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GENERAL ENVIRONMENTAL TEST GUIDELINES (GETG)

FOR

PROTOFLIGHT INSTRUMENTS AND EXPERIMENTS

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PREFACE

This document has been prepared to provide guidance to project offices and engineering organizations in the development of Environmental Test Programs at MSFC. Tailored application of these guidelines during the project planning phase and their implementation during the verification phase should result in increased assurance of mission performance while eliminating the costs associated with additional prototype test hardware.

This document was made possible through the contributions of the various MSFC Science and Engineering Laboratories and the efforts of Ms. Lynda Hoagland of Sverdrup Technology. Their efforts are appreciated. Copies of this document may be obtained from the MSFC Documentation Repository, Mail Code CN22D.

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INTRODUCTION

1.1 INTRODUCTION

1.1.1 PURPOSE

This document establishes environmental test guidelines for Marshall Space Flight Center (MSFC) Space Transportation System (STS) experiments and instruments in a protoflight verification program. The document elaborates on environmental test requirements, provides guidance for the choice of test options, and describes acceptable test methods for implementing the environmental test requirements so that a flight-qualified end item is delivered for integration into the next level of assembly for flight on the Space Transportation System.

The test methods of this document are developed to support the following objectives:

- Provide minimum guidelines for environmental tests.
- Assess payload hardware suitability for its intended flight environment.
- Disclose deficiencies and defects in manufacturing/assembly.
- Provide for corrective action in the event of failure.

1.1.2 SCOPE

This document presents MSFC project personnel and contractors with source material and methods for preparing environmental qualification and acceptance test requirements for protoflight hardware and is consistent with established MSFC STS policies. Qualification programs which involve prototype hardware development and extensive testing of nonflight hardware are outside the scope of this document, as are environments not encountered by the STS and its payloads during flight.

This document is divided into two sections. Section I of this document contains general information, ground rules and assumptions, definitions and acronyms, and applicable and reference documents. Section II contains environmental test guidelines for each of the recommended tests.

1.1.3 APPLICATION

Application of the guidelines outlined in this document early in the project is essential. Use of the protoflight philosophy instead of a prototype approach may save as much as 70% of the cost of the first flight article. The protoflight approach to environmental testing of flight hardware is chosen to minimize test hardware funding requirements. This is achieved by minimizing non-flight test hardware such as prototypes, engineering models, and other types of hardware that are not flight worthy. The test program is carefully structured to combine the elements of qualification and acceptance testing into a joint qualification/acceptance test. The test requirements are designed to provide for mission success while taking advantage of the protoflight concept. This combination of test parameters demonstrates integrity without exposing the hardware to unnecessary fatigue prior to actual flight.

Figure 1-1, Flowchart for Selecting Environmental Test Requirements, depicts a flow for generic hardware from conception to completion of the final environmental test requirements. The characteristics of the flow depicted in Figure 1-1, when used in conjunction with NMI 8010.1, "Classification of NASA Payloads," and MMI 8030.2, "Policy on MSFC Payloads," can better define program test requirements and increase the accuracy of associated cost forecasts during the early phases of a project. Determination of the required tests and classification, identified in Figure 1-1, are the first steps. The flowchart begins with a decision as to the type of hardware to be developed and moves to the classification selection criteria where an environmental test class is established. As the classification is being determined, the environmental test program can begin to take form. Initially the tests can be chosen based on the knowledge of the hardware's intended use. Once the classification has been determined and the hardware purpose further defined, general test requirements, scheduling, and cost may be developed. As the program matures, the specific environmental test requirements will be evolved and the program will move into verification activities. Verification activities produce test results, reports, and corrective action if required. Some of the decisions in this flowchart are programmatic, such as the determination of the class of the hardware. The class of the hardware is important to the environmental test program. Figure 1-2 depicts the decision criteria used to determine classification. The class is decided by the program manager based on cost/risk considerations. Regardless of classification, flight hardware safety and functionality are the prime objectives of the verification program.

The payload classifications are defined as follows:

- CLASS A (Minimum Risk Approach) Payloads for which a minimum mission risk approach is necessary due to the extremely high cost of failure both in dollars and intangible factors.

- CLASS B (Risk/Cost Compromise) Payloads for which there is a capability to recover from an in-flight failure even though the costs are significant both in dollars and intangible factors.

- CLASS C (Economically Reflyable or Repeatable) Payloads for which a repeat flight is planned or can be implemented with low cost impact. Environmental testing and end item screening may be limited only after STS safety and compatibility and payload functional testing are ensured.

- CLASS D (Minimum Single Attempt Cost) Payloads with worthwhile objectives but for which only one single low cost flight will be made. Test requirements are necessary for safety and compatibility purposes only.

Experiments, subsystems, or components that are determined to be Class A by the project undergo a full safety, qualification, and acceptance environmental test verification program. The qualification program will include a prototype development approach, which involves extensive testing of non-flight hardware. This type of program is outside of the scope of this document.

Experiments, subsystems, or components that are determined to be Class B may use the protoflight approach for environmental testing. The environmental test program should ensure safety requirement verification and performance verification to maximize mission success. An in-depth selection of environmental tests is very important for providing a high level of confidence in the flight hardware.

Experiments, subsystems, or components that are determined to be Class C may use the protoflight approach for environmental testing. The environmental test program should encompass all safety requirement verification as identified by payload hazard reports. Additional environmental tests to verify performance capability are strongly recommended for mission success. The mission manager must determine the cost versus benefits of each additional environmental test.

Experiments, subsystems, or components that are determined to be Class D may use the protoflight approach for environmental testing. The environmental test program is performed to verify safety requirements as determined by the payload hazard reports. Further environmental tests may be used to increase confidence in the flight hardware performance during the mission.

Payloads with B or C level classifications require a more stringent environmental test program in order to verify performance as well as safety.

For hardware with lower classifications higher risks of hardware failure are acceptable as long as there is no propagation of failure which would endanger the carrier vehicle, crew, or other payload elements. Environmental test selections will reflect these reduced requirements.

All hardware must conform with applicable NASA safety requirements. In the event of any conflict with this document, the safety requirements shall take precedence.

In addition to this document other available documentation (some of which is referenced in Paragraph 1.2) should also be used to make programmatic decisions on required environmental testing.

1.1.4 GROUND RULES AND ASSUMPTIONS

1. Verification can be accomplished by test, analysis, demonstration, or inspection, or a combination of the four. The guidelines outlined in this document apply to the environmental test portion of the verification program.

2. Testing is the preferred method of verification.

3. The verification program should assure that the end item delivered to the next higher assembly area has been flight-qualified.

4. The verification program is to maximize the benefits obtainable by implementing the protoflight program concept, that is by combining the elements of flight acceptance and prototype qualification testing.

5. The document does not address ground support equipment (GSE) or GSE facilities. GSE requirements for STS experiments are located in MSFC-SPEC-1548.

6. In general, all components/subsystems will be protoflight tested prior to assembly into the end item hardware.

7. Subsystem assembly and verification will be performed incrementally until the total end item is integrated and functions as a system.

8. All external interfaces will be verified prior to delivery of the end item to the next higher level of assembly.

9. A functional test will be performed before and after each environmental exposure. Functional tests shall also be performed during electromagnetic interference (EMI) and thermal vacuum testing of electronic hardware. A functional test will be performed between each axis of vibration. If functional during ascent/descent, a functional test should be performed during the acoustic test.

10. Development testing as described in paragraph 1.5.1, Development Phase, will not be conducted on hardware intended for flight.

11. Acceptance level testing will be conducted on the protoflight integrated assembly.

12. If follow-on unit(s) are required, they will undergo acceptance level testing at both the component and integrated flight system levels of assembly.

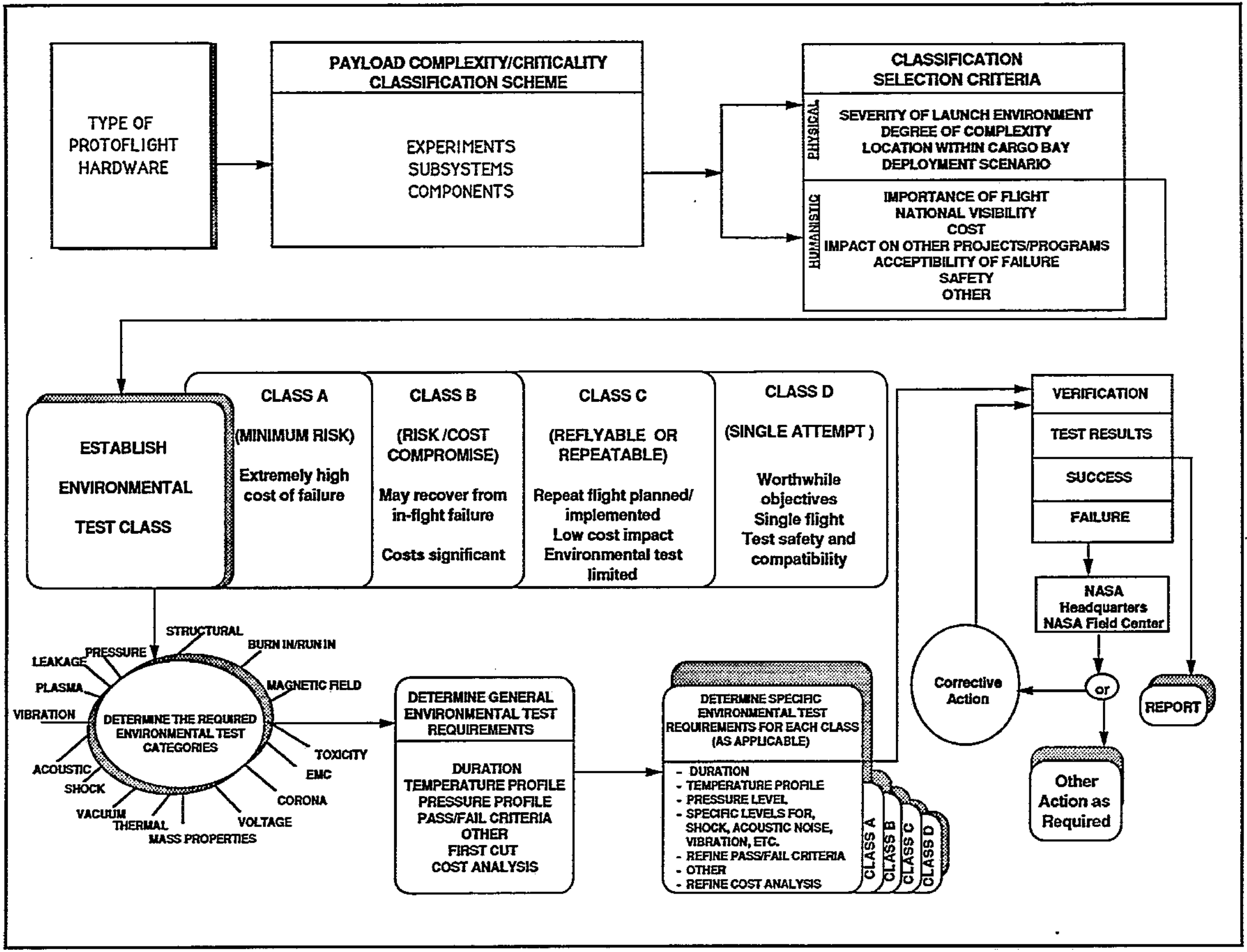


FIGURE 1-1 FLOWCHART FOR SELECTING ENVIRONMENTAL TEST REQUIREMENTS

DECISION FACTORS

- National significance of payload objectives
- Reflyability of payload
- Cost and schedule implications of in-flight soft failure [1]
- Alternative means of achieving objectives
- Impact on other projects and programs
- Any other factors that should be involved in determining the relative acceptability of risk of soft failure.

Is risk of soft failure unacceptable to the extent that systematic identification and avoidance or elimination of success critical [2] Single Failure Points (SFPs) should be a project policy?

YES → **CLASS A- MINIMUM RISK**

Payloads for which a minimum risk approach is clearly dictated by the prohibitively high cost of the consequences of failure, or by an unacceptable combination of costs and intangible factors associated with failure. (Not applicable to this document)

NO → To what extent is [3] re-flight, repeat flight, or achievement of objectives by other means an acceptable course of action in the event of in-flight soft failure?

ACCEPTABLE → **CLASS B- RISK/COST COMPROMISE**

Payloads for which an approach characterized by reasonable compromise between minimum risks and minimum costs is appropriate due to the capability to recover from in-flight failure by some means that is marginally acceptable even though it involves significantly high costs and /or highly undesirable intangible factors.

CLEARLY ACCEPTABLE → **CLASS C - ECONOMICALLY REFLYABLE OR REPEATABLE**

Payloads for which reflight or repeat flight is planned as a routine back-up in the event of inflight soft failure. Reflight or repeat flight costs are low enough to justify limiting qualification and acceptance testing to end item environmental screening. This is in addition to whatever is required for STS safety and compatibility and payload functional testing. There is no significant intangible or tangible impact of failure except the cost of repair and reflight, or repeat flight. Repair or reflight is estimable with reasonable confidence and is directly tradeable with inflight reliability enhancement costs. Therefore a decision criteria of minimum total expected cost is appropriate and practical.

NOT JUSTIFIED → **CLASS D- MINIMUM SINGLE ATTEMPT COST**

Payloads that have objectives worth achieving at a cost not to exceed the amount required for a single low cost attempt where formal verification requirements are limited to those necessary for safety and compatiblity.

The classification process consists of establishing the class as shown in this diagram and then identifying any exceptions to the subset of requirements for the class. The following notes are applicable to this diagram:

1. Soft Failure - Any equipment failure resulting in failure of a payload to meet its success criteria, but not resulting in any safety hazard, propagation of failure to the STS or any other payload equipment, or damage to hardware other than the failed payload.

2. A policy of "no avoidable SFPs" (Single Failure Points) is intended to serve as an identifiable benchmark at the lower threshold of Class A. The total project approach must of course be balanced and must therefore consider - in addition to treatment of SFPs - many engineering and assurance elements which are not addressed by MMI 8030.2A.

3. "Re-flight denotes the use of the same flight article to achieve the same objectives. "Repeat Flight" denotes the use of a new flight article to achieve the same objectives. "Alternate Means" denotes reformulation of missed objectives, and their achievement by other means such as integration into future projects.

FIGURE 1-2 PAYLOAD CLASSIFICATION PROCESS AND DEFINITION OF CLASSES

1.2 APPLICABLE AND REFERENCE DOCUMENTS LIST

The following documents are used to provide information, data, and requirements for the environmental testing of the protoflight hardware. The latest issue of the listed document should be used for environmental testing. Applicable documents are documents that are specifically referred to in the text of this document. Reference documents are listed for information. The reference documents were used in the research of this document and may be of interest/importance to the user.

