GEVS-SE
Rev A
June 1996

GENERAL ENVIRONMENTAL VERIFICATION SPECIFICATION

FOR STS & ELV

PAYLOADS, SUBSYSTEMS, AND COMPONENTS

Approved By:

Robert C. Baumann
Director of Flight Assurance

NASA GODDARD SPACE FLIGHT CENTER
Greenbelt, Maryland 20771

GEVS-SE
Rev A
June 1996

GENERAL ENVIRONMENTAL VERIFICATION SPECIFICATION

FOR STS & ELV

PAYLOADS, SUBSYSTEMS, AND COMPONENTS

Prepared by: ______________________ ________
J. Scott Milne Date
System Reliability & Safety Office

Approved by: ______________________ ________
Jay B. Garvin Date
Chief, System Reliability & Safety Office

Approved by: ______________________ ________
Robert C. Baumann Date
Director of Flight Assurance

NASA GODDARD SPACE FLIGHT CENTER
Greenbelt, Maryland 20771

Changes* to

GENERAL ENVIRONMENTAL VERIFICATION SPECIFICATION
FOR STS & ELV
PAYLOADS, SUBSYSTEMS, AND COMPONENTS

Change No.	Date	Nature of Change	Initials
Rev A.	June 96.	Rewrite	

* Recipients of the original distribution of this document will automatically receive changes. Others can request changes in a memorandum to the Assurance Requirements Office Information Center, Code 300.1, NASA/GSFC, Greenbelt, Maryland 20771

ORGANIZATION OF THE DOCUMENT

The document consists of two sections and eleven appendices as follows:

Section 1 - General information including definitions, safety precautions, administration of the test program, failure criteria, distribution of revisions, testing and space hardware, test facilities, and tolerances.

Section 2 - System and environmental verification program including structural dynamics, pressure profile, mass properties, electromagnetic compatibility, thermal-vacuum, thermal balance, humidity, leakage, contamination control, and end-to-end testing.

Appendices A through L, General information and Structural Dynamic Test Levels

TABLE OF CONTENTS

Paragraph
Page

SECTION 1 -- GENERAL INFORMATION

SECTION 2.4 - STRUCTURAL AND MECHANICAL

Paragraph Page

SECTION 2.5 - EMC

SECTION 2.6 - THERMAL

Paragraph Page

APPENDICES -- TEST MARGINS AND LEVELS

Appendix Pages

Tables

Figure Page

Figure Page

FIGURES

Figure

Figure | Page

SECTION 1

GENERAL INFORMATION

1.1 PURPOSE

This specification provides guidelines for the development of environmental verification requirements for GSFC payloads, subsystems and components and describes methods for implementing those requirements. It contains a baseline for demonstrating by test or analysis the satisfactory performance of hardware in the expected mission environments, and that minimum workmanship standards have been met.

It presents the GSFC project and its contractors with source material and a model for preparing a project verification plan and a verification specification. It is not intended to be used in toto for contractual direction; rather the GSFC project verification management must select from the options to fulfill the specific payload (spacecraft) requirements in accordance with the launch vehicle to be used, Space Transportation System (STS), Atlas, Delta, Pegasus, Scout, Titan, etc., or to cover other mission-specific considerations. Most of the verification program is generally the same for STS and the expendable launch vehicles (ELV) payloads (spacecraft); the differences are noted in the text and the tables.

It is consistent with established GSFC payload assurance requirements. It elaborates on those requirements, gives guideline test levels, provides guidance in the choice of test options, and describes acceptable test and analytical methods for implementing the requirements.

1.2 APPLICABILITY AND LIMITATIONS

The specification applies to GSFC hardware that is to be launched on either the STS or on an ELV. The verification policy is defined by Goddard Management Instruction (GMI) 5330.7. Hardware launched by balloons and sounding rockets is not included. In accordance with the GMI, the specification applies to the following:

a. All space flight hardware, including interface hardware, that is developed as part of a payload managed by GSFC, whether developed by (1) GSFC or any of its contractors, (2) another NASA center, or (3) an independent agency; and

b. All space flight hardware, including interface hardware, that is developed by GSFC or any of its contractors and that is provided to another NASA installation or independent agency as part of a payload that is not managed by GSFC.

The requirements of this specification are intended for high-reliability, Class B, payloads. However, the specification shall also serve as a model in form and provide source material for deriving either less stringent verification requirements and specifications for higher-risk, lower-cost payloads, Class C or D, or for more stringent requirements and specifications for Class A payloads.

The provisions herein are generally limited to the verification of STS or ELV payloads and to those activities (with emphasis on the environmental verification program) that are closely associated with such verification, such as workmanship and functional testing. If the payload is to be serviced or recovered by the STS, then all STS verification and safety requirements apply..

The specification is written in accordance with the current GSFC practice of using a single protoflight payload for both qualification testing and space flight (see definition of hardware, 1.8). The protoflight verification program, therefore, is given as the nominal test program.

1.3 THE GSFC VERIFICATION APPROACH

Goddard Space Flight Center endorses the full systems verification approach in which the entire payload is tested or verified under conditions that simulate the flight operations and flight environment as realistically as possible. The specification is written in accordance with that view. However, it is recognized that there may be unavoidable exceptions, or conditions which make it preferable to perform the verification activities at lower levels of assembly. For example, testing at lower levels of assembly may be necessary to produce sufficient environmentally induced stresses to uncover design and workmanship flaws. These test requirements should be tailored for each specific space program. For some projects, tailoring might relax the requirements in this standard; however, for other projects the requirements may be made more stringent to demonstrate more robustness or greater confidence in the system performance.

Since testing at the component (or unit) level, or lower level of assembly for large components, often becomes a primary part of the verification program, all components should be operating and monitored during all environmental tests if practicable.

Environmental verification of hardware is only a portion of the total assurance effort at GSFC that establishes confidence that a payload will function correctly and fly a successful mission. The environmental test program provides confidence that the design will perform when subjected to environments more severe than expected during the mission, and provides environmental stress screening to uncover workmanship defects.

The total verification process also includes the development of models representing the hardware, tests to verify the adequacy of the models, analyses, alignments, calibrations, functional/performance tests to verify proper operation, and finally end-to-end tests and simulations to show that the total system will perform as specified.

Other tests not included herein may be performed as required by the project. The level, procedure, and decision criteria for performing any such additional tests shall be included in the system verification plan and system verification specification (section 2.1).

1.4 OTHER ASSURANCE REQUIREMENTS

In addition to the verification program, the assurance effort include parts and materials selection and control, reliability assessment, quality assurance, software assurance, design reviews, and system safety.

1.5 RESPONSIBILITY FOR ADMINISTRATION

The responsibility and authority for decisions in applying the requirements of this specification rest with the project manager. The general/environmental requirements are intended for use by the flight project managers, assisted by the flight assurance managers, and verification managers in developing project-unique performance verification requirements, plans, and specifications that are consistent with current NASA program/project planning.

The requirements thus derived and the deviations from the requirements of this document are subject to review by the Director of Flight Assurance, GSFC.

1.6 DISTRIBUTION OF REVISIONS

Users who receive this document in the original distribution will also receive revisions and changes. Others can request changes from the Assurance Requirements Office Information Center, Code 300.1, NASA/GSFC, Greenbelt, Maryland, 2077l. Users are advised to contact the AROIC to make sure they have the latest revision.

1.7 APPLICABLE DOCUMENTS

The following documents may be needed in formulating the environmental test program. The user must ensure that the latest versions are procured and that the most recent changes and additions are included.

1.7.1 Safety Requirements - NSTS 1700.7, Safety Policy and Requirements for Payloads using the NSTS, states that "the safety of any hazardous payload safety-critical equipment shall be satisfactorily verified." Because testing is one of the acceptable methods for verifying safety compliance, the environmental test program may be influenced by safety considerations.

1.7.2 NSTS Interface Requirements - Portions of ICD 2-19001, Shuttle Orbiter/Cargo Standard Interfaces (Attachment 1 to NSTS 07700, Vol. XIV) have been incorporated herein primarily to make up part of the electromagnetic compatibility (EMC) provisions. ICD 2-19001 should also be consulted as indicated for implementing some of the other sections. Similarly, many of the provisions of NSTS 14046, Payload Interface Verification Requirements have been incorporated in this specification. STS users should, however, refer to that document to ensure full compliance.

1.7.3 ELV Payload User Manuals - The most recent versions of the following documents are applicable in accordance with the launch vehicle to be used by the project.

1.7.3.1 Ariane 4 User's Manual, Arianespace Inc., U.S. subsidiary, 700 13th St. N.W., Suite 230, Washington D.C. 20005.

1.7.3.2 Ariane 5 User's Manual, Arianespace Inc., U.S. subsidiary, 700 13th St. N.W., Suite 230, Washington D.C. 20005.

1.7.3.3 Atlas Mission Planner's Guide for the Atlas Launch Vehicle Family, Lockheed Martin Astronautics Commercial Launch Services, Inc., 5001 Kearny Villa Road, San Diego, California 92123.

