustic levels (possibly exceeding the acoustic requirement). Damage to thermal covers and insulation will result in increased power consumption, but should not cause loss of WPA function. If the damage is significant, temperature sensors will detect the anomalous performance and shutdown the WPA to a safe state (ORU replacement will be required). Electronic boxes are vented to limit the pressure differential; adequate venting will be verified by similarity to previous designs. Conformal coating for electrical components are baked for 48 hours at 180 degrees F and at 10E–04 torr during the manufacturing process, thus proving the survivability of this coating when exposed to space vacuum.

The Particulate Filter ORU, Multifiltration Bed ORU, Ion Exchange Bed ORU, Startup Filter ORU, Oxygen Filter ORU, pH Adjuster ORU, and Microbial Check Valve ORU will be accepted by analysis. These ORUs do not employ any components external to the proof pressure barrier, thus they can be accepted based on the proof/leak test conducted at the ORU level and the materials analysis for vacuum exposure.

The Process Controller for the WPA will be accepted by analysis and by similarity to the Electrical Interface Box (EIB) built and tested by Hamilton Sundstrand for the Temperature and Humidity Controller System. The Process Controller covers contain vent holes, as does the EIB, to allow each controller compartment to depressurize/repressurize during depress/repress. The holes are sized so that the maximum delta pressure across the controller housing and cover is 1.5 psid. At a delta pressure of 1.5 psid and with a factor of safety of 1.5, the minimum margin of safety is 0.03.

The Waste Water ORU will be accepted by analysis. A proof/leak test of the ORU will verify the structural integrity of the pressurized components within the ORU. If a depress/repress event occurs on–orbit, the isolation valve (Item Number WP/0421–1) that vents the tank will be closed per the automated shutdown of the WPA, thus protecting the tank bellows. The electronics in the Two–Phase Fluid Sensor (liquid sensor) will be verified by analysis and by similarity to the EIB. The Two–Phase Fluid Sensor cover contains a vent hole, as does the EIB, to allow the electrical compartment to depressurize/repressurize during depress/repress. The hole is sized so that the maximum delta pressure across the housing and cover is 1.2 psid. At a delta pressure of 1.2 psid and with a factor of safety of 1.5, the minimum margin of safety is 0.66. Materials analysis documented in the MIUL will verify the materials can survive the vacuum environment.

The Pump Separator ORU will be accepted by analysis. A proof/leak test of the ORU will verify the structural integrity of the hardware. This ORU does contain an acoustic cover external to the proof pressure barrier; an analysis based on a test of the first flight ORU acoustic cover assembly will verify the cover assembly is not damaged during exposure to a depress/repress event. If a depress/repress event occurs on–orbit, the isolation valve (Item Number 1101) will be closed per the automated shutdown of the WPA to isolate the water–containing items within the ORU from vacuum exposure through the Separator Filter ORU.

The Sensor ORU will be accepted by analysis based on a depress/repress test of the first ORU built for the flight program.

The Separator Filter ORU will be accepted by analysis based on a depress/repress test of the first ORU built for the flight program.

The Catalytic Reactor ORU will be accepted by analysis. A proof/leak test of the ORU will verify the structural integrity of the hardware. This ORU does contains components external to the proof pressure barrier, including a thermal cover, an insulated heat exchanger, an insulated manifold, an insulated frame, insulated fluid lines, a thermal wrap, and the electronics in the conductivity sensors; an analysis based on testing will verify these items are not damaged during exposure to a depress/repress event. One test will consist of an ORU that contains, as a minimum, the thermal cover, the insulated heat exchanger, the insulated manifold, and the insulated frame. A sample of an insulated fluid line and the thermal wrap will also be tested. The conductivity sensor electronics will be accepted by similarity to the conductivity sensor tested in the Sensor ORU, which contains a similar electronics design in its conductivity sensor. If a depress/repress event occurs on–orbit, the three–way valve (Item Number 0231) will isolate the water–containing items within the ORU from vacuum exposure through the oxygen vent line per the automated shutdown of the WPA.