1.2.1 APPLICABLE DOCUMENTS

MIL-STD-461	Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference
MIL-STD-462	Electromagnetic Emission and Susceptibility, Test Methods for
MIL-STD-463	Electromagnetic Interference Characteristics, Definition and Systems of Units
MIL-STD-810	Environmental Test Methods and Engineering Guidelines
MIL-STD-1541	Electromagnetic Compatibility Requirements for Space Systems
MIL-B-5087 (ASG)	Military Specification Bonding, Electrical, and Lightning Protection for Aerospace Systems
MSFC-STD-531	High Voltage Design Criteria
MSFC-SPEC-520	Electrical Transients, Systems and Components
MSFC-SPEC-521	Electromagnetic Compatibility (EMC) Requirements on Spacelab Payload Equipment
MSFC-SPEC-1238	Thermal Vacuum Bakeout Specification for Contamination Sensitive Hardware
MSFC-SPEC-1548	GSE Requirements for STS Experiments
MSFC-HDBK-505	Structural Strength Program Requirements
MSFC-HDBK-527	Materials Selection List for Space Hardware Systems
MSFC-HDBK-1453	Fracture Control Program Requirements
MSFC-PROC-536	Requirements and Procedures for Contamination Control due to Vacuum Outgassing
MSFC-PROC-1301	Guidelines for the Implementation of Required Materials Control Procedures
ICD 2-19001	Shuttle Orbiter/Cargo Standard Interfaces

JA-418	Payload Flight Equipment Requirements for Safety-Critical Structures
JSC 07700 (Vol XIV)	Space Shuttle System Payload Accommodations
MMI 8030.2	Policy on MSFC Payloads
NHB 5300.4 (3X)	Requirements for Electrostatic Discharge Control (Excluding Electronically Initiated Explosive Devices)
NHB 8060.1	Flammability, Odor and Offgassing Requirements and Test Procedures for Materials in Environments that Support Combustion
NMI 8010.1	Classification of NASA Payloads
NSTS 1700.7	National Space Transportation System Safety Policy
SLP/2104	Spacelab Payload Accommodation Handbook (SPAH)
SP-R-0022 (JSC)	General Specification, Vacuum Stability Requirements of Polymeric Material for Spacecraft Application

1.2.2 REFERENCE DOCUMENTS

MIL-STD-1540	Test Requirements for Space Vehicles
MIL-E-6051	Electromagnetic Compatibility Requirements, Systems
GEVS-STS	General Environmental Verification Specification for STS Payloads, Subsystems, and Components
NASA TM -86538	Design and Verification Guidelines for Vibroacoustic and Transient Environments

1.3 DEFINITIONS

ACCEPTANCE TESTS - The test process that demonstrates the hardware is acceptable for flight. The process also serves as a quality control screen to detect deficiencies and, normally, to provide the basis for delivery of an item under terms of a contract. Test levels and durations generally do not exceed the predicted extremes of the mission.

ACOUSTIC NOISE TEST - An environment induced by high-intensity acoustic noise associated with various segments of the flight profile. This noise manifests itself throughout the payload in the form of directly transmitted acoustic excitation and as structure-borne random vibration excitation.

ANALYSIS - A technical evaluation relating equipment design and use to predictions of actual design and operation. In cases where testing is not feasible, analysis may be used to verify requirements.

ASCENT/DESCENT PRESSURE PROFILE TEST - A test conducted by introducing differential pressure loads across the hardware. This test determines the ability of the hardware to withstand the mechanical stresses induced by ascent and descent.

BURN IN TEST - A test conducted on electrical components to demonstrate operability and prevent infant mortality by eliminating weak components. Specific electrical parameters are applied to the components for 300 hours.

COMPONENT - A functional subdivision of a subsystem and generally a self-contained combination of items performing a function necessary for the subsystem's operation. Examples are transmitters, integrated circuits, pulleys, actuators, motors, or batteries.

CORONA TEST - The presence of corona arcing is determined as a part of the Thermal Vacuum test.

DESIGN SPECIFICATION - A generic designation for a specification which describes functional and physical requirements for an article, usually at the component or higher levels of assembly. In its initial form, the design specification is a statement of functional requirements with only general coverage of physical and test requirements. The design specification evolves through the project life cycle to reflect progressive refinements in performance, design, configuration, and test requirements. In many projects the end-item specifications serve all the purposes of design specifications for the contract end-items. Design specifications provide the basis for technical and engineering management control. This control process is started with a document called the Contract End Item (CEI) Specification.

DESIGN QUALIFICATION TESTS - The test process that demonstrates the test item will meet performance specifications under simulated conditions. The simulated conditions are generally more severe than those expected from ground handling, launch, and orbital operations. The purpose is to uncover deficiencies in design and method of manufacture. The tests are not intended to exceed design safety margins or to introduce unrealistic modes of failure. Design qualification hardware is not normally flown, but may be used as spare hardware if deemed appropriate. Protoflight hardware is generally subjected to Qualification/Acceptance tests which combine elements of design qualification and acceptance tests.

DEVELOPMENT TEST - Tests performed during the development phase to verify the design concept and approach.

ELECTROMAGNETIC COMPATIBILITY (EMC) - The condition that prevails when various electronic devices are performing their non-degraded functions in a common electromagnetic environment. Electronic hardware is compatible when there is little or no interference or when the interference has no adverse effects

ELECTROMAGNETIC COMPATIBILITY TESTS - These tests are conducted to ensure that the hardware does not cause electromagnetic interference that will adversely affect the STS or other hardware.

ELECTROMAGNETIC INTERFERENCE (EMI) - Electromagnetic energy which interrupts, obstructs, or otherwise degrades or limits the effective performance of electrical equipment.

ELECTROMAGNETIC SUSCEPTIBILITY (EMS) - The tolerance of a component, subsystem, or system to all sources of conducted or radiated electromagnetic emissions.

ELECTROSTATIC DISCHARGE TESTS - A test conducted to verify there is no damage to hardware from the build up of static charge and its subsequent discharge.

END-TO-END TESTS - Tests performed on the integrated ground and flight system, including all elements of the payload, its control, stimulation, communication, and data processing to demonstrate that the entire system is operating in a manner to fulfill all mission requirements and objectives.

EXPERIMENT - Experiment is a term sometimes used to describe various parts of a payload. For example, an experiment may be a grouping of several different instruments that will be used to make an observation in space, or it may perform certain functions and record the data for interpretation in the laboratory.

FAILURE - A departure from the required design or operating specification in the functioning or operation of the hardware and/or software.

FUNCTIONAL TESTS - The operation of a unit in accordance with a defined procedure in order to determine the capability of the hardware to perform as required by the design or operating specification.

HARDWARE - As used in this document, the two major categories of hardware are as follows:

a. PROTOTYPE HARDWARE - This is hardware of a new design; it is usually subject to a design qualification test program and tested beyond expected life limits. It is not a flight article.

b. FLIGHT HARDWARE - This hardware is to be used operationally in space. It includes the following subsets:

    - <u>PROTOFLIGHT HARDWARE</u> -- Flight hardware of a new design. It is subject to a Qualification/Acceptance test program. The test article is the flight article.

    - <u>FOLLOW-ON HARDWARE</u> -- Flight hardware that is built in accordance with a design that has been qualified either through a prototype test program or by testing as protoflight hardware. Follow-on hardware is subject to a flight acceptance test program.

    - <u>SPARE HARDWARE</u> -- Flight hardware of a qualified design. It is subject to a flight acceptance test program. Spare hardware is used to replace flight hardware that is no longer acceptable for flight.

- REFLIGHT HARDWARE -- Flight hardware that has been used in space and is to be reused in the same way. The verification program to which it is subject depends on its past performance, current status, and the upcoming mission.

HIGH/LOW VOLTAGE TEST - A test to verify that the system will function correctly at upper and lower voltage operating limits and that the power sources and power users have compatible usage characteristics.

INSPECTION - A physical evaluation of equipment and associated documentation. Inspection is used to verify the physical dimensions, construction features, drawing compliance, workmanship, and physical condition of equipment.

INSTRUMENT - Hardware end item developed to perform a particular mission function. Several instruments make up an experiment.

LEAKAGE TEST - A test conducted on sealed components to verify that the leakage rate does not exceed design or operating specifications.

MAGNETIC PROPERTIES TEST - Tests performed to verify that magnetic fields produced by the hardware will not adversely affect its operation or the operation of other hardware.

MASS PROPERTIES TEST - The Mass Properties Test is a verification of the weight, center of gravity, and products and moments of inertia.

ORDNANCE SHOCK TEST - A test conducted to verify that the components can withstand the environment induced when a shock occurs, such as when pyrotechnic devices are actuated.

PAYLOAD - An integrated assemblage of experiments designed to perform a specified mission in space.

PLASMA TEST - A test conducted to expose components that are sensitive to ion electron volt energies to a charged particle environment so that the effects may be analyzed.

PROOF PRESSURE TEST - A test conducted to verify that there is no detrimental distortion, leakage, or rupture of the hardware or fittings when a particular pressure is applied. The pressure applied during the proof pressure test is usually greater than the pressure expected during the mission.

PROTOFLIGHT - Hardware designed and tested such that the test unit is also the flight unit. The protoflight unit is generally tested using the application of design qualification levels and flight acceptance durations. Testing to destruction is prohibited unless approved by the project manager. The refurbished unit is flown in space.

PROTOFLIGHT QUALIFICATION/ACCEPTANCE TESTS - A concept used in the protoflight design approach. The test program applies controlled environment test levels that generally exceed predicted flight parameters, but do not exceed design parameters. The test durations are, generally, those defined for acceptance tests.

PROTOTYPE - An emulation of the final design of a hardware unit that may be used in development testing or for interface design. It may or may not be functional. The prototype unit is generally tested beyond expected life-limits.

QUALIFICATION - Demonstration that design requirements and margins have been met.

RUN-IN TEST - A test conducted on mechanical and electromechanical components to condition moving parts and interfaces. This test helps to establish representative functional performance characteristics.

SUBSYSTEM - A functional subdivision of a payload consisting of two or more components. Examples are attitude control, electrical power, and communication subsystems.

TEMPERATURE CYCLE - A transition from some initial temperature condition to a specified temperature at one extreme (hot or cold) and then to a specified temperature at the opposite extreme (hot or cold) and returning to the initial temperature condition.

TEMPERATURE-HUMIDITY TESTS - Temperature-Humidity tests are conducted to verify that hardware functions as required when exposed to temperature and humidity extremes.

TEMPERATURE STABILIZATION - The condition that exists when the rate of change of temperature is within required limits. This occurs when the test item remains within the specified test temperature tolerance or when temperature variations will not adversely affect the test results or the hardware.

TEST - The operation of equipment under simulated or actual conditions. The equipment responses to specific environmental levels and durations are recorded.

THERMAL BALANCE TEST - A test conducted to verify the adequacy of the thermal model, the adequacy of the thermal design, and the capability of the thermal control system to maintain thermal conditions within established mission limits.

THERMAL CYCLING TEST - A test conducted to expose the hardware to temperature extremes and transitions to verify performance and to detect material and workmanship defects.

THERMAL-VACUUM TEST - This test demonstrates the capability of the test item to operate satisfactorily in vacuum at temperature extremes based on those expected for the mission. The test can also uncover latent defects in design, parts, and workmanship.

TORR - A measure of pressure that is equal to 1.3×10^{-3} (1/760) atmosphere.

TOXICITY TEST - A test performed to ensure that materials used in habitable environments do not evolve molecules that may be hazardous to crew members.

VERIFICATION - Methods (test, analysis, demonstration, and inspection) used to determine and ensure that the hardware conforms to design, construction, and performance requirements.

VIBRATION TEST - A test conducted to determine the resistance of hardware to stresses induced by random and sinusoidal excitation.

1.4 ABBREVIATIONS AND ACRONYMS

AC	Alternating Current
Am^2	Ampere Meter Squared
BW	Band Width
°C	degrees Celsius
CE	Conducted Emission
CG	Center of Gravity
cm	centimeters
CS	Conducted Susceptibility
dB	Decibels
DC	Direct Current
EED	Electro-Explosive Device
EMI/EMC	Electromagnetic Interference/Electromagnetic Compatibility
ESD	Electrostatic Discharge
g	gravity
°F	degrees Fahrenheit
GSE	Ground Support Equipment
Hg	Mercury
HPV	Highest Possible Voltage
Hz	Hertz
ICD	Interface Control Document
JSC	Johnson Space Center
LPV	Lowest Possible Voltage
MAX	Maximum
MIN	Minimum
mm	Millimeter
MSFC	Marshall Space Flight Center

N/A	Not applicable
N/R	Not Required
nT	nano (10^{-9}) Tesla
OSHA	Occupational Safety and Health Act
Pa	Pascals
pF	Pico (10^{-12})farad
PSI	Pounds per Square Inch
Q	Shock Response Spectrum Amplification Factor
RE	Radiated Emission
RH	Relative Humidity
RMS	Root Mean Square
RS	Radiated Susceptibility
SCCS	Standard Cubic Centimeters per Second
SFP	Single Failure Point
SPL	Sound Pressure Level
STS	Space Transportation System
T	Tesla
TBD	To Be Determined
$\mu N/m^2$	micro-Newtons per square meter
V	Volts

1.5 PROTOFLIGHT TESTING

1.5.1 TEST PHASES

There are three test phases, Development, Qualification/Acceptance, and Acceptance, defined in hardware development. The Qualification/Acceptance and Acceptance phases are the primary focus of this document. Figure 1-3 depicts testing associated with each of the three phases. Tables 1-1 and 1-2 represent requirements for a test program which will maximize the probability of mission success. Where a reduced test program is necessitated by programmatic cost/risk/classification considerations, test elimination should be done only after consultation with the appropriate technical organization(s).

A. Development Phase

The development test phase objective is to assure that testing of critical items at all levels of assembly is sufficient to validate the design approach. This generally requires the use of a development unit, and thus, this phase is included in this document for information and comparison to the two phases which are used in protoflight hardware development. The development or prototype unit will prove the feasibility of the design approach. It will demonstrate the capability of the selected configuration to satisfy performance requirements under ambient and environmental test conditions. This phase will develop confidence in the ability of the hardware to accomplish the mission objectives. Development tests are used in the confirmation of performance margins, manufacturability, testability, maintainability, reliability, life expectancy, and compatibility with system safety requirements. While the use of system level development tests is generally not applicable in a protoflight program, development testing on critical components, when possible, is recommended. It is recommended that development tests be conducted over a range of operating conditions exceeding the design limits to facilitate identification of marginal design features. Development tests do not satisfy the strength qualification requirements unless they meet or exceed the qualification test requirements of MSFC-HDBK-505, "Structural Strength Program Requirements."

There are occasions when a structural test article is used to verify design by using combined static plus dynamic loads. The strength qualification tests are generally performed in the static state. The tests are accomplished at the yield and ultimate levels when using the structural test article. The structural test article should be sufficiently representative of the proposed protoflight hardware that reliable data may be obtained. It will be instrumented and used in the modal survey, static, mechanisms development, acoustic noise, and mechanical compatibility tests. The results of the modal survey test will be used in assessing the predictions of the flight dynamic load analysis. The results of the acoustic test will be used to verify the acoustic math model. The unit may be used as the spacecraft mockup for wire and plumbing routing.

The design safety factors are contained in MSFC-HDBK-505, "Structural Strength Program Requirements," and in JA-418, "Payload Flight Equipment Requirements for Safety-Critical Structures."