1.7.3.4 Conestoga Payload User's Guide, EER Systems Corp., 1593 Spring Hill Road, Vienna, VA 22182

1.7.3.5 Delta II Payload Planner's Guide (MDC H 3224C), McDonnell Douglas Aerospace, 5301 Bolsa Ave., Huntington Beach, California 92647

1.7.3.6 Lockheed Martin Launch Vehicle User’s Guide, Preliminary Release, Lockheed Martin Astronautics, P.O. Box 179, Denver, Colorado 80201.

1.7.3.7 Commercial Pegasus Launch System-Payload User's Guide, Orbital Sciences Corporation, 21700 Atlantic Blvd., Dulles, VA. 20116.

1.7.3.8 Scout User's Manual, LTV Aerospace and Defense, Vought Missiles and Advanced Programs Division, P.O. Box 650003, Dallas, Texas 75265-0003.

1.7.3.9 Commercial Taurus Launch System-Payload User's Guide, Orbital Sciences Corporation, 21700 Atlantic Blvd., Dulles, VA. 20116.

1.7.3.10 Titan II Space Launch Vehicle: Payload User's Guide, Lockheed Martin Astronautics, P.O. Box 179, Denver, Colorado 80201.

1.7.3.11 Titan III Commercial Launch Services Customer Handbook, Lockheed Martin Astronautics, P.O. Box 179, Denver, Colorado 80201.

1.7.3.12 Titan IV User's Handbook (MCR-86-2541), Lockheed Martin Astronautics, P.O. Box 179, Denver, Colorado 80201.

1.7.4 Fracture Control and Stress Corrosion - NSTS 1700.7, above, states the policy on fracture control for the STS. MSFC-SPEC-522, Stress Corrosion Requirements, provides design criteria for preventing stress corrosion. Implementation of fracture control and stress corrosion prevention measures on GSFC projects shall be in accordance with GSFC document 731-0005-83, latest revision, Fracture Control Plan for Payloads Using the Space Transportation System, or Fracture Control Plan for Payloads Using Expendable Launch Vehicles.

1.7.5 Spacecraft Tracking and Data Network Simulation - STDN No. 101.6, Portable Simulation System and Simulations Operation Center Guide for TDRSS & GSTDN, describes the Spacecraft Tracking and Data Network (STDN) and the Tracking and Data Relay Satellite (TDRS)/Ground STDN network simulation programs, and the Simulations Operations Center (SOC). It also discusses end-to-end simulation techniques. STDN No. 408, TDRS and GSTDN Compatibility Test Van Functional Description and Capabilities, describes the equipment and the compatibility test system.

1.7.6 Deep Space Network (DSN) Simulation - The Deep Space Network/Flight Project Interface Design Handbook, 8l0-5, Jet Propulsion Laboratory, California Institute of Technology, Vol. I, Module TSS-10, describes existing payload (spacecraft) telemetry and command simulation capability. Vol. II describes proposed DSN capability.

1.7.7 Payload Bay Acoustic Study - The PACES computer program for making estimates of the effects of a payload on the acoustic environment of the payload bay is contained in NASA CR 159956, Space Shuttle Payload Bay Acoustic Protection Study, Vols. I through V.

l.7.8 Military Standards for EMC Testing - Pertinent sections of the following standards are needed to conduct the EMC tests:

a. MIL-STD-461C, Electromagnetic Interference Characteristics Requirements for Equipment.

b. MIL-STD-462, Electromagnetic Interference Characteristics, Measurement of, as amended by Notice l.

c. MIL-STD-463A, Definitions and Systems of Units, Electromagnetic Interference and Electromagnetic Compatibility Technology.

1.7.9 Military Standards for Non-Destructive Evaluation

a. MIL-I-6870E, Inspection Program Requirements, Non-Destructive Testing for Aircraft and Missile Materials and Parts.

b. MIL-STD-410D, Non-Destructive Testing, Personnel Qualification and Certification (Eddy Current, Liquid Penetrant, Magnetic Particle, Radiographic and Ultrasonic).

1.8 DEFINITIONS

The following definitions apply within the context of this specification:

Acceptance Tests: The verification process that demonstrates that hardware is acceptable for flight. It 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.

Assembly: See Level of Assembly.

Component: See Level of Assembly.

Configuration: The functional and physical characteristics of the payload and all its integral parts, assemblies and systems that are capable of fulfilling the fit, form and functional requirements defined by performance specifications and engineering drawings.

Contamination: The presence of materials of molecular or particulate nature which degrade the performance of hardware.

Design Qualification Tests: Tests intended to demonstrate that the test item will function within performance specifications under simulated conditions more severe than those expected from ground handling, launch, and orbital operations. Their purpose is to uncover deficiencies in design and method of manufacture. They are not intended to exceed design safety margins or to introduce unrealistic modes of failure. The design qualification tests may be to either “prototype” or “protoflight” test levels.

Design Specification: Generic designation for a specification that describes functional and physical requirements for an article, usually at the component level 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.

Electromagnetic Compatibility (EMC): The condition that prevails when various electronic devices are performing their functions according to design in a common electromagnetic environment.

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

Electromagnetic Susceptibility: Undesired response by a component, subsystem, or system to conducted or radiated electromagnetic emissions.

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

Failure: A departure from specification that is discovered in the functioning or operation of the hardware or software. See nonconformance.

Flight Acceptance: See Acceptance Tests.

Fracture Control Program: A systematic project activity to ensure that a payload intended for flight has sufficient structural integrity as to present no critical or catastrophic hazard. Also to ensure quality of performance in the structural area for any payload (spacecraft) project. Central to the program is fracture control analysis, which includes the concepts of fail-safe and safe-life, defined as follows:

a. Fail-safe: Ensures that a structural element, because of structural redundancy, will not cause collapse of the remaining structure or have any detrimental effects on mission performance.

b. Safe-life: Ensures that the largest flaw that could remain undetected after non-destructive examination would not grow to failure during the mission.

Functional Tests: The operation of a unit in accordance with a defined operational procedure to determine whether performance is within the specified requirements.
Hardware: As used in this document, there are two major categories of hardware as follows:

a. Prototype Hardware: Hardware of a new design; it is subject to a design qualification test program; it is not intended for flight.

b. Flight Hardware: Hardware to be used operationally in space. It includes the following subsets:

   (1) Protoflight Hardware: Flight hardware of a new design; it is subject to a qualification test program that combines elements of prototype and flight acceptance verification; that is, the application of design qualification test levels and flight acceptance test durations.

   (2) Follow-On Hardware: Flight hardware built in accordance with a design that has been qualified either as prototype or as protoflight hardware; follow-on hardware is subject to a flight acceptance test program.

   (3) Spare Hardware: Hardware the design of which has been proven in a design qualification test program; it is subject to a flight acceptance test program and is used to replace flight hardware that is no longer acceptable for flight.

   (4) Reflight Hardware: Flight hardware that has been used operationally 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.

Level of Assembly: The environmental test requirements of GEVS generally start at the component or unit level assembly and continue hardware/software build through the system level (referred to in GEVS as the payload or spacecraft level). The assurance program includes the part level. Verification testing may also include testing at the assembly and subassembly levels of assembly; for test recordkeeping these levels are combined into a "subassembly" level. The verification program continues through launch, and on-orbit performance. The following levels of assembly are used for describing test and analysis configurations:

Assembly: A functional subdivision of a component consisting of parts or subassemblies that perform functions necessary for the operation of the component as a whole. Examples are a power amplifier and gyroscope.

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 electronic box, transmitter, gyro package, actuator, motor, battery. For the purposes of this document, "component" and "unit" are used interchangeably.

Instrument: A spacecraft subsystem consisting of sensors and associated hardware for making measurements or observations in space. For the purposes of this document, an instrument is considered a subsystem (of the spacecraft).

Module: A major subdivision of the payload that is viewed as a physical and functional entity for the purposes of analysis, manufacturing, testing, and recordkeeping. Examples include spacecraft bus, science payload, and upper stage vehicle.

Part: A hardware element that is not normally subject to further subdivision or disassembly without destruction of design use. Examples include resistor, integrated circuit, relay, connector, bolt, and gaskets.

Payload: An integrated assemblage of modules, subsystems, etc., designed to perform a specified mission in space. For the purposes of this document, "payload" and "spacecraft" are used interchangeably. Other terms used to designate this level of assembly are Laboratory, Observatory, and satellite.

Spacecraft: See Payload. Other terms used to designate this level of assembly are Laboratory, Observatory, and satellite.

Section: A structurally integrated set of components and integrating hardware that form a subdivision of a subsystem, module, etc. A section forms a testable level of assembly, such as components/units mounted into a structural mounting tray or panel-like assembly, or components that are stacked.