The Gas Separator ORU will be accepted by similarity analysis. A proof/leak test of the ORU will verify the structural integrity of the hardware. This ORU contains components external to the proof pressure barrier, including a thermal cover, insulated lines, and an insulated frame. These items will be accepted by similarity to the items tested on the WPA Catalytic Reactor ORU.

The Reactor Health Sensor ORU will be accepted by analysis. A proof/leak test of the ORU will verify the structural integrity of the hardware. This ORU does employ electronics in the conductivity sensors that are external to the proof pressure barrier. The sensor electronics are accepted based on similarity to a test of the Sensor ORU, which contains a similar electronics design in its conductivity sensor. This ORU does employ a thermal wrap that is external to the proof pressure barrier. The thermal wrap is accepted based on similarity to the test of the WPA Catalytic Reactor ORU, which contains a similar thermal wrap.

The Water Storage ORU will be accepted by analysis. A proof/leak test of each ORU will verify the structural integrity of the ORU. If a depress/repress event occurs on–orbit, the isolation valve (Item Number 0421–3) that vents the tank will be closed per the automated shutdown of the WPA, thus protecting the tank bellows. The Electronics in the Two–Phase Sensor (gas sensor) will be verified by analysis; it is identical to the Two–Phase Fluid Sensor used in the Waste Water ORU.

The Water Delivery ORU will be accepted by analysis. A proof/leak test of each ORU will verify the structural integrity of the ORU. This ORU contains components external to the proof pressure barrier, including the acoustic wrap on the delivery pump; an analysis based on a test of the first flight ORU delivery pump/acoustic wrap assembly will verify the wrap is not damaged during exposure to a depress/repress event.

The rack resident muffler system will be accepted by similarity to the mufflers found in the Avionics Air Assembly built and tested by Hamilton Sundstrand. The remainder of the rack resident hardware located in the rack will be accepted based on analysis. These components include tubing, cables, and other similar structural components that can be accepted based on the proof/leak test conducted at the component level and the materials analysis for vacuum exposure.

APPENDIX E BOEING HOUSTON APPROVED EXCEPTIONS

The following is a list of exceptions to this document taken by Boeing Houston. The exceptions to this document in no way eliminate the Contractor's responsibility for showing compliance to the sections 3.2 through 3.7 of the applicable specification.

BOE–01:

ITEMS:

External Stowage Platform Attachment Device Active Assembly, Part Number 26900–10001

Sub Assemblies:
- Berthing Claw Assembly, Part Number 26900–20016
- Strut Assembly, Part Number 26900–20009
- Compliant FSE Boss, Part Number 26900–20010
- Guide Vane Assembly, Part Number 26900–20031
- Strut Capture Assembly, Part Number 26900–20120

External Stowage Platform Attachment Device Passive Assembly, Part Number 26900–10002

Sub Assemblies:
- Capture Bar Assembly, Part Number 26900–20043
- Load Release Assembly, Part Number 26900–20080
- Socket Ball Housing FSE, Part Number 26900–20039

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. “When there is no dedicated qualification test article and all production articles are intended for flight usage, the test shall be the same (as defined in paragraph 4.2 for component qualification) with the following exceptions…” This requires that the components undergo a protoflight thermal vacuum test in accordance with paragraph 4.2.2 as modified by paragraph 6.1.1A.

EXCEPTION:

A Protoflight Thermal Cycle Test of a duration of three cycles (including a differential temperature test iteration where required) on the External Stowage Platform Attachment Device Active and Passive Assemblies is performed in lieu of the Protoflight Thermal Vacuum test.