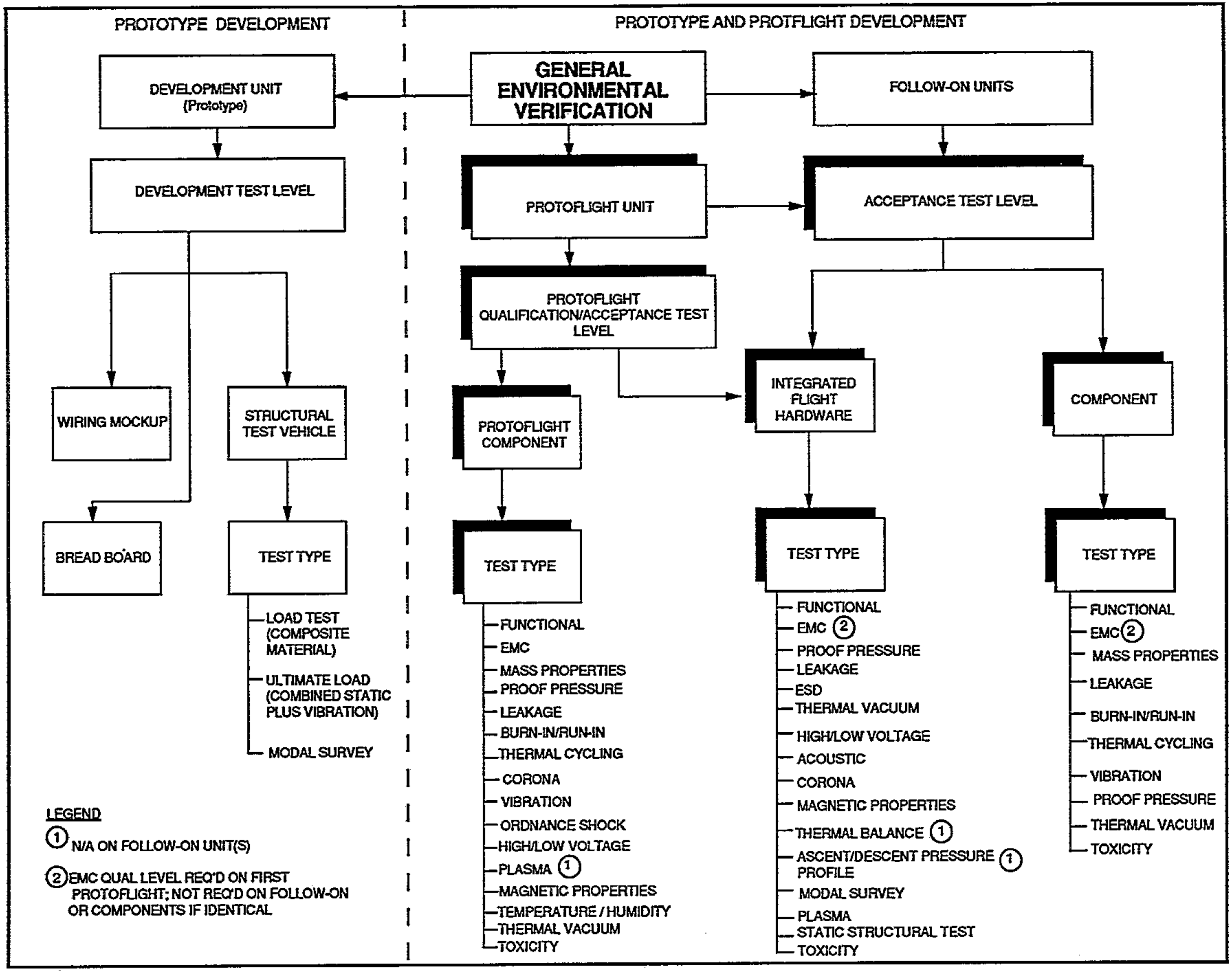


FIGURE 1-3 GENERAL ENVIRONMENTAL VERIFICATION DIAGRAM

B. Qualification/Acceptance Phase

A Qualification/Acceptance test program will be implemented at the protoflight component level only. This is accomplished, in most cases, by applying controlled environment test levels that generally exceed predicted flight level extremes but do not exceed the design values. The test durations are limited to those defined for flight acceptance testing. This type of program:

- Demonstrates that each component, as designed and built, exhibits a positive margin on the performance specification
- Avoids hardware fatigue by limiting the test durations to those defined for flight acceptance testing.

Table 1-1 lists the component Qualification/Acceptance tests for various types of equipment hardware.

C. Acceptance Phase

An Acceptance test program will be implemented for the protoflight integrated hardware.

The acceptance phase will demonstrate that the assembled protoflight hardware will meet the performance specifications when subjected to controlled environments generally equal to the maximum expected flight levels.

Table 1-2 lists the protoflight integrated acceptance level tests, both required and optional.

TABLE 1-1 Component Qualification/Acceptance Tests

TEST	REFERENCE PARAGRAPH	ELECTRONIC OR ELECT. EQUIP.	ANTENNAS	MECHANISMS OR ACTUATORS HYDRAULIC	MECHANISMS OR ACTUATORS ELECTROMECHANICAL	SOLAR PANELS	BATTERIES	VALVES	FLUID OR PROPULSIONEQUIP	PRESSURE VESSELS OR LINES	DEPLOYABLES	THRUSTERS OR ENGINES	THERMAL EQUIP. (HEAT PIPES, ETC).	OPTICAL EQUIP.	INSTRUMENTATION SENSORS	STRUCTURES
FUNCTIONAL	2.2.1	R (1)	R	R	R	R	R	R	R	R	R	R	R	R	R	R
VIBRATION	2.2.2.1	R	R (3)	R	R	R (3)	R	R	R (3)	R	R	R (3)	R	R	O (3)	O
ACOUSTIC NOISE (5)	2.2.2.2	O	O	O	O	R	O	O	O	O	O	O	O	O	O	O
ORDNANCE SHOCK	2.2.2.3	R	O	R	R	O	R	R	O	O	R	O	R	R	O	O
MASS PROPERTIES	2.2.2.4	R	R	R	R	R	R	R	R	R	R	R	R	R	R	R
PROOF PRESSURE	2.2.2.5	R (2)	O	R	O	O	R (2)	R	R (2)	R	O	R	O	O	O (7)	O
LEAKAGE	2.2.2.6	R (2)	O	R	O	O	R	R	R	R	O	O	O	O	O	O
STATIC STRUCTURAL	2.2.2.8	O	O	O	O	O	O	O	O	O	O	O	O	O	O	R
THERMAL VACUUM	2.2.3.1	R	O	R	R	O	R	R	R	O	R	R	R	R	R	O
THERMAL CYCLING (6)	2.2.3.2	R	O	R	R	R	R	O	O	O	O	O	R	O	O	O
TEMP/ HUMIDITY	2.2.3.4	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O
EMC	2.2.4	R	R	O	R	O	O	O	O	O	O	O	O	O	O	O
ESD	2.2.4.4	R	R	O	O	O	O	O	O	O	O	O	O	O	O	O
CORONA	2.2.4.5	O (4)	O	O	R	O	O	O	O	O	O	O	O	O	O	O
MAGNETIC PROPERTIES	2.2.4.6	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O
BURN/RUN-IN (8)	2.2.5	R	O	R	R	O	O	R	O	O	O	R	O	O	O	O
HIGH/LOW VOLTAGE	2.2.6	R	O	R	R	R	R	R	R	O	O	R	R	R	O	O
PLASMA	2.2.7	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O
TOXICITY	2.2.8	O	O	O	O	O	O	O	O	O	O	O	O	O	O	O

LEGEND: R = Required
O = Optional (An evaluation must be made for each application to determine if the test should be conducted.)

NOTES:
(1) Functional tests shall be conducted prior to and following each environmental test.
(2) On sealed equipment.
(3) An evaluation must be made to determine if the component type is such that an acoustic or vibration test is optional.
(4) The corona-arcing test is dependent on the susceptibility of the component to a vacuum environment.
(5) Acoustic noise tests are also to be run on isolated components and glass.
(6) An evaluation will determine if thermal cycling should be performed at vacuum. Operate equipment which is designed to function in a vacuum.
(7) An evaluation will determine if proof tests are required
(8) An evaluation will determine whether a burn-in or run-in test should be performed.

Table 1-2 summarizes the Required (R) and Optional (O) end item assembly acceptance tests.for protoflight integrated hardware.

TABLE 1-2 PROTOFLIGHT HARDWARE INTEGRATED ACCEPTANCE LEVEL TESTS

TEST	REFERENCE PARAGRAPH	REQUIRED (R) OR OPTIONAL (O)
FUNCTIONAL	2.2.1	R(1)
MODAL SURVEY	2.2.2.1	O(2)
ACOUSTIC	2.2.2.2	R(5)
PROOF PRESSURE	2.2.2.5	R
LEAKAGE	2.2.2.6	R
ASCENT/DESCENT	2.2.2.7	O(2)(3)
STATIC STRUCTURAL	2.2.2.8	O(2)
THERMAL VACUUM	2.2.3.1	R
THERMAL BALANCE	2.2.3.3	O(2)(3)
EMC	2.2.4	O(2)
ELECTROSTATIC DISCHARGE	2.2.4.4	O(2)
CORONA	2.2.4.5	O(2)
MAGNETIC PROPERTIES	2.2.4.6	O(2)
HIGH/LOW VOLTAGE	2.2.6	R
PLASMA	2.2.7	O(2)(3)
TOXICITY	2.2.8	O(2)

NOTES:
- (1) Functional tests are conducted prior to and following each test.
- (2) For optional tests, an evaluation must be made for each application to determine if the test should be conducted.
- (3) Test not applicable to follow-on integrated flight unit(s).
- (4) Test not required on follow-on unit(s) if are identical to the protoflight.
- (5) Acoustic tests are performed using Qualification/Acceptance test levels.

1.5.2 GENERAL TEST REQUIREMENTS

A. Test Tolerances

Each program should develop test tolerances compatible with the science and mission objectives. The tolerances imposed by higher level documentation and/or the project take precedence. The tolerances in this document represent typical test program tolerances. The values include measurement uncertainties. Note that all equipment used for testing shall be current in calibration.

1. Acoustics

Overall Level: ± 2 dB
1/3 Octave Band: ± 2 dB

2. Electromagnetic Compatibility

Voltage magnitude: ± 5% of the peak value
Current magnitude: ± 5% of the peak value
RF Amplitudes: ± 2 dB
Frequency: ± 2%
Distance: ± 5% of specified distance or ± 5 cm, whichever is greater.

3. Humidity

± 5% RH

4. Loads

Steady-State (Acceleration): + 5%, - 0%
Static: + 5%, - 0%

5 Magnetic Properties

Mapping distance measurement: ± 1 cm
Displacement of assembly center of gravity (cg) from rotation axis: ± 5 cm
Vertical displacement of single probe center-line from cg of assembly: ± 5 cm
Mapping turntable angular displacement: ± 3°
Magnetic field strength: ± 1 nT
Repeatability of magnetic measurements (short term): ± 5% or ± 2 nT, whichever is greater
Demagnetizing and Magnetizing Field Level: ± 5% of nominal

6. Mass Properties

Weight: ±0.2%
Center of Gravity: ±0.15 cm (±0.06 in.)
Moments of Inertia: ±1.5%

7. Mechanical Shock Pulse

Amplitude: +40%, -20%
Duration: ±10%

8. Pressure Levels

Greater than 1.3×10^4 Pa (Greater than 100 mm Hg): + 5%, - 0%
1.3×10^4 to 1.3×10^2 Pa (100 mm Hg to 1 mm Hg): + 10%, - 0%
1.3×10^2 to 1.3×10^1 Pa (1 mm Hg to 1 micron Hg): + 25%, - 0%
Less than 1.3×10^1 Pa (less than 1 micron Hg): + 80%, - 0%

9. Shock Response Spectrum (Q=10)

1/3 Octave Band Center Frequency Amplitude: +6, -3dB

(Shock response spectra monitored over 1/3 Octave Bands)

10. Temperature

± 2°C

11. Test Durations

+ 10%
- 0%

12. Vibration

Sinusoidal: Amplitude: +20, -10%
Frequency: ±10%

Random: RMS level: ± 10%
Power Spectral Density: +3, -1.5 dB

13. Deflection

+ 5%
- 0%

B. Performance

Deterioration or any change in performance of the test article could indicate a failure. A change in performance which detrimentally affects function or operation of the test article with respect to the design requirements of the article is considered to be a failure. A change in performance which clearly indicates a trend of degradation is considered to be a failure even is the measured performance is within specified minimums.

When a failure occurs, a test may continue to conclusion or be interrupted for corrective action. The determination of action should be carefully considered because any corrective action could affect the validity of previously performed tests. Previously performed tests shall be repeated as necessary to satisfactorily ensure performance.

Failures and anomalies shall be reported and documented.

C. Safety

Hazards associated with the environmental testing of the hardware should be identified. These hazards must be evaluated so that the test item, facilities, and personnel can be protected. All known hazards should be specified.

Hazards should be eliminated if feasible. If elimination of the hazard is not possible, automatic safety controls and warning devices should be incorporated into the system.

Facilities shall meet applicable Occupational Safety and Health Act (OSHA) standards and other requirements as appropriate. Personnel shall be trained and qualified to help ensure the safety of the payload, facilities, and personnel.

D. Spare Hardware

Spare hardware may be maintained in case of a flight hardware failure. Spare hardware must undergo an environmental test program just as if it were follow-on hardware. The anomalous hardware being replaced by the spare shall be evaluated for assurance that the fault is not generic.

E. Transportation and Handling

Care and caution should be exercised when packing, handling, and moving flight and flight spare hardware. Shipping containers shall be designed to prevent the imposition of stresses that exceed expected flight levels. Tests such as edgewise drop, cornerwise drop, and transportation vibration need not be performed on protoflight hardware being shipped in properly designed containers. Therefore, no transportation and handling tests are addressed in this document.

SECTION II
ENVIRONMENTAL TESTS

2.1 APPLICABILITY

This section contains the basic program for environmental testing of STS experiments, subsystems, and components. The basic provisions apply to protoflight hardware and are generally applicable to prototype hardware.

Each environmental test is organized in the following format.

A. Purpose - A short summary that describes the reasons for the test.

B. Requirements - Contains descriptions of the Qualification/Acceptance test requirements.

C. Acceptance Requirements - Contains descriptions of the Acceptance test requirements if applicable

The environmental activities herein are grouped by discipline. The tests described in this section are for Qualification/Acceptance unless specifically noted. No specific environmental test sequence is required, but the test program should be arranged in a way to best disclose problems and failures associated with the characteristics of the hardware and the mission objectives. It is strongly recommended that the vibration and acoustic noise tests precede the thermal-vacuum test unless there is an overriding reason to reverse that sequence. Table 2-1 contains a suggested test sequence for selected tests. The test sequence is included to illustrate a typical environmental test program. This sequence will also provide guidance to the program developer in the determination of environmental test program schedules and durations.

Environmental tests, such as fungus, sand, or rain, are not addressed in this document because in the majority of cases the protoflight hardware is maintained in a controlled environment. Should these tests be necessary, MIL-STD-810, "Environmental Test Methods and Engineering Guidelines," provides information on these tests.

TABLE 2-1 GENERAL ENVIRONMENTAL TEST SEQUENCE RECOMMENDATION FOR SELECTED TESTS

TEST	TEST SEQUENCE RECOMMENDATION[1]	
	COMPONENT	PAYLOAD
Functional[2]	1	1
Static Structural Test Modal Survey	2	2
Leakage	2, 5, 10	5, 7
Ordnance Shock	3	N/R
Vibration[3]	4	4
Acoustic[3]	4	4
Thermal Balance	N/R	8
Thermal Vacuum[5]	7	9
Burn-In/Run-In[4]	6	N/R
Proof Pressure	8	3, 6
Humidity	9	N/R
EMC	11	10

NOTES:

(1) Every test included in this document is not listed in this table. This is a general guide for test sequencing.

(2) Functional tests shall be conducted prior to and following each environmental test.

(3) Do not perform both vibration and acoustic noise at the payload level.

(4) Burn-In should precede Thermal Vacuum for electronic hardware.

(5) Includes Corona Testing / Evaluation

2.2 TEST METHODS

2.2.1 FUNCTIONAL TEST

A. Purpose

The functional test is a demonstration that the hardware and software meet their performance requirements when provided with appropriate stimuli.

B. Requirements

The initial functional test will serve as a baseline reference to determine if subsequent degradation of performance occurs. The functional test should demonstrate the operation of redundant circuitry. It should also demonstrate satisfactory performance in all operational modes within practical limits of cost, schedule, and environmental simulation capabilities.