Subassembly: A subdivision of an assembly. Examples are wire harness and loaded printed circuit boards.

Subsystem: A functional subdivision of a payload consisting of two or more components. Examples are structural, attitude control, electrical power, and communication subsystems. Also included as subsystems of the payload are the science instruments or experiments.

Unit: A functional subdivision of a subsystem, or instrument, and generally a self-contained combination of items performing a function necessary for the subsystem's operation. Examples are electronic box, transmitter, gyro package, actuator, motor, battery. For the purposes of this document, "component" and "unit" are used interchangeably.

Limit Level: The maximum expected flight level (consistent with the minimum probability levels of Table 2.4-2).

Margin: The amount by which hardware capability exceeds requirements.

Module: See Level of Assembly.

Nonconformance: A condition of any hardware, software, material, or service in which one or more characteristics do not conform to requirements.

Offgassing: The emanation of volatile matter of any kind from materials into a manned pressurized volume.

Outgassing: The emanation of volatile materials under vacuum conditions resulting in a mass loss and/or material condensation on nearby surfaces.

Part: See Level of Assembly.

Payload: See Level of Assembly.

Performance Verification: Determination by test, analysis, or a combination of the two that the payload element can operate as intended in a particular mission; this includes being satisfied that the design of the payload or element has been qualified and that the particular item has been accepted as true to the design and ready for flight operations.

Protoflight Testing: See Hardware.

Prototype Testing: See Hardware.

Qualification: See Design Qualification Tests.

Redundancy (of design): The use of more than one independent means of accomplishing a given function.

Section: See Level of Assembly.

Spacecraft: See Level of Assembly.

Subassembly: See Level of Assembly.

Subsystem: See Level of Assembly.

Temperature Cycle: A transition from some initial temperature condition to temperature stabilization at one extreme and then to temperature stabilization at the opposite extreme and returning to the initial temperature condition.

Temperature Stabilization: The condition that exists when the rate of change of temperatures has decreased to the point where the test item may be expected to remain within the specified test tolerance for the necessary duration or where further change is considered acceptable.

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-Vacuum Test: A test conducted to demonstrate the capability of the test item to operate satisfactorily in vacuum at temperatures based on those expected for the mission. The test, including the gradient shifts induced by cycling between temperature extremes, can also uncover latent defects in design, parts, and workmanship.

Unit: See Level of Assembly.

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

Workmanship Tests: Tests performed during the environmental verification program to verify adequate workmanship in the construction of a test item. It is often necessary to impose stressses beyond those predicted for the mission in order to uncover defects. Thus random vibration tests are conducted specifically to detect bad solder joints, loose or missing fasteners, improperly mounted parts, etc. Cycling between temperature extremes during thermal-vacuum testing and the presence of electromagnetic interference during EMC testing can also reveal the lack of proper construction and adequate workmanship.

1.9 ENVIRONMENTAL VERIFICATION COMMITTEE

It is recommended that the payload project establish an environmental verification committee. Its responsibilities should include assessment of environmental test requirements in accordance with current GSFC practices, approval of environmental verification plans and specifications, consideration of waivers, resolution of problems, and determination of corrective action. The committee should verify that the test program is adequate to enable the hardware to meet the mission objective, and it should evaluate test results to certify compliance with specifications. Members of the committee should include representatives of the following disciplines: payload management, instrument management, systems engineering, environmental testing, verification and flight assurance.

1.10 CRITERIA FOR UNSATISFACTORY PERFORMANCE

Deterioration or any change in performance of any test item that does or could in any manner prevent the item from meeting its functional, operational, or design requirements throughout its mission shall be reason to consider the test item as having failed. Other factors concerning failure are considered in the following paragraphs.

1.10.1 Failure Occurrence

When a failure (non-conformance or trend indicating that an out of spec condition will result) occurs, a determination shall be made as to the feasibility and value of continuing the test to its specified conclusion. If corrective action is taken, the test shall be repeated to the extent necessary to demonstrate that the test item's performance is satisfactory.

1.10.2 Failures with Retroactive Effects

If corrective action taken as a result of failure, e.g. redesign of a component, affects the validity of previously completed tests, prior tests shall be repeated to the extent necessary to demonstrate satisfactory performance.

1.10.3 Failure Reporting

Every failure shall be recorded and reported in accordance with the failure reporting provisions of the project.

1.10.4 Wear Out

If during a test sequence a test item is operated in excess of design life and wears out or becomes unsuitable for further testing from causes other than deficiencies, a spare may be substituted. If, however, the substitution affects the significance of test results, the test during which the item was replaced and any previously completed tests that are affected shall be repeated to the extent necessary to demonstrate satisfactory performance.

1.11 TEST SAFETY RESPONSIBILITIES

The following paragraphs define the responsibilities shared by the space project and facility management for planning and enforcing industrial safety measures taken during testing for the protection of personnel, the payload, and the test facility.

1.11.1 Operations Hazard Analysis, Responsibilities For

It shall be the joint responsibility of the test facility manager and the project manager to ensure that environmental tests and associated operations present no unacceptable hazard to the test item, facilities, or personnel. A test operations hazard analysis (OHA) shall be performed by the facility and project personnel to consider and evaluate all hazards presented by the interaction of the payload and the facility for each environmental test. All hazards discovered in the OHA shall be tracked to an agreed-upon resolution. The safety measures to be taken as a result of the OHA, as well as the safety measures between tests, shall be specified as requirements in the verification plan and verification specification.

1.11.2 Treatment of Hazards

As hazards are discovered, a considered attempt shall be made to eliminate them. This may be accomplished by redesign, controlling energy sources, revising the test, or by some other method. If the hazard cannot be eliminated, automatic safety controls shall be applied, for example: pressure relief devices, electrical circuit protection devices, or mechanical interlocks. If that is not possible or is too costly, warning devices shall be considered. If none of the foregoing methods are practicable, control procedures must be developed and applied. In practice, a combination of all four methods may be the best solution to the hazards posed by a complex system. Before any test begins, the project manager and test facility management shall agree on the hazard control method(s) that are to be used.

1.11.3 Facility Safety

The test facility manager shall verify that the test facility and normal operations present no unacceptable hazard to the test item, test and support equipment, or personnel. He shall ensure that facility personnel abide by all applicable regulations, observe all appropriate industrial safety measures, and follow all requirements for protective equipment. He shall ensure that all facility personnel are trained and qualified for their positions. Training should include the handling of emergencies by the simulation of emergency conditions. Analyses, tests, and inspections shall be performed to verify that the safety requirements are satisfied. The approach outlined in 1.11.2 shall be used to eliminate or control hazards.

1.11.4 Safety Responsibilities During Tests

The test facility manager shall appoint a safety officer to work closely with a safety officer designated by the space project. The facility designee shall ensure that the facility meets applicable Occupational Safety & Health Act (OSHA) and other requirements, that appropriate industrial safety measures are observed, and that protective equipment is provided for all personnel involved. The facility designee will ensure that facility personnel use the equipment provided and that the test operation does not present a hazard to the facility. The project designee shall ensure that project personnel use the equipment provided and that the test operation does not present a hazard to the space hardware, equipment, or personnel.

1.12 TESTING OF SPARE HARDWARE

A supply of selected spares is often maintained in case of the failure of flight hardware. As a minimum, spares must undergo a verification program equal to that required for follow-on hardware. Therefore, special consideration must be given to spares as follows:

a. Extent of Testing - The extent and type of testing shall be determined as part of the flight hardware test program. A spare unit may be used for qualification of the hardware by subjecting it to protoflight testing, and testing the flight hardware to acceptance levels.

b. Spares From Failed Elements - If a flight element is replaced for reasons of failure and is then repaired and redesignated as a spare, appropriate retesting shall be conducted.

c. Caution on the Use of Spares - When the need for a spare arises, immediate analysis and review of the failed hardware must be made. If failure occurs in a hardware item of which there are others of identical design, the fault may be generic and may affect all hardware of that design.

d. "One-Shot" Items - Some items may be degraded or expended during the integration and test period and replaced by spares. The spare that is used shall have met the required quality control standards or auxiliary tests for such items and shall be of qualified design. Examples are pyrotechnic devices, yo-yo despin weights, and elements that absorb impact energy by plastic yielding. When the replacement entails procedures that could jeopardize mission success, the replacement procedure should be successfully demonstrated with the hardware in the same configuration that it will be in when final replacement is to be accomplished.

1.13 TEST FACILITIES, CALIBRATION

The facilities and fixtures used in conducting tests shall be capable of producing and maintaining the test conditions prescribed with the test specimen installed and operating or not operating, as required. In any major test, facility performance should be verified prior to the test either by a review of its performance during a test that occurred a short time earlier or by conducting a test with a substitute test item. All equipment used for tests shall be in current calibration and so noted by tags and stickers.