RATIONALE:

Thermal vacuum testing would not reveal potential defects in workmanship, actuation or adjustment of mechanisms, or deterioration of materials of construction beyond what a thermal cycle test procedure could produce. Rigging costs to carry out all mechanism cycling plus integration and de–integration of top assemblies creates a very complex testing environment. Specifically:

A. There are no contained volumes on the assemblies in question subject to pressure differences. Actuated or adjusted mechanisms have sufficiently loose tolerances to accommodate material volume growth due to reduced pressure. For instance, the tightest tolerance occurring in either assemblies is a stainless steel roller bearing with tolerances of +0.0000 / –0.0002 inches within the bearing race. Hand calculations were used to prove negligible material growth by taking a 1 inch cube of aluminum alloy (7050, T7351) at standard conditions and lowering the pressure from 14.7 psia to 5.5E–12 psia. The aluminum is less stiff than stainless steel and would be expected to grow more in size than stainless steel. Linear growth of each edge of the aluminum cube increased from 1 inch to 1.0000005 inches. This dimensional delta is considered negligible in terms of the performance of the External Stowage Platform Attachment Device on–orbit functionality. Therefore, thermal vacuum tests would not prove functionality.

B. Material deterioration is not an area of concern. There are no materials of construction that exceed out–gas criterion levels in a vacuum environment. All lubricants used for the two assemblies are ‘dry’ lubricants (e.g. Ecoalube). Therefore, thermal vacuum tests would not yield significant results of material deterioration.

C. Finally, practical means of testing all External Stowage Platform Attachment Device mechanisms in a vacuum chamber other than that used in Human Thermal Vacuum testing do not exist. Remote actuation drivers that would operate in temperature extremes of –110 degrees F to 210 degrees F and on–orbit pressures would be costly and difficult to employ. The quality of integration and de–integration of the active and passive components would be subjectively assessed, a quality factor which cannot be acquired via remote activation.

Although SSP 41172 does not require a Thermal Cycling test for mechanisms, the External Stowage Platform Attachment Device test program will pursue thermal cycling, actuation, integration, and de–integration of all required components. The Thermal Cycle test plan includes a differential temperature test iteration where the assemblies to be integrated will be at a temperature delta of 150 degrees F and then mated to verify ease of integration. Each top assembly will be mated with the other at temperature extremes as part of the temperature delta iteration. Differential temperature integration of components will be performed where a credible on–orbit scenario exists.

Three cycles will be performed versus eight cycles as required in SSP 41172, paragraph 6.1.1B (Assembly and Components Protoflight Tests). Thermal Vacuum cycling requirements (paragraph 4.2.2.3) impose three thermal cycles (minimum) for testing the mechanisms. Eight cycles are intended to prove electrical components and their solder connections, which are subject to breakage in the presence of temperature changes. The External Stowage Platform Attachment Device has no electrical/electronic components and would not yield workmanship or material defects more readily from eight thermal cycles as it would from undergoing three cycles.

BOE–02:

ITEMS:

External Stowage Platform Attachment Device Active Assembly, Part Number 26900–10001

Sub Assemblies:
Berthing Claw Assembly, Part Number 26900–20016
Strut Assembly, Part Number 26900–20009
Compliant FSE Boss, Part Number 26900–20010
Guide Vane Assembly, Part Number 26900–20031
Strut Capture Assembly, Part Number 26900–20120

External Stowage Platform Attachment Device Passive Assembly, Part Number 26900–10002

Sub Assemblies:
- Capture Bar Assembly, Part Number 26900–20043
- Load Release Assembly, Part Number 26900–20080
- Socket Ball Housing FSE, Part Number 26900–20039

SSP 41172 REQUIREMENT:

Paragraph 6.1.1, Assembly/Components Protoflight Tests. "When there is no dedicated qualification test article and all production articles are intended for flight usage, the test shall be the same (as defined in paragraph 4.2 for component qualification) with the following exceptions…" This requires that the components undergo a protoflight random vibration test in accordance with paragraph 4.2.5 as modified by paragraph 6.1.1D.

EXCEPTION:

A Protoflight Random Vibration Test of the External Stowage Platform Attachment Device Active and Passive Assemblies (with all components listed above installed) is performed in lieu of the Protoflight Component Random Vibration test.

RATIONALE:

The components of the ESPAD are classified as "simple mechanisms" and are robust in design. The ESPAD assembly and its components are entirely mechanical with no electronics. Testing the ESP2 components at the assembly level in their launch configuration would best represent the flight environment for which the components would experience.