A comprehensive functional test should be conducted on each hardware element upon completion of integration for all assemblies, components, subsystems and the payload. A functional test should be performed prior to and after each environmental test. Additional limited functional tests should be conducted during the transitions to the hot and cold extremes of the temperature and/or thermal-vacuum test, at the hot and cold extreme dwells of the temperature and/or thermal-vacuum testing, between each axis of vibration, and at the conclusion of the environmental test sequence as appropriate.

Hardware bonding must be checked and proven. MIL-B-5087 (ASG), "Military Specification Bonding, Electrical, and Lightning Protection, For Aerospace Systems," has bonding requirements.

C. Acceptance

Functional tests should be performed before and after each acceptance test to demonstrate hardware conformance to requirements.

2.2.2 STRUCTURAL AND MECHANICAL TESTS

A series of tests and analyses should be conducted to verify the design of the hardware, specified factors of safety, interface compatibility, workmanship, and compliance with associated STS systems safety requirements.

The program assumes that the hardware design is sufficiently modularized to permit realistic environmental exposures at the subsystem level. When that is not possible, or at the project's discretion, compliance with the subsystem requirements must in general be accomplished at the payload or component level of assembly. It is emphasized that each subsystem of the payload (structure, power, command and data handling, etc.) must be environmentally tested.

2.2.2.1 VIBRATION TESTS

A. Purpose

Vibration testing is performed to determine the resistance of equipment to expected vibrational stresses. The sinusoidal vibration sweep will verify the natural frequencies of the hardware. The random vibration test will detect material, workmanship, and design defects which could cause flight degradation or failure. The vibration test is fundamental in the environmental test program of components.

B. Requirements

JSC 07700, Volume XIV, "Space Shuttle Systems Payload Accommodations," Paragraph 4 (Shuttle Environment), provides the basic vibration levels expected. Component vibration test levels for various masses and equipment locations can also be derived from this source. The "Spacelab Payload Accommodation Handbook (SPAH)," SLP/2104, contains vibration test criteria for Spacelab hardware.

The test item is subjected to a low level sinusoidal sweep in each of three orthogonal axes prior to the random vibration tests. The test item shall be instrumented with response accelerometers to measure and verify the natural frequency response. Very often a modal survey is performed on the integrated hardware. The data acquired is used to verify the structural analysis already performed.

The test item is subjected to random vibration along each of three mutually perpendicular axes for 1 minute each. When possible, the component random vibration spectrum will be based on levels measured at the component mounting locations during previous subsystem or payload testing. When such measurements are not available, the levels will be based on statistically estimated responses of similar components on similar structures. Actual measurements will be used if and when they become available.

The protoflight component qualification/acceptance vibration test level shall be at the maximum expected flight level (97.5 probability level) fc e minute in each of the three vehicle axes.

For the purpose of controlling vibration, calibrated control accelerometer(s) should be attached rigidly on the test fixture near the fixture-test item interface. The control accelerometers should be aligned with the axis of applied vibration. Accelerometers used for control must be responsive over the necessary frequency range and output sufficient charge or voltage for signal processing.

Random vibration test excitation spectrums, when the component is mounted on the vibrator, measured by the control accelerometer(s), shall be equalized such that the overall RMS level is within ±10% of that specified. The power spectral density should be within +3, -1.5 dB (unless otherwise noted) of the specified levels everywhere in the frequency band when analyzed with a spectrum analyzer that has filter bandwidths which do not exceed 25 Hz below 1200 Hz or 100 Hz above 1200 Hz. Sinusoidal vibration test spectrums, when the component is mounted on the vibrator, measured by the control accelerometers, should be such that the overall amplitude is within +20%, -10% (unless otherwise noted) of the specified levels.

Before and after the sinusoidal and random vibration exposure, the test item should be examined and functionally tested. During the test, performance should be monitored as required to detect suspected failure modes such as contamination, poor connectors, cracked boards and cold soldering.

The test item should be attached to the test equipment by a rigid fixture. The mounting should simulate, insofar as practicable, the actual mounting of the item in the payload with particular attention given to duplicating the mounting contact area. In mating the test item to the fixture, a flight-type mounting (including vibration isolators, if part of the design) and fasteners should be used. Normally sealed items should be pressurized during test to their pre-launch pressure. Test articles should be in flight configuration. Film, samples, coolant, etc. should be installed or simulated before the start of vibration tests. In cases where significant changes in strength, stiffness, or applied load result from variations in internal and external pressure during the launch phase, a special test should be considered to cover those effects.

C. Acceptance Requirements

The vibration acceptance test levels should be 6 dB down (0.25 power level) from the qualification test levels and be applied for one minute in each of three orthogonal axes.

For reflight hardware, the amount of retest is dependent on the amount of rework performed on the hardware after completion of its previous flight. For such hardware, the basic objective of the structural/mechanical testing is to verify the integrity of the workmanship. If no disassembly and rework of a subsystem has taken place, the test may not be necessary. Subsystems that have been disassembled shall be subjected to an acoustic or vibration test after reassembly for workmanship purposes. Any component that has been disassembled should, after reassembly, be exposed to random vibration. As a minimum, the levels shall equal those necessary to demonstrate adequate workmanship. More comprehensive exposures on components shall be considered if the rework has been extensive.

2.2.2.2 ACOUSTIC TESTS

A. Purpose

The acoustic test demonstrates the ability of the protoflight integrated flight system and follow-on unit(s) to withstand or to operate in the acoustic vibration environment imposed during flight. Acoustic testing will be the recommended method of exciting the end item assembly rather than a random vibration test.

B. Requirements

JSC 07700, Volume XIV, "Space Shuttle Systems Payload Accommodations," Paragraph 4 (Shuttle Environment), provides the basic internal payload bay acoustic levels to be expected for the Shuttle payload. The "Spacelab Payload Accommodation Handbook (SPAH)," SLP/2104 contains acoustic test criteria for Spacelab hardware. The protoflight element shall be functionally tested and monitored in all of its operational modes (as permitted in the mechanical launch-operative configuration) during the acoustic test. Acoustic noise tests should cover the test spectrum at the maximum predicted sound pressure levels during flight. Test loads induced should not exceed the plastic deformation point (proportional limit) when testing the protoflight safety critical flight hardware. Exposure test time duration should be 1 minute. The measured overall spectrum level during the test should be within ± 2 dB of the specified value. The individual octave band and 1/3 octave band tolerance should be ± 2 dB unless otherwise specified.

Protoflight hardware should be subjected to an acoustic test in a reverberant sound pressure field. Table 2-2 presents the verification levels and durations for protoflight hardware. If acoustic protective devices are employed in the payload design, these protective devices should be part of the acoustic test. If testing of the devices is impractical, their attenuation characteristics may be used to adjust the levels of Table 2-2.

The acoustic test should be conducted in a reverberant chamber large enough to maintain a uniform sound field at all points surrounding the test item. The test should be kept within the specified tolerances. The sound pressure level is controlled at one-third octave band resolution. The preferred method of control is to average four or more microphones with a real-time device that effectively averages the sound pressure level in each filter band. When real-time averaging is not practicable, a survey of the chamber shall be performed to determine the single point that is most suitable for control of the acoustic test. Regardless of the control method employed, a minimum of four microphones should be positioned around the test chamber at sufficient distance from all surfaces to avoid absorption or re-radiation effects.

A distance from any surface of at least 1/4 the wavelength at the lowest frequency of interest is recommended. It is recognized that this is impossible to achieve in some facilities, particularly when noise levels are specified to frequencies as low as 25 Hz. In such cases, the microphones should be located in positions so as to be affected as little as possible by surface effects.

The preferred method of preparing for an acoustic test is to preshape the spectrum of the acoustic field with a dummy test item in the chamber. If no such item is readily available, it is possible to preshape the spectrum in an empty chamber. In that case, however, a low-level test should be performed after the test item has been placed in the chamber to permit final adjustments to the shape of the acoustic spectrum.

The boundary conditions under which the hardware is supported during test shall duplicate those expected during flight. When that is not feasible, the test item shall be mounted in the test chamber in such a manner as to be isolated from all energy inputs on a soft suspension system (natural frequency less than 20 Hz). The test item shall be a sufficient distance from chamber surfaces to minimize surface effects. During test, the test item should be in an operational configuration, both electrically and mechanically, representative of its configuration at liftoff.

C. Acceptance Requirements

The Acceptance test requirements for the acoustic noise test, if required, shall be conducted at 6 dB below the qualification levels for a duration of one minute.

TABLE 2-2

ACOUSTIC QUALIFICATION CRITERIA

One Third Octave Center Frequency (Hz)	Sound Pressure Level in dB re: 20 $\mu N/m^2$
20	118.0
25	120.0
31.5	122.0
40	124.0
50	125.5
63	127.0
80	128.0
100	128.5
125	129.0
160	129.5
200	129.5
250	129.0
315	128.5
400	127.5
500	125.0
630	123.0
800	121.0
1000	119.0
1250	117.5
1600	116.0
2000	114.0
2500	112.0
3150	110.5
4000	108.5
5000	106.5
6300	105.0
8000	103.0
10000	101.0
Overall SPL	139 dB
Protoflight Duration: One minute	

2.2.2.3 ORDNANCE SHOCK TEST

A. Purpose

The ordnance shock test demonstrates the capability of the component to withstand or, if appropriate, to operate in a dynamic pyrotechnic shock environment which is predicted to be imposed upon the component in flight.

B. Requirements

The principal source of self-induced shock occurs when pyrotechnic and pneumatic devices are actuated to release booms, solar arrays, etc. The impact of deployable devices as they reach their operational position at the "end of travel" is a likely source of significant shock.

The pyrotechnic shock environment at the component attachment points can be characterized as a decaying oscillatory transient containing frequencies from 100 to 10,000 Hz and which decays to a few percent of its maximum acceleration in 5 to 15 milliseconds. The component configuration should represent the electrical and mechanical operational modes appropriate to the flight shock during the pyrotechnic shock test. Acceleration should be measured in the three mutually perpendicular axes at the equipment attach points. Shock response spectra analyses with 5 percent damping (Q = 10) should be performed at 1/3 octave band center frequencies. The equipment undergoing test should be subjected to a sufficient number of suitable shocks to meet the specified test conditions at least once along each of three orthogonal axes. The component should operate without failure, malfunction, or out-of-tolerance conditions during the shock test and/or post-test checkout.

A suitable test shock for each direction of each axis is one that yields a response spectrum that equals or exceeds the required test spectrum over the specified frequency range. The spectra are to be determined for positive and negative maximum accelerations (either maximum absolute or equivalent static), Q = 10, and at least 1/3-octave frequency intervals. If the required test spectrum can be satisfied simultaneously in both directions along an axis, only one shock is required for that axis. If the requirement can only be satisfied in one direction, it is permissible to change the test setup and impose one additional shock to satisfy the spectrum requirement in the other direction; setup change possibilities are to reverse the polarity of the test shock time history or to reverse the test item orientation. It is permissible to simultaneously meet the test requirements along more than one axis with a single test shock. It is conceivable that a minimum of one test shock repetition will satisfy the requirements for both directions of all three orthogonal axes. At the other extreme, a total of six shocks will be required if each shock only satisfies the test requirements in one direction of one axis.An alternative test is to mount the test specimen in a representative structure in the actual location that will be used in service. The combination of pyrotechnic devices which will create the most severe environment shall each be fired one time for protoflight qualification/acceptance.

C. Acceptance Requirements

Ordnance shock tests are performed as Qualification/Acceptance tests. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements for ordnance shock tests.

2.2.2.4 MASS PROPERTIES

A. Purpose

The weights and mass center location of components and subsystems shall be determined as necessary for their subsequent integration into a complete protoflight spacecraft or experiment. In some cases, measurements of moments and products of inertia also may be required. Measurement tolerances require individual specification that is compatible with the purposes of the measurement. Measurement may include, but is not to be limited to the following:

- Data to meet mass property control requirements of the protoflight integrated flight system
- Input data for an analytical mass property estimate or tabulation of a larger assembly or complete spacecraft
- Mass property data for a spare unit that may be substituted in a spacecraft or experiment

B. Requirements

Hardware mass property requirements are mission-dependent. They are determined on a case-by-case basis. The mass properties of the hardware shall be verified by analysis and/or measurement. When mass properties are to be derived by analysis, it may be necessary to make some direct measurements of subsystems and components to attain the accuracy required for the mission and to ensure that analytical determination of payload mass properties is feasible. Determination of the mass properties of various subsystems should be sufficiently accurate that, when combined analytically to derive the mass properties of the payload, the uncertainties will be small enough to ensure compliance with payload mass property requirements. If analytic determination of payload mass properties is not feasible, then direct measurement is required. The following mass properties should be determined.

(a) Weight, Center of Gravity, and Moment of Inertia--Weight, center of gravity, and moment of inertia are used in predicting performance during launch, orbital operations, and return from orbit. The parameters are determined for all configurations necessary to evaluate flight performance in accordance with mission requirements.

(b) Balance--Hardware is balanced in accordance with mission requirements. Balance may be achieved analytically or by direct measurement.

- Procedure for Direct Measurement--The usual procedure for direct measurement is to perform an initial balance before initiation of the environmental verification program and a final balance after completion of the program.

One purpose of the initial balance is to ensure the feasibility of attaining the stipulated final balance. A residual unbalance of not more than four times the final balance requirement is the recommended objective of initial balance. Another reason for doing the initial balance prior to environmental exposures is to evaluate the method of attaching the balance weights and the effect of the weights on the operation of the hardware during the environmental exposures. Final balance is done after completion of all environmental testing in order to properly adjust for all changes to weight distribution.

- Maintaining Balance--It is recommended that changes to the hardware which may affect weight distribution be minimized after completion of the final balance effort. The effects of such changes (including any disassembly, hardware substitution, or other configuration change) on the residual unbalance of the hardware should be assessed. This involves sufficient dimensional measurement and mass properties determination to permit a judgment as to whether the configuration changes have caused the residual unbalance to exceed requirements. If so, additional balance operations may be necessary.

- Correcting Unbalance--To correct unbalance, weights may be attached, removed, or relocated. The amount of residual unbalance for all appropriate configurations is determined and recorded for comparison with the balance requirements. Balance operations include interface, fit, and alignment checks necessary to ensure that alignment of geometric axes is comparable with requirements.

  Balancing operations include measurement and tabulation of weights and mass center locations (referenced to hardware coordinates) of appendages, motors, and other elements that may not be assembled for balancing. The data is analyzed to determine unbalance contributed by such elements to each appropriate configuration.

  The facilities and procedures for balancing shall be fully defined at the time of initial balance, and sufficient exploratory balancing operations shall be performed to provide confidence that the final balance can be accomplished satisfactorily and expeditiously.

C. Acceptance Requirements

Mass properties are determined as part of the Qualification/Acceptance test phase. The Acceptance test requirements , if required, are the same as the Qualification/Acceptance test requirements for Mass Property tests.

2.2.2.5 PROOF PRESSURE TEST

A. Purpose

The proof pressure test will detect defects due to material and workmanship which could result in failure of items, such as pressure vessels, fittings, lines, and valves.

B. Requirements

The component should be subjected to a proof pressure test in the operational environment if possible. If it is not possible to perform the test in the operational environment, the test is performed, but the effects of the operational environment on the material and fracture control concerns are to be factored into the proof test pressure level. The proof pressure is dependent upon the use, type of component, material, and other factors. The proof pressure test factors are specified in MSFC-HDBK-505, "Structural Strength Program Requirements." The maximum design pressure is defined in NSTS 1700.7, "National Space Transportation System Safety Policy." Fracture control requirements are found in MSFC-HDBK-1453, "Fracture Control Program Requirements." A proof pressure cycle consists of raising the internal pressure (hydrostatically or pneumatically, as required) to the proof pressure and maintaining it for the required amount of time before decreasing to ambient pressure.