1.14 TEST CONDITION TOLERANCES

In the absence of a rationale for other test condition tolerances, the following shall be used; the values include measurement uncertainties:

Acoustics	Overall Level:	≤ 1 dB	
	l/3 Octave Band Tolerance:	Frequency (Hz)	Tolerance (dB)
		f ≤ 40	+3, -6
		40< F < 3150	± 3
		f ≥ 3150	+3, -6
Antenna Pattern Determination		± 2 dB	
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	
Humidity		± 5% RH	
Loads	Steady-State (Acceleration):	± 5%	
	Static:	± 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 centerline from cg of assembly:		± 5 cm
	Mapping turntable angular displacement:		± 3 degrees
	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
Mass Properties	Weight:		± 0.2%
	Center of Gravity:		± 0.15cm (± 0.06 in.)
	Moments of Inertia:		± 1.5%
Mechanical Shock	Response Spectrum:		+25%, -10%
	Time History:		± 10%
Pressure	Greater than 1.3 X 10^4 Pa (Greater than 100 mm Hg):		± 5%
	1.3 X 10^4 to 1.3 X 10^2 Pa (100 mm Hg to 1 mm Hg):		± 10%
	1.3 X 10^2 to 1.3 X 10^1 Pa (1 mm Hg to 1 micron):		± 25%
	Less than 1.3 X 10^1 Pa (less than 1 micron):		± 80%
Temperature			± 2°C
Vibration	Sinusoidal:	Amplitude	± 10%
		Frequency	± 2%
	Random:	RMS level	± 10%
		Accel. Spectral Density	± 3 dB

SECTION 2.1

SYSTEM PERFORMANCE VERIFICATION

2.l SYSTEM PERFORMANCE VERIFICATION

This section applies to all payloads (spacecraft), subsystems (including instruments), and components. The basic provisions apply to all flight hardware, and associated software, that will fly in the STS cargo bay and to spacecraft that will be launched by expendable launch vehicles (ELVs).

The GEVS, as its name implies, provides basic requirements and guidelines for an environmental verification program. This represents only a portion of the overall system verification and must be integrated into the total system program which verifies that the system will meet the mission requirements. A system performance verification program documenting the overall verification plan, implementation, and results is required which will provide traceability from mission specification requirements to launch and initial on-orbit capability. This will also provide the baseline for tracking on-orbit performance versus pre-launch capability.

2.1.1 Documentation Requirements

The following documents are required and shall be delivered and approved in accordance with the Contracts Schedule.

2.1.1.1 System Performance Verification Plan

A system performance verification plan shall be prepared defining the tasks and methods required to determine the ability of the system (or instrument) to meet each program-level performance requirement (structural, thermal, optical, electrical, guidance/control, RF/telemetry, science, mission operational, etc.) and to measure specification compliance. Limitations in the ability to verify any performance requirement shall be addressed, including the addition of supplemental tests and/or analyses that will be performed and a risk assessment of the inability to verify the requirement.

The plan shall address how compliance with each specification requirement will be verified. If verification relies on the results of measurements and/or analyses performed at lower (or other) levels of assembly, this dependence shall be described.

For each analysis activity, the plan shall include objectives, a description of the mathematical model, assumptions on which the models will be based, required output, criteria for assessing the acceptability of the results, the interaction with related test activity, if any, and requirements for reports. Analysis results shall take into account tolerance build-ups in the parameters being used.

2.1.1.1.1 Environmental Verification Plan

An environmental verification plan shall be prepared, either as part of the System Verification Plan or as a separate document, that prescribes the tests and analyses that will collectively demonstrate that the hardware and software comply with the environmental verification requirements

The environmental verification plan shall provide the overall approach to accomplishing the environmental verification program. For each test, it shall include the level of assembly, the configuration of the item, objectives, facilities, instrumentation, safety considerations, contamination control, test phases and profiles, necessary functional operations, personnel responsibilities, and requirement for procedures and reports. It shall also define a rationale for retest determination that does not invalidate previous verification activities. When appropriate, the interaction of the test and analysis activity shall be described.

Limitations in the environmental verification program which preclude the verification by test of any system requirement shall be documented. Examples of limitations in the ability to demonstrate requirements include:

- Inability to deploy hardware in a 1-g environment.
- Facility limitations which do not allow testing at system level of assembly.
- Inability to perform certain tests because of contamination control requirements.
- Inability to perform powered-on testing because of voltage breakdown concerns.

Alternative tests and analyses shall be evaluated and implemented as appropriate, and an assessment of program risk shall be included in the System Performance Verification Plan.

2.1.1.2 System Performance Verification Matrix

A System Performance Verification Matrix shall be prepared, and maintained, to show each specification requirement, the reference source (to the specific paragraph or line item), the method of compliance, applicable procedure references, results, report reference numbers, etc. This matrix shall be included in the system review data packages showing the current verification status as applicable

2.1.1.2.1 Environmental Test Matrix

As an adjunct to the environmental verification plan, an environmental test matrix shall be prepared that summarizes all tests that will be performed on each component, each subsystem, and the payload. The purpose is to provide a ready reference to the contents of the test program in order to prevent the deletion of a portion thereof without an alternative means of accomplishing the objectives; it has the additional purpose of ensuring that all flight hardware has been subjected to environmental exposures that are sufficient to demonstrate acceptable workmanship. In addition, the matrix shall provide traceability of the qualification heritage of hardware. All flight hardware, spares and prototypes (when appropriate) shall be included in the matrix. Details of each test shall be provided (e.g., number of thermal cycles, temperature extremes, vibration levels). It shall also relate the design environments to the test environments and to the anticipated mission environments. The matrix shall be prepared in conjunction with the initial environmental verification plan and shall be updated as changes occur.

A sample test matrix is given in Figure 2.1-1. The electrical performance tests that are required to be performed before, during, and following the environmental verification test program are not shown in this sample matrix. Other performance tests, measurements, demonstrations, alignments, etc. (electrical, mechanical, optical, etc.), that must be performed to verify hardware/software requirements are also not included in this Environmental Test Matrix. However they shall be included in the System Performance Verification Plan.

The test matrix does not have to conform to this format; any format that clearly displays the pertinent information is acceptable.

A complementary matrix shall be kept showing the tests that have been performed on each component, subsystem, or payload (or applicable level of assembly). This should include tests performed on prototypes or engineering units used in the qualification program, and should indicate test results (pass/fail or malfunctions).

2.1.1.3 Environmental Verification Specification

An environmental verification specification shall be prepared that defines the specific environmental parameters that each hardware element is subjected to either by test or analysis in order to demonstrate its ability to meet the mission performance requirements. Such things as payload peculiarities and interaction with the launch vehicle (STS or ELV) shall be taken into account.

2.1.1.4 Performance Verification Procedures

For each verification test activity conducted at the component, subsystem, and payload levels (or other appropriate levels) of assembly, a verification procedure shall be prepared that describes the configuration of the test article, how each test activity contained in the verification plan and specification will be implemented.

Test procedures shall contain details such as instrumentation monitoring, facility control sequences, test article functions, test parameters, pass/fail criteria, quality control checkpoints, data collection and reporting requirements. The procedures also shall address safety and contamination control provisions.

2.1.1.5 Verification Reports

After each component, subsystem, payload, etc., verification activity has been completed, a report shall be submitted in accordance with the Contract Schedule. For each environmental test activity, the report shall contain, as a minimum, the information in the sample test report contained in Figure 2.1-2a and 2.1-2b. For each analysis activity, the report shall describe the degree to which the objectives were accomplished, how well the mathematical model was validated by related test data, and other such significant results. In addition, as-run verification procedures and all test and analysis data shall be retained for review.

2.1.1.6 System Performance Verification Report

At the conclusion of the verification program, a final System Performance Verification Report shall be delivered comparing the hardware/software specifications with the final verified values (whether measured or computed). It is recommended that this report be subdivided by subsytem/instrument.

The System Performance Verification Report shall be maintained "real-time" throughout the program summarizing the successful completion of verification activities, and showing that the applicable system performance specifications have been acceptably complied with prior to integration of hardware/software into the next higher level of assembly.

The initial report shall be provided for the PDR. Current versions shall then be provided for review at major systems reviews.

The final pre-launch System Verification Report shall be available for approval for the FRR (Flight Readiness Review).

Following initial on-orbit checkout, the System Verification Report shall be completed, and delivered in accordance with the contract schedule.

2.1.1.7 Instrument Verification Documentation

The documentation requirements of sections 2.1.1.1 through 2.1.1.6 also apply to instruments. Following integration of the instruments onto the spacecraft, the spacecraft System Verification Report will include the instrument information.