Valves should be hydrostatically or pneumatically pressurized to the proof pressure with the valve in both the open and closed positions. The required number of proof pressure cycles should be applied to the inlet port for the required amount of time as specified. Following the pressurization period, reduce the inlet pressure to ambient pressure. Pressure vessels, pressure lines, and fittings should be subjected to the required number of proof pressure cycles.

The requirement for successful Qualification/Acceptance is no detrimental distortion, leak, or rupture.

All flight units will be proof pressure tested.

C. Acceptance Requirements

The proof pressure test is performed as a Qualification/Acceptance test. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements for pressure tests.

2.2.2.6 LEAKAGE TEST

A. Purpose

The leakage test demonstrates the capability of the sealed components to meet the leakage rate requirements specified in the component specifications. This ascertains the capability of the sealed component to maintain a satisfactory seal in the environment which is to be imposed upon the component in test, transit, and flight.

B. Requirements

A leak check should be performed prior to any environmental test activity to provide baseline information. In addition to this initial leak test, the sealed component leakage test should be repeated before and after the vibration and/or acoustic test. The final leakage test may be accomplished as part of the thermal vacuum test.

Leak testing can be performed in a variety of different ways. Some of these methods of detecting and quantifying leaks in order of leak magnitude are as follows:

- Audible Leak Test -- Some leaks can be detected with the unaided ear. A variation of this method uses an audio amplifier to detect very small leaks.

- Bubble Leak Test -- Soap solution is used to detect small leaks in gas pressurized articles. This method can detect leaks with rates down to about $1X10^{-4}$ Standard Cubic Centimeters per Second (SCCS).

- Pressure Decay Test -- This test is performed by pressurizing the test article to a known pressure, maintaining the same temperature and recording the pressure drop over a known period of time. The test should be conducted under steady-state conditions, such as stable pumping, pressures, and temperatures. If time constraints do not permit the imposition of such conditions, a special test method should be devised.

- Mass Spectrometer Test -- This test is used to detect small leaks in the range of $1X10^{-1}$ to $1X10^{-9}$ SCCS. There are several ways to locate and quantify leaks with a mass spectrometer. The test article can be evacuated and connected to the leak detector while the surface of the article is probed with a small jet of helium. With this method it is possible to locate the leak and measure its size within the stated range.

  Another similar technique uses the evacuated test article, but it is covered with a plastic bag or similar item that contains helium. This method is faster in determining if a leak exists. It can quantify the leak rate but does not give the location of the leak.

A reverse technique pressurizes the test article with helium and either sniffs the surface with a tube from the leak detector, which locates but does not quantify, or uses a method that catches all the escaping helium in a container attached to the leak detector for quantifying but not pinpointing the leak location.

If dynamic seals are used, the item should be operated during the test. Otherwise, operation is not required.

C. Acceptance Requirements

Leakage tests are performed as a Qualification/Acceptance test. The final leakage test is generally performed as part of the thermal vacuum test. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements for leakage tests.

2.2.2.7 ASCENT/DESCENT PRESSURE PROFILE TEST

A. Purpose

The purpose of this test is to demonstrate the spacecraft's or experiment's response to short-term effects of pressure venting. This is the ability to withstand the mechanical stresses induced by the differential pressures loads developed across the end item surfaces and exposed surfaces of the individual components. The descent pressure test will ascertain the payload's or experiment's ability to withstand recompression stresses due to an emergency abort of the STS. This test may be performed at the subsystem or payload level of assembly.

B. Requirements

The launch configured protoflight unit should be used in the test because it provides for the simultaneous verification of the adequacy of the structure, enclosures, and all the components. The ascent/descent venting pressure profile should be applied one time to the protoflight unit.

The pressure profile is verified by a combination of analysis and test. The analysis should estimate the pressure differential induced by the nominal launch and re-entry trajectories across susceptible hardware. The test pressure profiles are determined from the generic ascent and descent pressure profiles for the orbiter found in JSC 07700,Volume XIV, "Space Shuttle System Payload Accommodations." The limit pressure profile is determined by using the predicted pressure-time profile for the nominal trajectory of the particular mission.

The pumping capacity of the selected facility should be capable of matching the desired pressure profile within + 5%, - 0% at all times. This is to account for uncertainties in the time that the maximum pressure differential occurs.

C. Acceptance Requirements

The pressure profile test, if required, is performed as a Qualification/Acceptance test. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements.

2.2.2.8 STATIC STRUCTURAL TEST

A. Purpose

The Static Structural Test determines if the structure has sufficient strength to withstand the application of flight loads.

B. Requirements

The strength qualification tests are usually performed as static tests. The loads should duplicate or envelope all flight loads and include all environmental effects (temperature, pressure, etc.).

When a structural test article is used, the static structural tests should be performed at the yield and ultimate levels specified by MSFC-HDBK-505, "Structural Strength Program Requirements."

With prior written approval from MSFC, testing of protoflight hardware may be conducted at levels lower than those required for static qualification tests. When the protoflight testing approach is used, every flight unit must be tested.

C. Acceptance Requirements

The Static Structural Acceptance test requirements are listed in MSFC-HDBK-505. There are no separate Acceptance test requirements if the protoflight Qualification/Acceptance approach is used.

2.2.3 VACUUM, THERMAL, AND HUMIDITY TESTS

A. General Information

The following capabilities should be demonstrated to satisfy requirements for the vacuum, thermal, and humidity areas:

- The hardware should perform satisfactorily in the vacuum and thermal environment of space.
- The thermal design and the thermal control system should maintain the affected hardware within the established mission thermal limits.
- The hardware should withstand, as required, the temperature and humidity conditions of transportation, storage, the orbiter cargo bay, and the orbiter manned spaces.

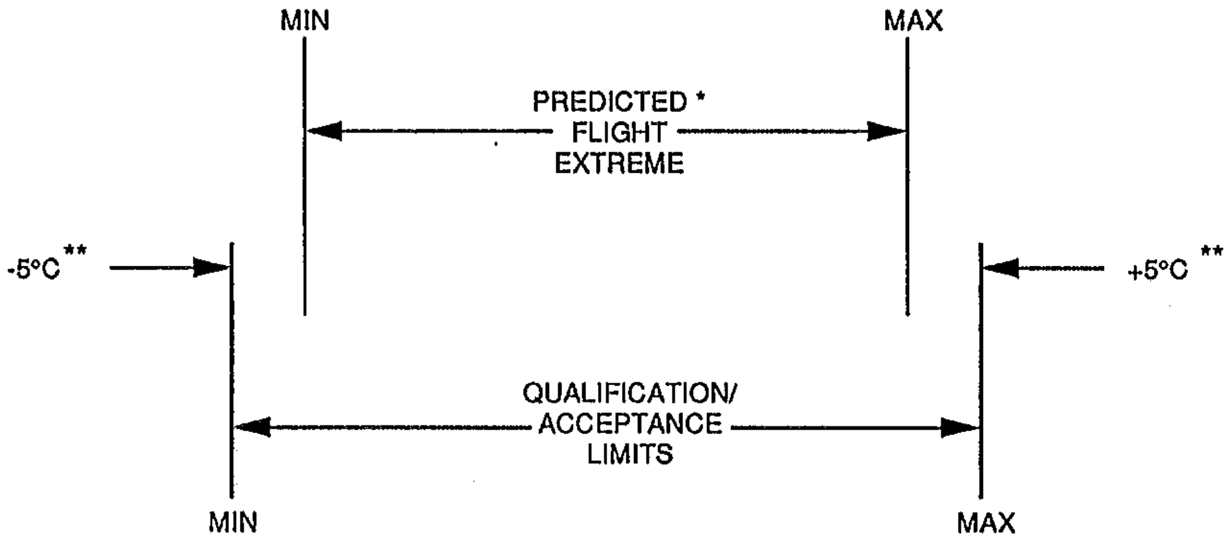
Figure 2-1 shows the protoflight thermal environment qualification/acceptance limits.

Table 2-3 summarizes the tests and analyses that collectively will serve to fulfill the general requirements of the various thermal tests. Tests noted in the table may require supporting analyses and vice versa. The order in which demonstrations are conducted shall be determined by the project and shall be specified.

Hardware mounted in pressurized compartments need not be verified for the vacuum environment, but this hardware must be tested for proper thermal performance. The tests to be performed are described below:

- THERMAL VACUUM -- This test is performed to verify that the hardware can operate satisfactorily in a vacuum environment at high and low temperature extremes.
- THERMAL CYCLING -- This test is performed to demonstrate operability and performance during extreme temperature exposures and transitions and to detect material and workmanship defects. This test should be run in conjunction with the thermal vacuum test on hardware that must operate in the vacuum environment.
- THERMAL BALANCE -- This test is performed to verify the analytical thermal model for validation of thermal design. It is performed in a vacuum environment. This test can be performed in conjunction with the thermal vacuum test, but increased instrumentation is required for verification of thermal balance.
- HUMIDITY -- These tests are performed on hardware that is not located in a controlled environment to verify performance after exposure to temperature and humidity conditions.

NOTE: The effects of temperature testing on the structure and structural strength should be considered prior to testing.



* APPROPRIATE MARGINS ARE INCLUDED IN DETERMINING THE MAXIMUM OR MINIMUM PREDICTED TEMPERATURES. THE MARGIN ACCOUNTS FOR UNCERTAINTIES IN PARAMETERS SUCH AS COMPLICATED VIEW FACTORS, SURFACE PROPERTIES, CONTAMINATION, RADIATION ENVIRONMENT, JOINT CONDUCTION, AND INADEQUATE GROUND SIMULATION. THE MARGINS MAYBE ZERO IF THE PREDICTIONS ARE CONSERVATIVE AND BASED ON WORST CASE CONDITIONS.

** A MARGIN OF 5°C IS SPECIFIED FOR THERMAL QUALIFICATION/ACCEPTANCE TESTS, IF IT DOES NOT EXCEED DESIGN LEVELS. THE 5°C ACCOUNTS FOR A 3°C REQUIRED MARGIN PLUS ± 2°C TEST CONDITION TOLERANCE IN THE -55°C TO +180°C RANGE.

NOTE: THERMAL TESTS COULD HAVE STRUCTURAL IMPLICATIONS THAT COULD CAUSE PROBLEMS IN OTHER AREAS. COORDINATION AMONG THE VARIOUS DISCIPLINES IS ESSENTIAL .

FIGURE 2-1

PROTOFLIGHT THERMAL QUALIFICATION/ ACCEPTANCE LIMITS

TABLE 2-3

VACUUM, THERMAL AND HUMIDITY REQUIREMENTS

REQUIREMENT	LEVEL OF ASSEMBLY	
	Subsystem or Component	Payload or Highest Practicable Level of Assembly
Thermal Vacuum(1)	T	T
Thermal Balance(1)	T/A	T/A
Temperature-Humidity (Manned Environment)(2)	T	N/A
Temperature-Humidity (Descent & Landing)(3)	A	A
Leakage	T	T
Thermal Cycling	T	N/A

NOTES: (1)Applies to hardware carried in the unpressurized cargo bay.

(2)Applies to hardware supporting payloads in the cargo bay.

(3)Applies to hardware that must retain a specified performance (e.g. reflectance or throughput) after return from orbit and is carried in the unpressurized cargo bay.

LEGEND: T - Test required.

A - Analysis required; test may be required to substantiate the analysis.

T/A - Test is highly desirable, however an analysis is mandatory.

N/A - Not Applicable

2.2.3.1 THERMAL VACUUM TEST

A. Purpose

All experiments and components are subjected to thermal-vacuum testing to demonstrate that they can operate satisfactorily at the nominal mission operating temperatures, at temperatures exceeding the extremes predicted for the mission, and during temperature transitions. The test should also demonstrate the ability of equipment to function satisfactorily after being driven to the extreme temperatures that might be encountered outside the range over which it is expected to function. Cold/Hot turn-ons should be demonstrated where applicable.

B. Requirements

An analysis should be made for each component and its application to determine whether the component is or is not susceptible to vacuum. Consideration should be given to conducting the thermal balance verification and thermal cycling in conjunction with the thermal-vacuum test program. A combined test is often technically and economically advantageous. A combined test must satisfy the requirements of both tests.

Care should be taken during the test to prevent environmental conditions that cause unrealistic failure modes. For instance, maximum rates of temperature changes shall not exceed acceptable limits. The limits are based on hardware characteristics or orbital predictions, whichever are more demanding.

Elements of a test item can be sensitive to contamination arising from test operation or from the test item itself. Materials selection in accordance with MSFC-HDBK-527, "Materials Selection List for Space Hardware Systems," should minimize this potential problem. If the test item contains sensitive elements, the test chamber should be examined to ensure that it is not a significant source of contamination. Particular care should be taken that potential contaminants emanating from the test item are not masked by contaminants from the chamber or the test equipment. Contamination sensitive hardware is subjected to a Thermal-Vacuum Bakeout Test in accordance with the requirements of MSFC-SPEC-1238, "Thermal Vacuum Bakeout Specification for Contamination Sensitive Hardware." The following documents contain useful information on materials test requirements and thermal vacuum procedures:

- MSFC-PROC-536,"Requirements and Procedures for Contamination Control Due to Vacuum Outgassing."
- MSFC-PROC-1301, "Guidelines for the Implementation of Required Materials Control Procedures."
- SP-R-0022, "General Specification Vacuum Stability Requirements of Polymeric Material for Spacecraft Application."

Transitions from cold to hot conditions increase contamination hazards because material that has accreted on the chamber walls may evaporate and deposit on the relatively cool test item. Transitions should be conducted at rates sufficiently slow (as determined by design) to prevent contamination from occurring. Testing should start with a cold soak and end with a hot soak. However, if for some reason the last exposure is a cold one, the test procedure should include a phase to warm the test item before the chamber is returned to ambient conditions so that the item will remain the warmest area in the test chamber, thus decreasing the likelihood of its contamination during that critical period.

Thermal criteria should be established to provide margins to compensate for uncertainties in the thermal parameters and to provide stress conditions to detect unsatisfactory performance that would not otherwise be uncovered prior to flight.

When a thermal balance test precedes the thermal vacuum test, results from that test should be used to update thermal analysis and refine the thermal vacuum test criteria.

The maximum and minimum temperatures to be imposed during the test shall be based on a survey of the predicted temperatures that each component would undergo during the mission. The temperature of the components should reach the required extremes during the test.

A temperature margin of no less than 5°C above the predicted mission maximum operating conditions and 5°C below the minimum operating conditions (and non-operating) should be used in establishing test temperatures. This temperature margin will help to ensure that the components reach the required temperature extremes without being overstressed. In cases where confidence in the predicted environment is high, this margin may be reduced accordingly.

The time duration for the four conditions of thermal stress; i.e., ambient, transient, cold, and hot; should be considered on a case by case basis. Thermal vacuum cycling (hot/cold stress conditions and number of transitions) will be program dependent. Temperature stabilization should be based on the temperature at a selected location or group of locations. These locations should be based on knowledge or analysis that ensure that components or critical parts of the system shall have achieved stabilization for the required time during the testing cycle. As an example, the temperature sensors should be mounted on the component base plate or the heat sink to which the component is mounted. Temperature stabilization is defined as when the rate of change of the test item temperature remains within the required tolerance of the test temperature for 1 hour with stable test chamber conditions. The temperature tolerance defining temperature stabilization is on the order of tenths or hundredths of a degree when performing a Thermal Balance Test (Paragraph 2.2.3.3) in conjunction with the Thermal Vacuum Test, .

Cycling between temperature extremes has the purpose of checking performance at other than stabilized conditions. Cycling causes temperature gradient shifts, thus inducing stresses intended to uncover incipient problems.

At the highest level of integration, hardware should be subjected to a minimum of four (4) thermal vacuum test cycles. Components and subsystems should be subjected to a minimum of eight (8) thermal-vacuum temperature cycles. During the cycling, the hardware should be operating and its performance monitored. Components should be subjected to one non-operational thermal vacuum test cycle. This cycle should be performed at the required temperatures and chamber pressure with the hardware non-operating. A functional check will verify hardware performance prior to the start of the thermal vacuum test. Another functional check will be performed after the thermal vacuum test is completed.

The total test duration should be sufficient to demonstrate performance and uncover early failures. The duration varies with the time spent at the temperature levels and with such factors as the number of mission-critical operating modes, the test item thermal inertia, and test facility characteristics. With the project's permission, the total test duration can be significantly reduced by turning off the test item's power during transitions to cold. The power should be left on for the first transition to cold to uncover any operating problems. Because of the length of time involved, it may be impractical to conduct a comprehensive functional test program during thermal-vacuum testing. A limited functional test may be substituted if satisfactory performance is demonstrated for the major mission critical modes of operation.

Hardware should be exposed to each temperature extreme until the components with the largest thermal mass reach temperature stabilization and the functional tests have been performed.

The chamber pressure after conducting the electrical discharge checks shall be less than 1.33×10^{-3} Pa (1×10^{-5} Torr).

Turn-on capability should be demonstrated under vacuum at least twice at each of the low and high temperatures. The ability to function through the voltage breakdown region should be demonstrated if applicable to mission requirements.

Spare components should undergo a test program in which the number of thermal cycles equals the sum of component-level and integrated-level test cycles. The durations of the tests at the upper and lower temperatures should equal to those of similar items intended for flight. Any redundant components in the test item should also be exercised during the test program by switching them on in the manner they would be selected on orbit. These redundant components would be given significant operating time during the test program, including cold and hot starts.

The test item configuration during ground tests should be as close as practical to the actual flight configuration, including simulation of all thermal interfaces. The components should be thermally coated during testing if thermal coating is used during flight. The mounting surfaces should have the same treatment as during flight. Critical temperatures should be monitored throughout the test and should be "alarmed" if possible. The operational modes of the payload should be monitored as required.

Items that are electrically operational during pressure transitions should undergo an electrical discharge check. This is to ensure that they will not be permanently damaged from electrical discharge during the ascent and early orbital phases of the mission or during descent and landing. The test should include checks for electrical discharge during the corresponding phases of the vacuum chamber operations.

An outgassing phase is included to permit a large portion of the volatiles to be removed from the test item. The outgassing phase is incorporated into a hot exposure that will occur during thermal vacuum testing. The test article is cycled from the cold temperature to the hot temperature at the beginning of vacuum testing and remains at this temperature until the contamination control monitors indicate that the outgassing has decreased to an acceptable level.

The presence of corona and arcing is determined on all suspect components where voltage levels are sufficient for this phenomena to occur during the reduction of pressure of the thermal-vacuum test chamber.

Test item start-up capability should be demonstrated to verify that the test item will turn on after exposure to extreme temperatures that may occur in orbit. For this check, the test item may be in either a commanded-off or undervoltage-recycle mode or high voltage mode.

The temperature controls should be adjusted to cause the test item to stabilize at the lower test temperature. Cold turn-on capability should be demonstrated as required. The demonstration may be conducted at the start of the cold condition. The duration of the cold phase should be sufficient to permit the performance of the functional tests after temperature stabilization occurs.

The test item should remain in an operational mode during the transitions between temperatures so that its functioning can be monitored under a changing environment. This requirement need not apply when turn-on of the test item is to be demonstrated after a particular transition.

The temperature controls should be adjusted to cause the test item to stabilize at the upper test temperature. Hot turn-on capability is to be demonstrated as required. The duration of this phase should be sufficient to permit the performance of the functional tests after temperature stabilization occurs.

If the mission includes a requirement for the test item to remain in an operational mode throughout the descent and landing phases, the test should include a segment to verify that capability.

Figure 2-2 illustrates the thermal vacuum profile and test cycle for the protoflight integrated test. It includes a corona test and subjects the test item to hot and cold extremes under vacuum conditions. The corona test will verify that the test item will not experience permanent damage due to electrical discharge during the ascent and early mission phases when pressures are decreasing from those prior to launch toward the final space vacuum levels.

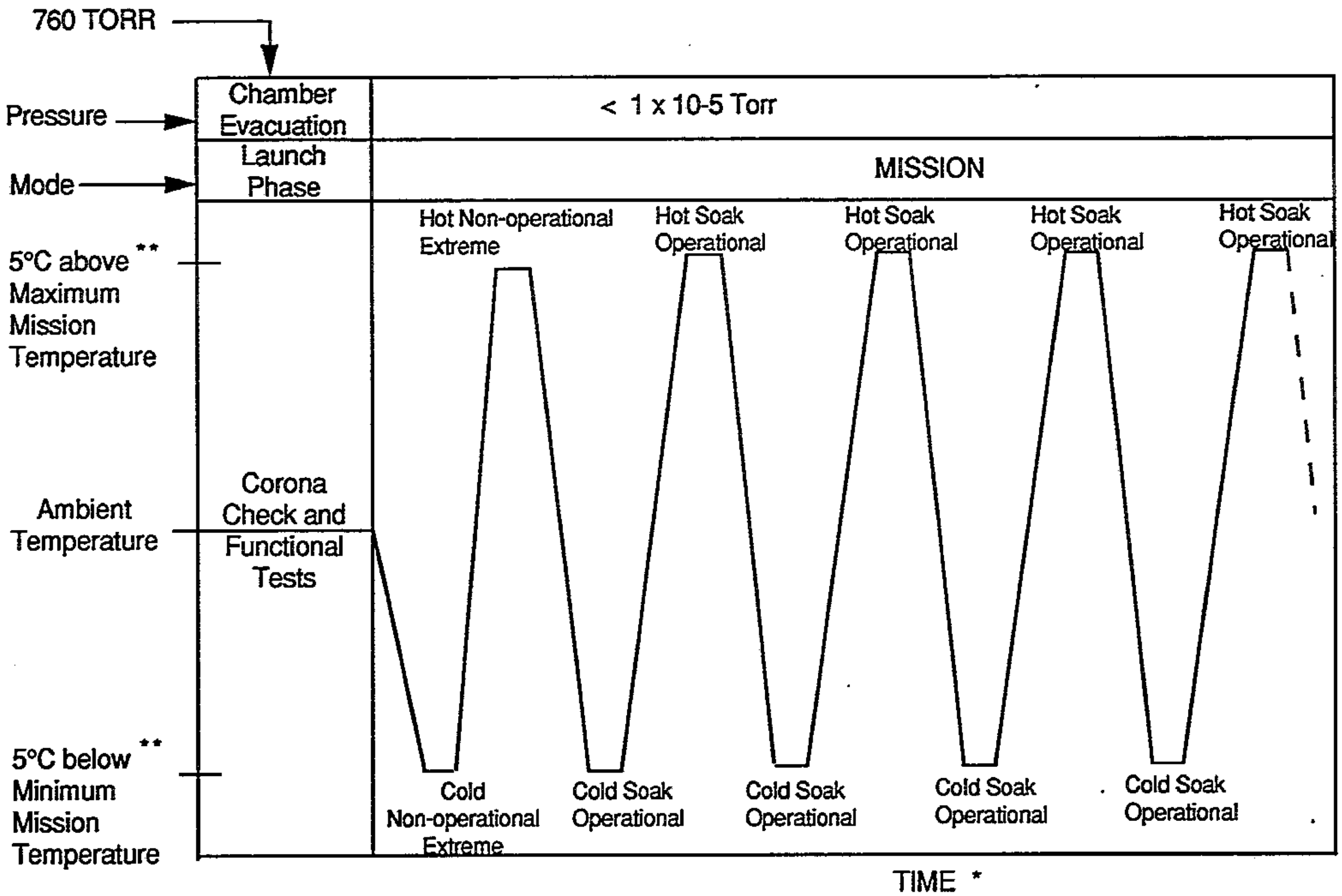
The integrated protoflight Qualification/Acceptance thermal-vacuum performance test should satisfy the following minimum requirements as depicted in Figure 2-2:

- Electrical discharge check.
- Hardware operation at the extreme cold temperature stress conditions.
- Hardware operation at the extreme hot temperature stress conditions.
- Four transitions between temperature extremes while operating.
- A hot or cold soak period, whichever is the case, is included at the four (4) transitions or dwell times. Soak time should be determined on case by case basis.

Special tests may be required to evaluate unique features, such as an attitude control subsystem or a radiation cooler, or to demonstrate the performance of external devices such as solar array hinges or experiment booms that are deployed at some time after the payload has attained orbit. The test configuration should reflect the conditions of flight operations as much as possible. When items undergoing test include unusual equipment, special care must be exercised to ensure that the equipment does not present a hazard to the test item or the facility or to personnel.

C. Acceptance Requirements

Thermal vacuum tests are performed as Qualification/Acceptance tests. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements.



* The time required for each temperature transition period (rate of change) will vary with the mission requirements, hardware, and test equipment.

** For items where environment definition has high confidence, the temperatures may be reduced accordingly.

Figure 2-2 Thermal Vacuum Test Cycle for Integrated Protoflight Hardware

2.2.3.2 THERMAL CYCLING TEST

A. Purpose

The Thermal Cycling Test is performed to detect latent material and workmanship defects. The test is completed prior to the installation of components into the next level of integration. Performance of the test at the component level allows any necessary changes or corrections to be made with the least impact on the program. Thermal cycling should always be performed if the hardware is exposed to a temperature gradient of 20°C.

B. Requirements

An analysis should be made for each component and its application to determine whether the component is or is not susceptible to vacuum. If the component is susceptible, the Thermal-Vacuum Test of Paragraph 2.2.3.1 is applicable. Components that are determined to be insensitive to vacuum may be temperature cycled in an air or nitrogen environment at ambient pressure. Component orientation with respect to the one "g" gravity field is such that the resultant naturally induced convection effects are accounted for or that thermally beneficial heat transfer effects are minimized.

The components that will be used for flight should be exposed to a minimum of eight (8) temperature cycles. The cycles may be performed at various stages of component integration. The temperature cycles are considered cumulative.

Thermal criteria should be established to provide margins to compensate for uncertainties in the thermal parameters and to provide stress conditions to detect unsatisfactory performance that would not otherwise be uncovered prior to flight. The maximum and minimum temperatures to be imposed during the test shall be based on a survey of the predicted temperatures that each component would undergo during the mission. The temperature of the components should reach the required extremes during the test.

A temperature margin of no less than 5°C above the predicted mission maximum operating conditions and 5°C below the minimum operating conditions (and non-operating) should be used in establishing test temperatures. This temperature margin will help to ensure that the components reach the required temperature extremes without being overstressed. In cases where confidence in the predicted environment is high this margin may be reduced accordingly.

The time duration for the four conditions of thermal stress; i.e., ambient, transient, cold, and hot; should be considered on a case by case basis. Thermal cycling (hot/cold stress conditions and number of transitions) is dependent on the planned mission as to the orbital cycling modes that simulate the sunshine and darkness cycles which comprise the hot and cold extremes. Temperature stabilization should be based on the temperature at a selected location or group of locations. These locations should be based on knowledge or analysis that ensures that components have achieved stabilization for the required time during the testing cycle. As an example, the temperature sensors should be mounted on the component base plate or the heat sink to which the component is mounted. Critical temperatures should be monitored.

Test item start-up capability should be demonstrated to verify that the test item will turn on after exposure to extreme temperatures that may occur in orbit. For this check, the test item may be in either a commanded-off or undervoltage-recycle mode or high voltage mode. Three cold starts and three hot starts of the test item should be conducted.

The temperature controls should be adjusted to cause the test item to stabilize at the lower test temperature. Components should be exposed to the temperature extreme until the component with the largest thermal mass reaches temperature stabilization. Cold turn-on capability should be demonstrated as required. The demonstration may be conducted at the start of the cold condition. The duration of the cold phase should be sufficient to permit the performance of the functional test after temperature stabilization occurs. The test item should remain in an operational mode during the transitions between temperatures so that its functioning can be monitored under a changing environment. This requirement need not apply when turn-on of the test item is to be demonstrated after a particular transition.

The temperature controls should be adjusted to cause the test item to stabilize at the upper test temperature. Components should be exposed to the temperature extreme until the component with the largest thermal mass reaches temperature stabilization. Hot turn-on capability is to be demonstrated as required. The duration of this phase should be sufficient to permit the performance of the functional tests after temperature stabilization occurs.

If the mission includes a requirement for the test item to remain in an operational mode through the descent and landing phases, the test should include a segment to verify that capability.

The total test duration should be sufficient to demonstrate performance and uncover early failures. The duration varies with the time spent at the temperature levels and with such factors as the number of mission-critical operating modes, the test item thermal inertia, and test facility characteristics. With the project's permission, the total test duration can be significantly reduced by turning off the test item's power during transitions to cold. The power should be left on for the first transition to cold to uncover any operating problems. Because of the length of time involved, it may be impractical to conduct a comprehensive functional test program during thermal-vacuum testing. A limited functional test may be substituted if satisfactory performance is demonstrated for the major mission critical modes of operation. The test item configuration during ground tests should be as close as practical to the actual flight configuration, including simulation of all thermal interfaces. The components should be thermally coated during testing if thermal coating is used during flight. The mounting surfaces should have the same treatment as during flight.

NOTE: The effects of thermal testing and the durations of the thermal tests on the structure and the structural strength and fatigue life should be considered prior to any testing.

C. Acceptance Requirements

Thermal cycling tests are performed as Qualification/Acceptance tests. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements.

2.2.3.3 THERMAL BALANCE TEST

A. Purpose

The thermal balance test is performed to verify the protoflight end item analytical thermal model. It demonstrates the ability to meet design requirements under vacuum conditions and temperature extremes. The test demonstrates the ability of the thermal control system to maintain components, subsystems, and the entire integrated flight system within the design temperature range plus a design margin of safety.

Analytical thermal models are developed of the payload, its elements, and the mission environment for the purpose of predicting the thermal performance of the payload and its elements during the mission. The models can also be used to predict the thermal performance of the payload or its elements in a known test chamber environment. Correlation of the results of the chamber thermal balance tests with predictions based on the analytical model provide a means for validating the thermal design and for improving model accuracy in predicting performance during the mission. At the same time, a thermal balance test can provide the basis for evaluating the performance of the as-built thermal control system. Predictions based on the analytical thermal model should be compatible with the test conditions.

B. Requirements

It is preferable to conduct a thermal balance test at the integrated level of assembly. Alternative methods may be used to verify thermal design if necessary. These alternatives are:

- Test at lower levels of assembly, incorporate the results into the payload analytical thermal model, and compare the results with predictions of the analytical thermal model.

- Test a thermally similar physical representation of the flight payload, in other words, a physical thermal model. Compare the results with predictions based on the payload analytical thermal model.

A decision must be made as to the method used to simulate thermal inputs. The type of simulation to be used is generally obvious due to the chamber size, solar beam size, and payload size. In determining the method of simulation to be used one should always try to choose the simulation type that requires the minimum number of assumptions and calculations to determine the environmental inputs. Listed below are some methods of simulation and major assumptions needed to have a successful test.

- Solar Simulation and Planetary Irradiation -- The major requirement is to understand the effective absorptivity and emissivity based on the source used to simulate the sun and planet. The change in effective absorptivity can be quite large; the emissivity change is quite small.

- Skin Heaters -- The absorbed energy from all sources must be driven into the skin using heaters. The absorptivity and inputs from other sources must be calculated. For simple shaped spacecraft this is an excellent test.