Figure 2.1-1

ENVIRONMENTAL TEST MATRIX FOR

____

____

REVISION NO. ______

REVISION DATE _____

PAGE _______ OF _______

HARDWARE DESCRIPTION					
LEVEL OF ASSEMBLY	ITEM	S/N	UNIT TYPE	SUPPLIER	QUALIFICATION STATUS

STRUCTURAL MECHANICAL												
MODAL SURVEY	STATIC LOADS	ACCELERATION	SINE BURST	SINE VIBRATION	RANDOM VIBRATION	ACOUSTICS	MECHANICAL SHOCK	PRESSURE PROFILE	MECHANICAL FUNCTION	TORQUE RATIO	LIFE TESTS	MASS PROPERTIES

EMC & MAGNETICS																					
EMISSIONS											SUSCEPTIBILITY										
CONDUCTED						RADIATED					CONDUCTED					RADIATED					
DC POWER LEADS	AC POWER LEADS	POWER LEADS	SPIKES ON ORBITER DC POWER LINES	SPIKES ON ORBITER AC POWER LINES	ANTENNA TERMINALS	MAGNETIC FIELD (S/S PAYLOADS)	AC MAGNETIC FIELD	E-FIELDS	PAYLOAD TRANSMITTERS	SPURIOUS (XMTR ANTENNA)	POWER LINE	INTERMODULATION PRODUCTS	SIGNAL REJECTION	CROSS MODULATION	POWER LINE TRANSIENTS	E-FIELD (GENERAL COMPATIBILITY)	COMPATIBILITY WITH ORBITER XMTRS	ORBITER UNINTENTIONAL E-FIELD	MAGNETIC-FIELD SUSCEPTIBILITY	MAGNETIC PROPERTIES	

THERMAL								
LEAK	NUMBER OF THERMAL-VACUUM CYCLES	NO. OF THERMAL CYCLES (NON-VAC)	THERMAL BALANCE	TEMPERATURE-HUMIDITY	BAKEOUT	FLIGHT TEMPERATURE PREDICTIONS	TEST TEMPERATURE LIMITS	REMARKS

LEGEND

LEVEL OF ASSEMBLY

- S/C - SPACECRAFT
- P/L - PAYLOAD
- MOD - MODULE
- S - SUBSYSTEM
- I - INSTRUMENT
- SEC - SECTION
- C - COMPONENT

UNIT TYPE

- D - DEVELOPMENT MODEL
- EM - ENGINEERING MODEL
- PT - PROTOTYPE
- PF - PROTOFLIGHT
- F - FLIGHT
- S - SPARE

QUALIFICATION STATUS

1. - COMPLETE QUAL. REQUIRED
2. - PARTIAL QUAL. REQUIRED (SEE REMARKS)
3. - OTHERWISE QUALIFIED (SEE REMARKS)

NOTES:

SYSTEM PERFORMANCE VERIFICATION

VERIFICATION TEST REPORT

Page _____ of _____

PROJECT ______________________________

TEST ITEM ______________________________

MANUFACTURER ______________________________

SERIAL NUMBER ______________________________

LEVEL OF ASSEMBLY

- ☐ SUBASSEMBLY or ASSEMBLY
- ☐ UNIT/COMPONENT
- ☐ SECTION
- ☐ SUBSYSTEM/INSTRUMENT
- ☐ MODULE
- ☐ SPACECRAFT/PAYLOAD

HARDWARE

- ☐ ENGINEERING MODEL
- ☐ PROTOTYPE
- ☐ PROTOFLIGHT
- ☐ FLIGHT
- ☐ SPARE

TEST

- ☐ INITIAL TEST
  - STARTING DATE OF INITIAL TEST ______________
- ☐ RETEST
  - ☐ PARTIAL
  - ☐ FULL

STRUCTURAL - MECHANICAL

- ☐ STRUCTURAL LOADS
  - ☐ STATIC
  - ☐ ACCEL.
  - ☐ SINE BURST
- ☐ VIBRATION
  - ☐ RANDOM
  - ☐ SINE
- ☐ ACOUSTICS
- ☐ MECHANICAL SHOCK
  - ☐ ACTUATION
  - ☐ SIMULATED
- ☐ MECHANICAL FUNCTION
- ☐ MODAL SURVEY
- ☐ PRESSURE PROFILE
- ☐ MASS PROPERTIES
- ☐ OTHER (explain)

ELECTROMAGNETIC COMPATIBILITY

- ☐ CONDUCTED EMISSIONS
- ☐ RADIATED EMISSION
- ☐ CONDUCTED SUSCEPTIBILITY
- ☐ RADIATED SUSCEPTIBILITY
- ☐ MAGNETIC PROPERTIES

ELECTRICAL PERFORMANCE

- ☐ LPT
- ☐ CPT
- ☐ END-TO-END
  - ☐ COMPATIBILITY TEST
  - ☐ MISSION SIMULATIONS

THERMAL

- ☐ THERMAL-VACUUM
  - (no. of cycles _____)
- ☐ THERMAL CYCLING
  - (no. of cycles _____)
- ☐ THERMAL BALANCE
- ☐ TEMPERATURE-HUMIDITY
- ☐ LEAKAGE
- ☐ OTHER (explain)

OPTICAL

- ☐ EXPLAIN

VERIFICATION PROCEDURE NO.: __________ REV. _____ DATE __________

APPLICABLE VERIFICATION PLAN: ______________________________

FACILITY DESCRIPTION: ______________________________

LOCATION: ______________________________

TEST LOG REFERENCE: ______________________________

COMMENTS:

SIGNATURES

COGNIZANT ENGINEER FOR TEST ITEM: ______________________________ DATE: __________

QUALITY ASSURANCE REPRESENTATIVE: ______________________________ DATE: __________
(if required)

Figure 2.1-2a Verification Test Report

VERIFICATION TEST REPORT (Continued) Page_____of_____

Date (add time for thermal and temperature tests)	Note beginning and end of actual activity, deviations from the planned procedure, and discrepancies in test times or performance. (State if there were no deviations or discrepancies.)	Malfunction Report Number and Date (if applicable)
	(use additional paper as required)	

The activities covered by these reports include tests and measurements performed for the purpose of verifying the flightworthiness of hardware at the component, subsystem, and payload levels of assembly. These reports shall also be provided for such other activities as the project may designate.

These reports shall be completed and transmitted to the GSFC Technical Officer or Contracting Officer (as appropriate) within 30 days after completion of an activity. Legible, reproducible, handwritten completed forms are acceptable.

Material felt necessary to clarify this report may be attached. However, in general, test logs and data should be retained by those responsible for the test item unless they are specifically requested.

The forms shall be signed by the quality assurance representative and the person responsible for the test or his designated representative; the signatures represent concurrence that the data is as accurate as possible given the constraints of time imposed by quick-response reporting.

This report does not replace the need for maintaining complete logs, records, etc.; it is intended to document the implementation of the verification program and to provide a minimum amount of information as to the performance of the test item.

Figure 2.1-2b Verification Test Report (cont.)

SECTION 2.2

ENVIRONMENTAL VERIFICATION

2.2 APPLICABILITY

Sections 2.3 through 2.8 give the basic environmental verification program for verifying payloads, subsystems, and components as follows:

2.3 Electrical Function & Performance
2.4 Structural and Mechanical
2.5 EMC
2.6 Thermal
2.7 Contamination Control
2.8 End-to-End Testing (payloads/spacecraft)

The verification program applies to payloads that will fly in the STS cargo bay and to spacecraft that will be launched by expendable launch vehicles (ELVs). Provisions that are specific to STS or ELV payloads are noted in the text and tables. For the purposes of this document, a spacecraft is considered a payload, and an instrument is considered to be a subsystem when determining the environmental verification requirements.

The basic provisions are written assuming protoflight hardware. They are, in general, also applicable to prototype hardware. Acceptance requirements are also given for the flight acceptance of previously qualified hardware. This applies to follow-on hardware (multiple copies of the same item) developed for the program, or hardware (from another program) qualified by similarity.

2.2.1 Test Sequence and Level of Assembly

The verification activities herein are grouped by discipline; they are not in a recommended sequence of performance. 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.

In cases where the magnetic properties of the hardware need to be controlled, the dc magnetics testing should be performed after vibration testing. This provides an opportunity to correct for any magnetization of the flight hardware caused by fields associated with the vibration test equipment.

Table 2.2-1 provides a hierarchy of levels of assembly for the flight hardware, with examples. These level designators are based on those used in the Space Systems Engineering Database developed by The Aerospace Corporation for the Air Force, and agreed to by NASA Headquarters, GSFC, and JPL.. The GEVS environmental test requirements generally start at the “unit” level and end at the “system segment” level. However, screening and life-tests often occur at lower levels, and overall system verification continues beyond the “system segment” level.