- Heater Plates -- The information in the Skin Heaters test is needed plus the exchange factor between the heater plates and spacecraft. This can be an acceptable test if one is not allowed to touch the spacecraft outer skins.

  The internal power dissipated in a spacecraft or subsystem should be measured to an accuracy of 1%, if possible. Prior to the test the power consumption and line losses of the individual components should be measured whenever possible.

- Assumptions -- Extraneous effects such as gaseous conduction in residual atmosphere should be kept negligible by vacuum conditions in the chamber. Pressures below 1.33 x 10^{-3} Pa (1 x 10^{-5} Torr) are usually sufficiently low. Care should be taken to prevent conditions such as test-configuration-induced contamination that cause an unrealistic degradation of the test item. Devices to monitor contamination should be included as necessary.

  The number of energy balance conditions simulated during the test should be sufficient to verify the thermal design. The duration of the thermal balance test depends on the mission, payload design, payload operating modes, and times to reach stabilization. The exposures should be long enough for the payload to reach stabilization so that temperature distributions in the steady-state conditions may be verified.

  The protoflight unit Qualification/Acceptance thermal test will be performed at the acceptance level. The temperature conditions for the subsystems acceptance are +5°C above the maximum predicted flight extreme and -5°C below the minimum predicted flight extreme. The vacuum level should be less than 1.33 x 10^{-3} Pa (1 x 10^{-5} Torr). If an area is controlled by an active thermal control subsystem that has been previously qualified, then the temperature conditions may be limited accordingly.

  Allowable differences between predicted and measured temperatures are established by the cognizant thermal analyst. Verification of the thermal design is considered accomplished if the delta between the analytical and measured temperature is no greater than 5°C.

C. Acceptance Requirements

The thermal balance test is performed as a Qualification/Acceptance test. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements for thermal balance tests.

2.2.3.4 TEMPERATURE-HUMIDITY TESTS

2.2.3.4.1 MANNED ENVIRONMENT

A. Purpose

The test applies to payloads that are to be located in manned spaces of the STS and to equipment located in manned spaces for the control or support of payloads located in the unpressurized cargo bay.

B. Requirements

If the environment is such that condensation can occur, tests should be conducted to demonstrate that the hardware can function under the severest conditions that can be expected. Temperature cycling duration, performance, and other requirements, except for vacuum, should be as noted in the Thermal-Cycling paragraph, 2.2.3.2, of this document.

The hardware should be tested at temperature and relative humidity conditions at least 10°C and 10 % RH beyond the limits expected during the mission. The upper humidity conditions, however, should not exceed 95% RH unless condensation can occur during the mission. If that condition is possible, tests should be conducted to demonstrate that the hardware can function properly after such exposure.

C. Acceptance Requirements

The Manned Spaces Temperature-Humidity test is performed as a Qualification/Acceptance test. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements for temperature-humidity tests of manned environments.

2.2.3.4.2 DESCENT AND LANDING

A. Purpose

This test demonstrates the ability of the hardware in unpressurized areas of the cargo bay to survive the temperature and humidity conditions expected during descent and landing without unacceptable degradation. This test applies to hardware that must return from orbit with a specified performance capability.

B. Requirements

Because the survival requirement usually applies only to a portion of the test item, tests may be conducted on that portion before it is integrated with the payload, thus protecting other portions of the payload from the potentially degrading environments.

If an item must survive the descent and landing environment but the test would make it unflightworthy, then the test should not be performed on the flight item. Instead, an analysis based on tests of engineering or prototype models, or other such evidence, should be used to verify that the test item can satisfactorily withstand the return from orbit.

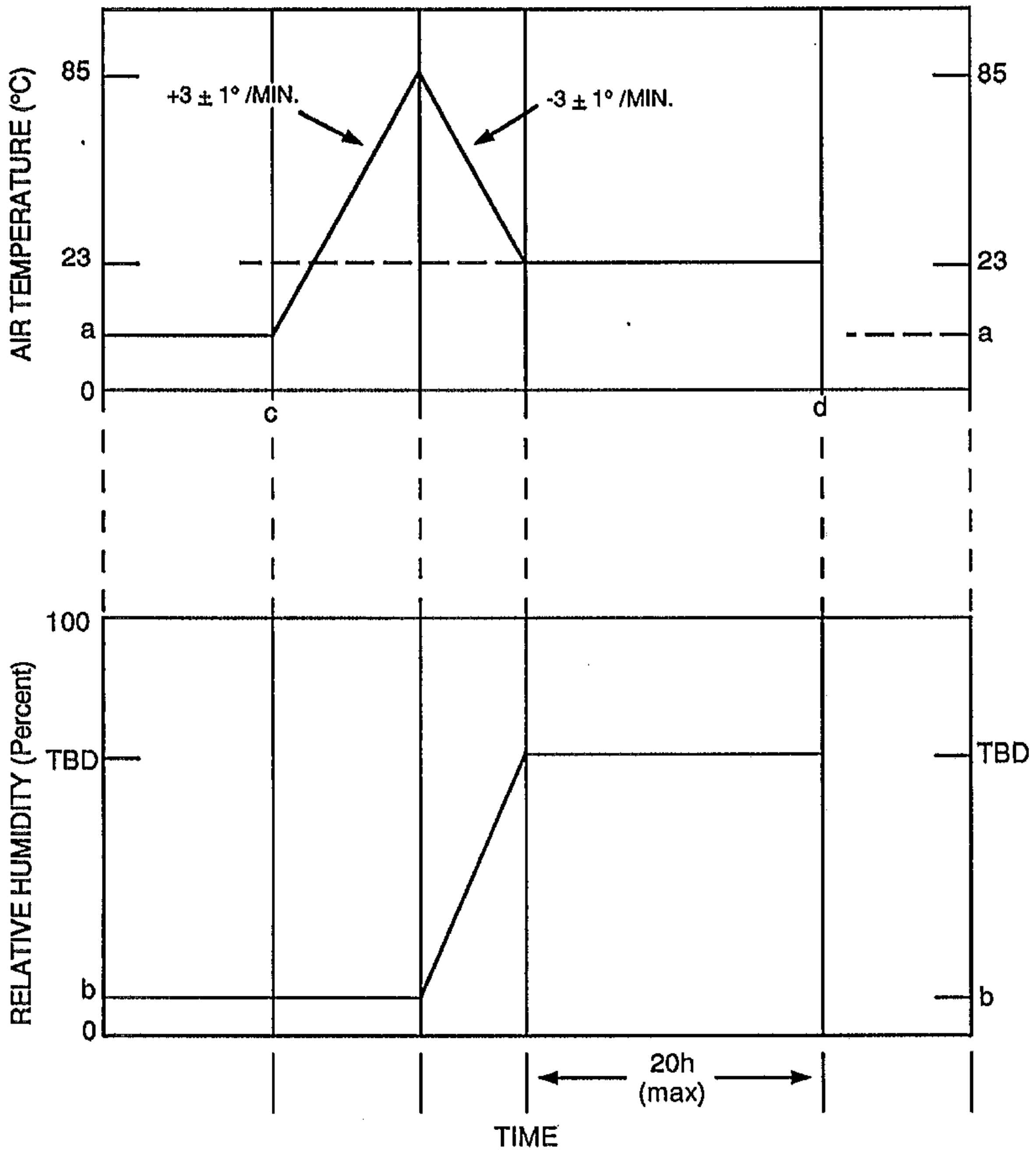
The test item should be placed in a temperature-humidity chamber and a functional performance test should be performed before the item is exposed to the test environment. If a functional performance test was conducted as part of the post-test checkout of the preceding test, those test results may be sufficient.

The temperature and humidity profiles of Figure 2-3 should be followed for the test with the payload in a configuration appropriate for the descent and landing phase.

Electrical function tests should be conducted after the test exposure to determine whether or not acceptable limits of degradation have been exceeded.

C. Acceptance Requirements

The Descent and Landing Temperature-Humidity test is performed as a Qualification/Acceptance test. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements for descent and landing temperature-humidity tests.



LEGEND:

a- Temperature of Payload at Deorbit
b- Minimum Chamber Relative Humidity
c- Payload Temperature Stabilized

FIGURE 2-3
TEMPERATURE-HUMIDITY PROFILE FOR DESCENT AND LANDING DEMONSTRATION

2.2.4 ELECTROMAGNETIC COMPATIBILITY REQUIREMENTS TESTS

A. Purpose

The Electromagnetic Compatibility (EMC) tests ensure that the hardware does not generate Electromagnetic Interference (EMI) that will adversely affect the safety and operation of the STS or the operation of other equipment. These tests also ensure that the hardware is not adversely affected by EMI generated by the STS, other equipment, or by the hardware itself.

B. Requirements

A specific group of EMC requirements are imposed by Johnson Space Center (JSC) on hardware that operates on orbiter power or that operates on its own power within or near the orbiter. Those requirements are defined in the ICD-2-19001, "Shuttle Orbiter/Cargo Standard Interfaces," document.

MSFC-SPEC-521, "Electromagnetic Compatibility (EMC) Requirements on Spacelab Payload Equipment," contains the minimum EMC requirements to be imposed on protoflight hardware. The requirements of MSFC-SPEC-521 are derived from ICD-2-19001 and the SPAH, as well as other documentation. Guidelines are specified in MSFC-SPEC-521 to ensure that hardware does not generate EMI that would affect the operation of its own subsystems or the operation of other equipment. Guidelines are also provided to ensure that hardware will not be susceptible to EMI from its own subsystems, other equipment, or the orbiter. Those guidelines include an assurance that the hardware can operate satisfactorily within the environments usually encountered during integration and ground testing. However, some hardware may have particularly sensitive sensors and electrical devices that are inherently susceptible to the low-level EMI that is usually found in those ground environments. Special considerations and procedures must be developed to meet individual needs.

Table 2-4 is a matrix of EMC tests that apply to a wide range of hardware. Tests are prescribed at the component and subsystem levels of assembly. The project must select the requirements that fit the characteristics of the particular mission and hardware.

The tests are based on the requirements of MIL-STD-461A, "Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference," and MIL-STD-462, "Test Methods for Electromagnetic Emission and Susceptibility," as amended by Notice 2, and MIL-STD-463A, "Electromagnetic Interference Characteristics, Definition and Systems of Units." The latest revisions of MIL-STD-461, MIL-STD-462 and MIL-STD-463 may be used if the test requirements meet or exceed the EMC test requirements of ICD 2-19001. It is recommended the document of latest revision be used whenever possible. MSFC-SPEC-521 contains the appropriate portions of the military standards to use in protoflight hardware testing. The tests and their limits are to be considered minimum requirements; however, they may be revised as appropriate for a particular payload or mission if project approval is obtained. More stringent requirements may be necessary, as for example for hardware with very sensitive electric field or magnetic field measurements. The tests and their limits should be documented.

TABLE 2-4

EMC REQUIREMENTS PER LEVEL OF ASSEMBLY

	Type	Test	Component	Subsystem
Emissions	CE	DC Power Leads	Sb, Rb, Rf	Sb, Rb, Rf
	CE	AC Power Leads	Sb, Rb	Sb, Rb
	CE	Spikes on DC Power Lines	Sb	Sb
	CE	Transient Spikes on AC Power Lines	Sb	Sb
	CE	Antenna Terminals	Rf	
	RE	AC Magnetic-Field	-	Sb
	RE	AC Magnetic-Field	Rb, Rf	Rb, Rf
	RE	E-Fields	Rb, Rf	Rb, Rf
	RE	Payload Transmitters	-	Rb
	RE	Spurious	-	Rb, Rf
Susceptibility	CS	Power Lines	Rb, Rf	Rb, Rf
	CS	Intermodulation Products	Rb, Rf	Rf
	CS	Signal Rejection	Rb, Rf	Rf
	CS	Cross Modulation	Rb, Rf	Rf
	CS	Power Line Transients	Rb, Rf	Rb, Rf
	RS	E-Field (General Compatibility)	Rb, Rf	Rb, Rf
	RS	Compatibility with Orbiter's Transmitters	-	Rb, Rf
	RS	E-Field (Operational Compatibility)	-	Rf
	RS	Magnetic-Field Susceptibility	Rf, Rb	Rf, Rb
		Magnetic Properties	Rf	Rf

LEGEND:
- Sb - All Items required by Interface Control Document (ICD)
- Rb - Test to insure no unintentional generation of EMI
- Rf - Test to insure immunity to susceptibility of anticipated electromagnetic environment
- CE - Conducted Emission
- CS - Conducted Susceptibility
- RE - Radiated Emission
- RS - Radiated Susceptibility

2.2.4.2.2 Radiated Emission (RE) Limits

Radiated emission limits and requirements should be applied to hardware as defined below:

- Radiated AC magnetic field levels produced by all hardware at distances of 1 meter should be in accordance with requirements.

2.2.4.1 Safety and Controls

During prelaunch and pre-release checkout, sensitive detectors and hardware may require special procedures to protect them from the damage of high-level radiated emissions. If such procedures are required, they should also be applied during EMC testing. Operational control procedures should also be instituted for EMC testing during pre-release checkout to minimize interference with the orbiter and other payloads.

Except for bridgewires, live electroexplosive devices (EEDs) used to initiate spacecraft functions, such as boom and antenna deployment should be replaced by inert EEDs. When that is not possible, special safety precautions should be taken to ensure the safety of the hardware and its operating personnel.

Spurious signals that lie above specified testing limits should be eliminated. Spurious signals that are below specified limits should be analyzed. If a subsequent change in frequency or amplitude is possible, the spurious signals should be eliminated to protect the hardware and instruments from the possibility of interference. Retest should be performed to verify that intended solutions are effective.

2.2.4.2 Emission Requirements Tests

2.2.4.2.1 Conducted Emission (CE) Limits

Conducted emission limits and requirements on power leads, as well as on antenna terminals, should be applied to hardware as defined below. The requirements do not apply to intrasystem or secondary power leads to sub-units unless they are specifically included in a hardware specification.

- Narrowband conducted emissions on DC and AC power, and power-return leads should be limited to the levels specified for payloads.

- Broadband conducted emissions on DC and AC power, and power-return leads should be limited to the levels specified for payloads.

- Spikes produced by payloads on DC power lines from the orbiter to the hardware because of switching and other operations should not exceed the defined limits. The requirement applies to turn-on and turn-off and switching through all operational modes. The impedance of the test power source used to simulate the orbiter power system should be defined.

- Transient spikes produced by hardware on AC power lines from the orbiter to the payloads should not exceed the defined limits.

- Conducted emissions on the antenna terminals of hardware receivers and transmitters in key-up modes should not exceed narrowband and broadband emission requirements. Harmonic and all other spurious emissions from transmitters in the key-down mode should not exceed the defined limits. The test is conducted on receivers and transmitters before they are integrated with their antenna systems.

- Radiated AC magnetic field levels produced by free flyers and their subsystems should be limited. This requirement may be deleted with project approval for free flyers that do not include subsystems or instruments that are inherently susceptible to low level, low frequency AC magnetic fields, however, the requirements above still apply. If the detectors or devices have high sensitivities to magnetic fields, more stringent limits on magnetic field emission may be required.

- Radiated narrowband and broadband electric field levels produced by hardware should not exceed the specified levels. This requirement may be impacted by specific project requirements. This may necessitate the use of a low noise pre-amplifier in the test setup prior to the EMI receiver.

- Radiated spurious and harmonic emissions from payload transmitter antennas should not exceed the limits for those of conducted emissions on antenna terminals.

2.2.4.3 Susceptibility Requirements Tests

ICD-2-19001 defines the worst-case levels of shuttle-produced emissions in the payload bay. The levels have been incorporated into the following susceptibility requirements for individual subsystems and components.

2.2.4.3.1 Conducted Susceptibility (CS) Requirements

- DC Power Bus Ripple and CS tests should be performed.

- A transient signal should be applied to power lines. Transients of various amplitudes and pulse widths can occur on primary power lines.

2.2.4.3.2 Radiated Susceptibility (RS) Requirements

The following tests should be applied to individual components and subsystems:

- The hardware should be exposed to external electromagnetic signals in accordance with the RS requirements. Intentional E-field sensors on hardware that operate within the frequency range of the test should be removed or disabled without otherwise disabling the payload during the test. The RS requirements of the referenced documentation are tailored to meet the requirements for STS payloads. The test should demonstrate that hardware (exclusive of E-field sensors) can meet their performance objectives.

- Hardware that could be susceptible to the magnetic fields generated by the STS should be tested for susceptibility in a suitable magnetic test facility.

C. Acceptance Requirements

EMC tests are performed as Qualification/Acceptance tests. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements for EMC tests.

2.2.4.4 ELECTROSTATIC DISCHARGE TEST

A. Purpose

The electrostatic discharge (ESD) test verifies the spacecraft's immunity to electrostatic buildup and its subsequent discharge to components and circuitry.

B. Requirements

Electrostatic discharge tests should be performed on end items and circuits which are located in close proximity to electrical insulating materials on the outer envelope of the protoflight integrated flight system. The tests should cover both radiated and conducted effects of ESD.

Electronic assemblies should not be damaged by electrostatic discharge of less than 2000 volts. Electronic components should not be damaged by electrostatic discharge of less than 4000 volts. These voltages represent charges that may be discharged from personnel during equipment installation or from astronauts during on-orbit maintenance. Electrostatic characteristics of a typical human may be simulated by a 100 pF capacitor charged with the voltages stated above and discharged through a 1500 ohm resistor to the case or pin in question. The requirements may be found in NHB 5300.4 (3X), "Requirements for Electrostatic Discharge Control (Excluding Electronically Initiated Explosive Devices)."

The test setup should simulate the operational wiring and grounding scheme. The equipment under test should be bonded to the ground plane by the same method used for the vehicle installation. For synchronous orbits, a pulsed discharge, at a pulse rate of 1 per second for a period of 30 seconds, should be established at a level of 10 kilovolts and at a distance of 30 cm (12 inches) from each exposed face of the test sample. The test should be repeated by one of the following two methods:

- Using a direct discharge from one test electrode to each top corner of the test sample for equipment exposed to the direct space environment, such as samples external to a shielded space vehicle.
- Impressing the series current from the arc-discharge through the mounting surfaces of the test sample for equipment installed within a shielded space vehicle.

If test sample failures are observed at 10 kilovolts, the voltage level should be decreased to the point where satisfactory equipment operation is obtained, and that level noted. For orbits other than geosynchronous, the applicable test voltage should be determined as part of the system evaluation. MIL-STD-1541, "Electromagnetic Compatibility Requirements for Space Systems," describes a circuit capable of performing ESD tests. Flight configurations containing solar arrays, together with the circuits entering from the flight vehicle exterior, may require special tests to determine the adequacy of the design.

C. Acceptance Requirements

Electrostatic Discharge tests are performed as Qualification/Acceptance tests. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements for electrostatic discharge tests.

2.2.4.5 CORONA TEST

A. Purpose

Corona-arcing tests are considered an adjunct to the thermal-vacuum test. The presence of corona and arcing is to be determined on all suspect components where voltage levels are sufficient for this phenomena to occur during the reduction of pressure to the thermal-vacuum chamber.

B. Requirements

Equipment that will not operate during launch or otherwise be subject to corona damage should have electrical power applied only after the test pressure level has been reached. During the vacuum chamber evacuation from ambient pressure to 1.0×10^{-5} Torr (1.33×10^{-3} Pa) or less, components that are operational during vehicle ascent should be powered and operating. Critical parameters should be monitored on the powered components. During the reduction of pressure, corona and arcing should not be permitted.

The presence of corona may be determined using several different methods. MSFC-STD-531,"High Voltage Design Criteria," contains basic corona test philosophy and requirements.

Thermal bakeout in accordance with MSFC-SPEC-1238, "Thermal Vacuum Bakeout Specification for Contamination Sensitive Hardware," of sensitive items should help reduce corona-arcing phenomena.

C. Acceptance Requirements

Corona tests are performed as Qualification/Acceptance tests. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance test requirements for corona tests.

2.2.4.6 MAGNETIC PROPERTIES TEST

A. Purpose

A spacecraft whose magnetic properties could affect its own operation, including its complement of instruments, the magnetic field of which must be controlled, should be tested at the component, subsystem, and at the spacecraft levels of assembly, as appropriate.

B. Requirements

NOTE: Spacecraft with magnetic sensors may have more stringent requirements than contained herein.

Initial Perm Test -- The maximum DC magnetic fields produced by a spacecraft and by each of its components following manufacture should not exceed 3.0 and 0.2 Am^2 (dipole moment), respectively. These values are applicable at a distance of approximately three times the maximum linear dimension of the item under test.

Perm Levels After Exposure to Magnetic Field -- The maximum magnetic field produced by a spacecraft and each of its components after exposures to magnetic field test levels of 15×10^{-4} tesla should not exceed 5.0 and 0.3 Am^2, respectively.

Perm Levels After Exposure to Deperm Test -- The maximum magnetic fields produced by a spacecraft and each of its components after exposures to magnetic field deperm levels of 30×10^{-4} tesla for spacecraft and 50×10^{-4} tesla for components should not exceed 2.0 and 0.1 Am^2, respectively.

Induced Magnetic Field Measurement -- To obtain information for spacecraft magnetic design and testing, the induced magnetic field of components should be measured while the components are turned off and exposed to a magnetic field test level of 0.6×10^{-4} tesla. The measurement should be made by a test magnetometer that can null the magnetic test field.

Stray Magnetic Field Measurement -- A spacecraft and each of its components should not produce magnetic fields due to internal current flows in excess of 0.5 and 0.05 Am^2, respectively.

Subsystems should also be tested in accordance with these requirements. The requirements are determined on a case by case basis. Subsystem limits should be designated such that the fully integrated spacecraft can meet its magnetic requirements.

C. Acceptance Requirements

The magnetic properties tests are performed as Qualification/Acceptance tests. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance level test requirements for magnetic properties testing.

2.2.5 BURN-IN/RUN-IN TEST

A. Purpose

The intent of the burn-in test is to detect latent material and workmanship defects which occur early in the component use prior to installation of the components into the make-up of the protoflight end item, follow-on unit(s), and experiments.

The purpose of the run-in test is to condition moving parts and interfaces and to establish representative performance characteristics. The actual operation of mechanical or electromechanical components may provide different characteristics than anticipated.

B. Requirements

Burn-In

The specified input electrical parameters should be applied to the component for the required number of hours or cycles. Electronic equipment is cycled on/off during burn-in. Realistic duty cycles for all devices should be included in the test requirements. Devices with specific duty cycles or that pulse continuously should be tested in a manner which allows for duty cycle uniqueness but still provides adequate burn-in.

Both electrical and electronic components should have 300 hours of operation prior to completion of the qualification/acceptance testing. The last 100 hours should be failure free.

The test run time for functional testing and thermal vacuum temperature cycling may be included as part of the 300 hour burn-in test.

Run-In

The run-in test is performed to condition moving parts and interfaces. It is important to establish known functional performance so there are no problems later in the program. Some of the areas where this test is necessary are lubricant distribution characteristics, the actual seating of motor brushes, and bearing performance. The run-in test should be performed at actual operating requirements so that representative operating characteristics of the components may be noted.

The actual test parameters are determined on a case by case basis.

C. Acceptance Requirements

The Burn-In/Run-In test is performed as a Qualification/Acceptance test. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance level test requirements for burn-in/run-in tests.

2.2.6 HIGH/LOW VOLTAGE TEST

A. Purpose

The power subsystem test should assure that steady state and transient compatibility exists between the power sources and the power users.

B. Requirements

(a) Steady State

As illustrated in Figure 2-4, the high/low voltage qualification and acceptance criteria for a 28 volt system are as follows:

- High and low voltage steady state limits should be defined as acceptance criteria for the power sources.
- A maximum voltage drop should be defined as acceptance criteria for the power distribution network between the power sources and the power users.
- The standard voltage requirements are 28 V ± 4 volts (including cable drop).
- The lowest possible operating voltage that a power user operating at 28 volts nominal would expect to operate at is 23.75 volts.
- The highest possible operating voltage that a power user operating at 28 volts nominal would expect to operate at is 32.25 volts.
- The hardware should not be damaged by voltage inputs up to 38.0 volts for a duration of up to 50 milliseconds. The voltage input and time duration will be dependent on the characteristics of the power source.
- The hardware should not be damaged by voltage inputs as low as 20.0 volts. The hardware is not required to operate at this low voltage.

NOTE: For power sources providing voltages other than 28 ± 4 volts, the tests would have to be modified to properly test the operability and survivability of the hardware.

(b) Transients

MSFC-SPEC-520, "Electrical Transients, Systems and Components," should be considered in the design of equipment using bus power and used during the acceptance test of the protoflight and follow-on unit(s). The specification presents the control and test requirements to be imposed for limiting system and subsystem power bus transient generation and methods to verify immunity to specified transient levels on input power leads.

C. Acceptance Requirements

The integrated systems acceptance tests shall demonstrate operability with the power source outputs set at the high and low voltage steady state limits.

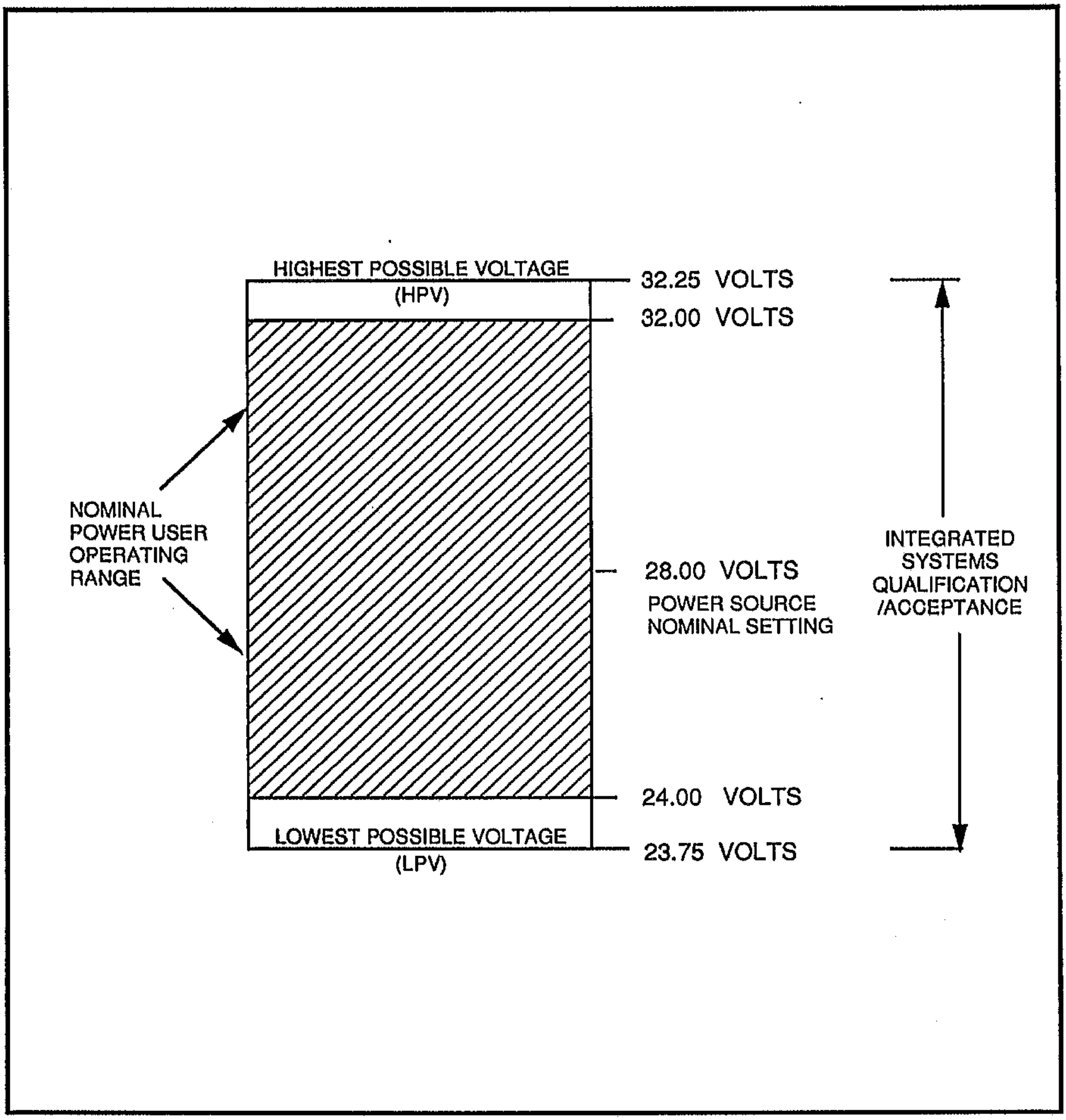


FIGURE 2-4
PROTOFLIGHT STEADY STATE VOLTAGE QUALIFICATION
AND ACCEPTANCE CRITERIA FOR A 28 VOLT SYSTEM

2.2.7 PLASMA TEST

A. Purpose

Since the densities of charged particles (O+, H+, NO+, and electrons) of thermal energies in the orbital environment plus the ions relative electron volt energies can initiate breakdown in the high voltage or other sensitive circuits and devices, a plasma test should be conducted to subject the components to a simulation of the charged particle environment. This test should be performed on components which have an elevated potential in relation to plasma and would be susceptible to damage.

B. Requirements

Consideration should be given to perform the plasma test during the thermal-vacuum test sequence following the corona check that occurs during the vacuum chamber pumpdown. When the chamber has been evacuated to a pressure level of less than 5×10^{-5} Torr (6.7×10^{-3} Pa), the plasma environment is established and its characteristics and effects analyzed. The energies and densities of the charged particles should be determined by the intended mission and estimated spacecraft potential. A functional test of the component in-flight configuration should then be performed in the plasma environment. An evaluation should be made for each application to determine if the test should be conducted.

C. Acceptance Requirements

The plasma test is performed as a Qualification/Acceptance test. The Acceptance test requirements, if required, are the same as the Qualification/Acceptance level test requirements for plasma tests.

2.2.8 TOXICITY TEST

A. Purpose

The toxicity test is performed on nonmetallic and certain metallic materials that are in a closed habitable environment where molecules or fragments of molecules may evolve and be hazardous to crew members. The test allows the materials to outgass the molecules to a level which is not hazardous to the crew.

B. Requirements

All materials must be evaluated for acceptability from a toxicological standpoint by using the criteria of NHB 8060.1, "Flammability, Odor and Offgassing Requirements and Test Procedures for Materials in Environments that Support Combustion." The toxicity test is performed at the component and integrated levels. The following procedures are used for components:

- Offgassed as assembled article. The summation of T values of all offgassed constituent products must not exceed 0.5.

- Evaluated on a materials basis when individual materials are used to make up components. The summation of T values of all offgassed constituent products must not exceed 0.5.

- Tested.as a single component, but more than one will be flown. The summation of T values for all flight units must not exceed 0.5.

- Bulk materials and other materials not inside containers are evaluated individually using the ratings and material codes in MSFC-HDBK-527, "Materials Selection List for Space Hardware Systems." The summation of T values for each individual material must not exceed 0.5.

C. Acceptance Requirements

The toxicity test or analysis is performed as a Qualification/Acceptance test. The Acceptance test requirements,if required, are the same as the Qualification/Acceptance test requirements.