2.2.2 Verification Program Tailoring

The environmental test requirements are written assuming a low-risk program. The environmental program should be tailored to reflect the hardware classification, mission objectives, hardware characteristics such as physical size and complexity, and the level of risk accepted by the project. For example, the "trouble-free-performance" requirement may be varied from the baseline to reflect mission duration and risk acceptance. This document also assumes that the payload/spacecraft is of modular design and can be tested at the unit/component, subsystem/instrument, and system/spacecraft levels of assembly. Often this is not the case. The project must develop a verification program that satisfies the intent of the required verification program while taking into consideration the specific characteristics of the mission and the hardware. For example:

- A spacecraft subsystem, or instrument, may be a functional subdivision of the spacecraft, but it may be distributed throughout the spacecraft rather than being a physical entity. In this case, the environmental tests, and associated functional tests, must be performed at physical levels of assembly (component, section, module, system or instrument [refer to Appendix A - hardware level of assembly]) that are appropriate for the specific hardware. Performance tests and calibrations may still be performed on the functional subsystem or instrument.

- The physical size of the system may necessitate testing at other levels of assembly. Facility limitations may not allow certain environmental tests to be performed at the system level. In this case, testing should be performed at the highest practicable level. Also, for very large systems or subsystems/instruments, tests at additional levels of assembly may be added in order to adequately verify the hardware design, workmanship and/or performance.

- For small payloads, the subsystem level environmental tests may be skipped in favor of testing at the component and system/spacecraft levels. Similarly, for very small instruments the GSFC project may elect to not test all components in favor of testing at the instrument level. These decisions must be made carefully, especially regarding bypassing lower level testing for instruments, because of the increased risk to the program (schedule, cost, etc.) of finding problems late in the planned schedule.

- In some cases, because of the hardware configuration it may be reasonable to test more than one component at a time. The components may be stacked in their flight configuration, and may therefore be tested as a "section". Part of the decision process must consider the physical size and mass of the hardware. The test configuration must allow for adequate dynamic or thermal stress inputs to the hardware to uncover design errors and workmanship flaws.

- Some test requirements stated as subsystem/instrument requirements may be satisfied at a higher level of assembly if approved by the GSFC project. For example, externally induced mechanical shock test requirements may be satisfied at the system level by firing the environment-producing pyro. A simulation of this environment is difficult, especially for large subsystems or instruments.

- Aspects of the design and/or mission may negate certain test conditions to be imposed. For example, if the on-orbit temperature variations are small, less than 5°C, then consideration should be given to waiving the thermal-vacuum cycling at the system, or instrument, level of assembly in favor of increasing the hot and cold dwell times.

The same process must be applied when developing the test plan for an instrument. While guideline testing is required at the instrument component and all-up instrument levels of assembly, additional test levels may be called for because of hardware complexity or physical size.

Table 2.2-1
Flight System Hardware
Levels of Assembly

.

LEVEL OF ASSEMBLY	EXAMPLES
Space System	NASA Spacecraft
Project or Program	TDRS TIROS GOES
Operating System	Operating Space System
Integrated Systems	Integrated Flight System (Spacecraft + Upperstage + Launch Vehicle)
System Segment (Satellite, Payload, Spacecraft, Laboratory, Observatory, Space Vehicle, etc.)	(Spacecraft Bus + Science Payload) Launch Vehicle IUS
Module	Spacecraft Bus Science Payload Payload Fairing
Subsystem	Instrument/Experiment, Structure, Attitude Control, C & DH, Thermal Control, Electrical Power, TT & C, Propulsion
Section (group of units/components not a subsystem)	Electronic Tray or Pallette, Stacked Units/Components Electronic Boxes Mounted on Panel, Solar Array Sections
Unit (Component)	Electronic Box, Gyro Package, Motor, Actuator, Battery, Receiver, Transmitter, Antenna, Solar Panel, Valve Regulator
Subassembly (combines assembly and subassembly)	Assembly (Power Amplifier, Gyroscope) Subassembly (Wire Harness, Loaded Printed Circuit Card)
Part	Resistor, Capacitor, IC, Switch, Connector Bolt, Screw, Gasket, Bracket, Valve Stem

2.2.3 Test Factors/Durations

Test factors/durations for prototype, protoflight, and acceptance are given in Table 2.2-2. While the acceptance test margin is provided, the test may or may not be required for a specific mission.

Table 2.2-2
Test Factors/Durations

Test	Prototype (Qual.)	Protoflight (Qual.)	Acceptance
Structural Loads[1] Test Level Analysis (show positive margins for all ultimate failure modes)	1.25 x Limit Load 1.4 x Limit Load	1.25 x Limit Load 1.4 x Limit Load	1.0x Limit Load 1.4 x Limit Load
Acoustics Level[2] Duration	Limit Level + 3dB 2 minutes	Limit Level + 3dB 1 minute	Limit Level 1 minute
Random Vibration Level[2] Duration	Limit Level + 3dB 2 minutes/axis	Limit Level + 3dB 1 minute/axis	Limit Level 1 minute/axis
Sine Vibration[3] Level Sweep Rate	1.25 x Limit Level 2 oct/min	1.25 x Limit Level 4 oct/min	Limit Level 4 oct/min
Acceleration (Centrifuge) Level Duration	1.25 x Limit Level 1 minute	1.25 x Limit Level 30 seconds	Limit Level 30 seconds
Mechanical Shock Actual Device Simulated	2 actuations 1.4 x Limit Level 2 x Each Axis	2 actuations 1.4 x Limit Level 1 x Each Axis	1 actuations Limit Level 1 x Each Axis
Thermal-Vacuum	Max./min. predict. ± 10°C	Max./min. predict. ± 10°C	Max./min. predict.
Thermal Cycling[4]	Max./min. predict. ± 15°C	Max./min. predict. ± 15°C	Max./min. predict. ± 5°C
EMC & Magnetics	As Specified for Mission	Same	Same

1 - If qualified by analysis only, positive margins must be shown for load factors of 2.0 on yield and 2.6 on ultimate. Composite materials cannot be qualified by analysis alone.

Note: Test and Analysis levels for beryllium structure are 1.4 x Limit Level for both qualification and acceptance testing, and 1.6 x Limit Level for analysis on ultimate. Also composite structure, including metal matrix, requires acceptance testing to 1.25 x Limit Level.

2 - As a minimum, the test level shall be equal to or greater than the workmanship level.

3 - The sweep direction should be evaluated and chosen to minimize the risk of damage to the hardware. If a sine sweep is used to satisfy the loads or other requirements, rather than to simulate an oscillatory mission environment, a faster sweep rate may be considered, e.g., 6-8 oct/min to reduce the potential for over stress.

4 - It is recommended that the number of thermal cycles be increased by 50% for thermal cycle (ambient pressure) testing.

SECTION 2.3

ELECTRICAL FUNCTION & PERFORMANCE

2.3 ELECTRICAL FUNCTION TEST REQUIREMENTS

The following paragraphs describe the required electrical functional and performance tests that verify the payload's operation before, during, and after environmental testing. These tests along with all other calibrations, functional/performance tests, measurements/ demonstrations, alignments (and alignment verifications), end-to-end tests, simulations, etc., that are part of the overall verification program shall be described in the System Performance Verification Plan.

2.3.1 Electrical Interface Tests

Before the integration of an assembly, component, or subsystem into the next higher hardware assembly, electrical interface tests shall be performed to verify that all interface signals are within acceptable limits of applicable performance specifications.

Prior to mating with other hardware, electrical harnessing shall be tested to verify proper characteristics; such as, routing of electrical signals, impedance, isolation, and overall workmanship.

2.3.2 Comprehensive Performance Tests

A comprehensive performance test (CPT) shall be conducted on each hardware element after each stage of assembly: component, subsystem and payload. When environmental testing is performed at a given level of assembly, additional comprehensive performance tests shall be conducted during the hot and cold extremes of the temperature or thermal-vacuum test for both maximum and minimum input voltage, and at the conclusion of the environmental test sequence, as well as at other times prescribed in the verification plan, specification, and procedures.

The comprehensive performance test shall be a detailed demonstration that the hardware and software meet their performance requirements within allowable tolerances. The test shall demonstrate operation of all redundant circuitry and satisfactory performance in all operational modes within practical limits of cost, schedule, and environmental simulation capabilities. The initial CPT shall serve as a baseline against which the results of all later CPTs can be readily compared.

At the payload level, the comprehensive performance test shall demonstrate that, with the application of known stimuli, the payload will produce the expected responses. At lower levels of assembly, the test shall demonstrate that, when provided with appropriate inputs, internal performance is satisfactory and outputs are within acceptable limits.

2.3.3 Limited Performance Tests

Limited performance tests (LPT) shall be performed before, during, and after environmental tests, as appropriate, in order to demonstrate that functional capability has not been degraded by the tests. The limited tests are also used in cases where comprehensive performance testing is not warranted or not practicable. LPTs shall demonstrate that the performance of selected hardware and software functions is within acceptable limits. Specific times when LPTs will be performed shall be prescribed in the verification specification.

2.3.4 Performance Operating Time and Failure-Free Performance Testing

One-thousand (1000) hours of operating/power-on time should be accumulated on all flight electronic hardware, and spares prior to launch.

In addition, at the conclusion of the performance verification program, payloads shall have demonstrated failure-free performance testing for at least the last 350 hours of operation. The demonstration may be conducted at the subsystem level of assembly when payload integration is accomplished at the launch site and the 350-hour demonstration cannot practicably be accomplished on the integrated payload. Failure-free operation during the thermal-vacuum test exposure is included as part of the demonstration with 100 hours of the trouble-free operation being logged at the hot-dwell temperatures and 100 hours being logged at the cold-dwell temperature. Major hardware changes during or after the verification program shall invalidate previous demonstration.

The general intent of the above requirements is to accumulate 1000 hours of operating time on all flight hardware, and to demonstrate trouble-free performance at high-, low-, and nominal temperature. However, it is understood that under certain conditions this goal may not be met. For example hardware change-out just prior to launch may not provide sufficient time to demonstrate these requirements. Also, the retest requirements following component failure during system level thermal vacuum, or other tests, must be evaluated on a case-by-case basis taking into account the criticality of the hardware element and the risk impact on achieving mission goals.

The guideline time requirements should be tailored up or down to reflect hardware classification, and mission duration.

2.3.5 Limited-Life Electrical Elements

A life test program shall be considered for electrical elements that have limited lifetimes. The verification plan shall address the life test program, identifying the electrical elements that require such testing, describing the test hardware that will be used, and the test methods that will be employed.

SECTION 2.4

STRUCTURAL AND MECHANICAL

2.4 STRUCTURAL AND MECHANICAL VERIFICATION REQUIREMENTS

A series of tests and analyses shall be conducted to demonstrate that the flight hardware is qualified for the expected mission environments and that the design of the hardware complies with the specified verification requirements such as factors of safety, interface compatibility, structural reliability, workmanship, and associated elements of system safety.

Table 2.4-1 specifies the structural and mechanical verification activities. When the tests and analyses are planned, consideration must be given to the expected environments of structural loads, vibroacoustics, sine vibration, mechanical shock, and pressure profiles induced during all phases of the mission; for example, during launch, insertion into final orbit, preparation for orbital operations, and STS (or Pegasus carrier aircraft) descent and landing. Verification must also be accomplished to ensure that the transportation and handling environments are enveloped by the expected mission environments. Mass properties and proper mechanical functioning shall also be verified.

Of equal importance with qualifying the hardware for expected mission environments are the testing for workmanship and structural reliability, which are intended to provide a high probability of proper operation during the mission. In some cases, the expected mission environment is rather benign and produces test levels insufficient to expose workmanship defects. The verification test must envelope the expected mission levels, with appropriate margins added for qualification, and impose sufficient stress to detect workmanship faults. Flight load and dynamic environment levels are probabilistic quantities. Selection of probability levels for flight limit level loads/environments to be used for payload design and testing is the responsibility of the payload project manager, but in no event shall the probability levels be less than the minimum levels in Table 2.4-2. Specific structural reliability requirements regarding fracture control for STS and ELV payloads, beryllium structure, composite structure, bonded structural joints, and glass structural elements are given in 2.4.1.4.

The program outlined in Table 2.4-1 assumes that the payload 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 be accomplished at a higher or lower level of assembly. For example, structural load tests of some components may be necessary if they cannot be properly applied during testing at higher levels of assembly.

Ground handling, transportation and test fixtures shall be analyzed and tested for proper strength as required by safety, and shall be verified for stability for applicable configurations as appropriate.

2.4.1 Structural Loads Qualification

Qualification of the payload for the structural loads environment requires a combination of test and analysis. A test-verified finite element model of the payload must be developed and a coupled loads analysis of the payload/launch vehicle (STS or ELV) performed.

The analytical results define the limit loads for the payload (subsystems and components) and show compatibility with the launch vehicle for all critical phases of the mission. If the payload is to be launched on an ELV but retrieved and returned by STS, analyses must be performed to determine limit loads and compatibility with both vehicles.

TABLE 2.4-1
Structural and Mechanical Verification Test Requirements

Requirement	Payload/ Spacecraft	Subsystem/ Instrument	Unit (Component) Including Instrument Units (Components)
Structural Loads Modal Survey Design Qualification Structural Reliability Primary & Secondary Structure	 * * *	 T A,T/A^1 (A,T)^1	 * * *
Vibroacoustics Acoustics Random Vibration	 T T^2	 T^2 T^2	 T^2 T
Sine Vibration	T^3,T^4	T^3,T^5	T^3,T^6
Mechanical Shock	T	T^7	-
Mechanical Function	A,T	A,T	-
Pressure Profile	-	A,T^2	A
Mass Properties	A/T	A,T^2	*

* = May be performed at payload or component level of assembly if appropriate.

A = Analysis required.

T = Test required.

A/T = Analysis and/or test.

A,T/A^1 = Analysis and Test or analysis only if no-test factors of safety given in 2.4.1.1.1 are used.

(A,T)^1 = Combination of fracture analysis and proof tests on selected elements, with special attention given to beryllium, composites, and bonded joints.

T^2 = Test must be performed unless assessment justifies deletion.

T^3 = Test performed to simulate any sustained periodic mission environment, or to satisfy oth er requirement (loads, low frequency transient vibration).

T^4 = Test must be performed for ELV payloads, if practicable , to simulate transient and any sustained periodic vibration mission environment.

T^5 = Test must be performed for ELV payload instrument s and for ELV payload subsystems if not performed at payload level of assembly due to test facility limitations ; to simulate sine transient and any sustained periodic vibration mission environment.

T^6 = Test must be performed for ELV payload, instruments, and components to simulate sine transient and any sustained periodic vibration mission environment.

T^7 = Test required for self-induced shocks, but may be performed at payload level of assembly for externally induced shocks.

TABLE 2.4-2
Minimum Probability-Level Requirements
for Flight Limit (maximum expected) Level

Requirement	Minimum Probability Level	
	STS Payloads	ELV Payloads
Structural Loads	99.87/50 (1),(2)	97.72/50 (2),(3)
Vibroacoustics Acoustics Random Vibration	95/50 (4)	95/50
Sine Vibration	99.87/50 (2),(5)	97.72/50 (2)
Mechanical Shock	95/50	95/50

Notes:

(1) 99.87% probability of not exceeding level, estimated with 50% confidence. Equal to the mean plus three-sigma level for normal distributions.

(2) When parametric statistical methods are used to determine the limit level, the data should be tested to show a satisfactory fit to the assumed underlying distribution.

(3) 97.72% probability of not exceeding level, estimated with 50% confidence. Equal to the mean plus two-sigma level for normal distributions.

(4) Equal to, or greater than, the ninety-fifth percentile value, estimated with 50% confidence.

(5) Sine vibration applies to STS payloads only if required to simulate sustained periodic environment from upper stages or apogee motors, etc..

A modal test shall be performed for each payload (at the subsystem/instrument or other appropriate level of assembly) to verify that the analytical model adequately represents the dynamic behavior of the hardware. The test-verified model shall then be used to predict the maximum expected load for each critical loading condition, including handling and transportation, vibroacoustic effects during lift-off, insertion into final orbit, orbital operations, thermal effects during landing, etc., as appropriate for the particular mission. If the payload configuration is different for various phases of the mission, the structural loads qualification program, including the modal survey, must consider the different configurations. The maximum loads resulting from the analysis define the limit loads.

The launch loads environment is made up of a combination of steady-state, low-frequency transient, and higher-frequency vibroacoustic loads. To determine the combined loads for any phase of the launch the root-sum-square (RSS) of the low- and high-frequency dynamic components are superimposed upon the steady-state component.

$$N_i = S_i \pm [(L_i)^2 + (R_i)^2]^{1/2}$$

where N_i, S_i, L_i, and R_i are the combined load factor, steady-state load factor, low-frequency dynamic load factor, and high-frequency random vibration load factor, respectively, for the i'th axis. In some cases, the steady-state and low-frequency dynamic load factors are combined into a low-frequency transient load factor A_i. In this case, the steady-state value must be separated out before the RSS operation.

As an example: For the STS lift-off there is negligible steady-state acceleration in the Y and Z directions; all the load factors in these directions are vibrational. However, the STS X-axis load factor contains approximately 1.5 g's due to the steady-state lift-off acceleration. This steady-state acceleration (a negative quantity) must be removed from the RSS operation and added algebraically:

$$N_{xmax} = -1.5 + [(A_{xmax} + 1.5)^2 + (R_x)^2]^{1/2}$$

$$N_{xmin} = -1.5 - [(A_{xmin} + 1.5)^2 + (R_x)^2]^{1/2}$$

$$N_y = \pm [(A_y)^2 + (R_y)^2]^{1/2}$$

$$N_z = \pm [(A_z)^2 + (R_z)^2]^{1/2}$$

The resulting N_x, N_y, and N_z must then be considered to be acting simultaneously and in all combinations. The above combination procedure may be extended to forces or stresses by replacing the load factors with the appropriate forces or stresses produced by those load factors.

The maximum load at landing for the STS shall be considered to be a combination of the low-frequency transient landing loads and the thermally induced loads. These load environments shall be obtained by combining the worst-case combination of the low-frequency transient landing loads in the X, Y, and Z axes simultaneously with the thermally induced loads.

Also included in the STS liftoff and landing loads are contributions from trunnion friction, and trunnion misalignment loads due to lack of trunnion interface planarity.

Other STS environments, such as ascent and descent quasi-static loads, emergency landing, RMS operations, berthing, on-orbit OMS/RCS firing during repair and maintenance missions, must also be investigated as potential design drivers.

When determining the limit loads for ELV launches, consideration must be given to the timing of the loading events; the maximum steady state and dynamic events occur at different times in the launch and may provide too conservative an estimate if combined. Also, the frequency band of the vibroacoustic energy to be combined must be evaluated on a case-by-case basis. Flight events which must be considered for inclusion in the coupled loads analysis for various ELV's are listed in Table 2.4-3. If the verification cycle analysis or payload test-verified model is not available, the latest analytical data should be used in conjunction with a suitable uncertainty factor.
Each subsystem/instrument shall then be qualified by loads testing to 1.25 times the limit loads defined above. The loads test shall be accompanied by stress analysis showing

positive margins of safety at 1.4 times the limit load for all ultimate failure modes such as fracture or buckling. In some cases, qualification by analysis may be allowed (see 2.4.1.3). Special design and test factors of safety are required for beryllium structure (see 2.4.1.3.1).

2.4.1.1 Coupled load analysis - A coupled load analysis, combining the launch vehicle and payload, shall be performed to support the verification of positive stress margins and sufficient clearances during the launch.

2.4.1.1.1 Analysis - Strength Verification - A finite element model shall be developed (and verified by test) that analytically simulates the payload's mass and stiffness characteristics, for the purpose of performing a coupled loads analysis. The model shall be of sufficient detail to make possible an analysis that defines the payload's modal frequencies and displacements below a specified frequency that is dependent on the fidelity of the launch vehicle finite element model. For the STS, all significant modes below 50 Hz and for ELV all significant modes below 70 Hz are sufficient unless higher-frequency modes are required by the launch vehicle manufacturer.

The model is then coupled with the model of the STS or ELV and any upper-stage propulsion system. The combined coupled model is used to conduct a coupled loads analysis that evaluates all potentially critical loading conditions. Forcing functions used in the coupled loads analysis shall be defined at the flight limit level consistent with the minimum probability levels of Table 2.4-2. The results of the coupled loads analysis shall be reviewed to determine the worst-case loads. These constitute the set of limit loads that are used to evaluate member loads and stresses.

For STS payloads, the analysis shall include estimates of loads induced by effects such as trunnion friction, trunnion non-planarity, vibroacoustics at lift-off and thermal environments during the STS landing. In addition, if the hardware is intended for multiple flights or if the design is intended for multiple applications, variations in configuration or other parameters that may influence the maximum load shall be considered in the analysis.

For ELV payloads, the coupled loads analysis shall consider the flight events listed in Table 2.4-3, which gives events processed for some ELVs, plus any other events recommended by the ELV organization. None of the flight events listed in Table 2.4-3 shall be deleted from the coupled loads analysis unless it is shown by base drive analysis of the cantilevered spacecraft and adapter that there are no significant spacecraft vibration modes in frequency bands of significant launch vehicle forcing functions and coupled-mode responses. For example, it should be confirmed that there are no spacecraft structural components or subsystems (upper platforms, antenna supports, scientific instruments, etc.) which can experience high dynamic responses during flight events such as lift-off or sustained, pogo-like oscillations before deleting these events. For the evaluation of flight events to include in the coupled loads analysis, an appropriate tolerance should be applied to all potentially significant spacecraft modal frequencies unless verified by modal survey testing.

Normally, the design and verification of payloads shall not be burdened by transportation and handling environments that exceed stresses expected during launch, orbit, or return. Rather, shipping containers shall be designed to prevent the imposition of such stresses. To verify this, a documented analysis shall be prepared on shipping and handling equipment to define the loads transmitted to flight hardware. When transportation and handling loads are not enveloped by the maximum expected flight loads, the transportation and handling loads shall be included in the set of limit loads.

For those hardware items that will later be subjected to a strength qualification test, a stress analysis shall be performed to provide confidence that the risk of failing the strength test is

TABLE 2.4-3
ELV Flight Event Loading Conditions to Consider for Coupled Loads Analysis
of Combined Payload and Launch Vehicle

ELV	Flight Event Loading Conditions*
Atlas I, II, IIA, and IIAS	Launch (liftoff) Transonic Flight Winds Pogo (prior to BECO) BECO/BPJ MECO (final MECO)
Delta II (all series)	Liftoff Transonic Max Q First Pre-MECO Second Pre-MECO Prior to MECO MECO
Titan II	Liftoff Max. Airloads Stage I burnout Stage II Shutdown
Titan III and Titan IV	Liftoff Max. Buffet (transonic) Max. Air loads (Max. Qα) Stage I Burnout Stage II Shutdown
Pegasus (including XL version)	Taxi and Captive Flight Drop Transient Aerodynamic Pull-up First, Second, and Third Stage Burn-out Abort Landing

* Minimum list of conditions which must be considered; the launch vehicle organization should be consulted regarding any recommended additional conditions to consider. The significance of the various loading conditions may vary with the payload weight and dynamic characteristics. For ELVs not listed above, consult the launch vehicle organization for the flight events that are considered during their coupled loads analyses.

small and to demonstrate compliance with the launch vehicle (STS or ELV) interface verification and safety requirements. The analysis shall show positive margins at stresses corresponding to a loading of 1.4 times the limit load for all ultimate failure modes such as fracture or buckling. In addition, the analysis shall show that for a loading equal to the limit load, the maximum allowable loads at the STS interface points (or ELV flight adapter) are not exceeded, that no detrimental permanent deformations will occur, and that no excessive deformations occur that might constitute a hazard to the launch vehicle (or its crew). See 2.4.1.4 for special requirements for beryllium structure.

For payloads, or payload elements, whose strength is qualified by analysis, the objective of the stress analysis is to demonstrate with a high degree of confidence that there is essentially no chance of failure during flight. For all elements that are to be qualified by analysis, positive strength margins on yield shall be shown to exist at stresses equal to 2.0 times those induced by the limit loads, and positive margins on ultimate shall be shown to exist at stresses equal to 2.6 times those induced by the limit loads. For exceptions, see 2.4.1.3. When qualification by analysis is used, the upper frequency of the modal survey may have to be increased. In addition, at stresses equal to the limit load, the analysis shall show that the maximum allowable loads at the STS interface points (or ELV flight adapter) are not exceeded, that no detrimental permanent deformations will occur, and that no excessive deformations occur that might constitute a hazard to the launch vehicle (or its crew).

2.4.1.1.2 Analysis - Clearance Verification - Analysis shall be conducted for all STS and ELV payloads to verify adequate dynamic clearances between the payload and launch vehicle and between members within the payload for all significant ground test and flight conditions.

a. During Powered Flight - The coupled loads analysis shall be used to verify adequate clearances during flight within the STS cargo bay or ELV payload fairing. One part of the coupled loads analysis output transformation matrices shall contain displacement data that will allow calculation of loss of clearance between critical extremities of the payload and adjacent surfaces of the STS or ELV. For ELV payloads, the analysis shall consider clearances between the payload and ELV payload fairing (and its acoustic blankets if used, including blanket expansion due to venting) and between the payload and ELV attach fitting, as applicable. For the clearance calculations the following factors shall be considered:

   1. Worst-case payload and vehicle manufacturing and assembly tolerances as derived from as-built engineering drawings.

   2. Worst-case payload/vehicle integration "stacking" tolerances related to interface mating surface parallelism, perpendicularity and concentricity, plus bolt positional tolerances, ELV payload fairing ovality, etc.

   3. Quasi-static and dynamic flight loads, including coupled steady-state and transient sinusoidal vibration, vibroacoustics and venting loads, as applicable. Typically, either liftoff or the transonic buffet and maximum airloads cause the greatest relative deflections between the vehicle and payload.

b. During ELV Payload Fairing Separation - A fairing separation analysis based on ground separation test of the fairing, shall be used to verify adequate clearances between the separating fairing sections and payload extremities. Effects of fairing section shell-mode oscillations, fairing rocking, vehicle residual rates, transient coupled-mode oscillations, thrust accelerations, and vehicle control-jet firings shall be considered, as applicable.

c. During Payload Separation - A payload separation analysis shall be used to verify adequate clearances between the payload and the STS or ELV during separation. The analysis shall include effects of factors such as vehicle residual rates, forces and impulses imparted by the separation system (including lateral impulses due to separation clampbands) and vehicle retro-rocket plumes impinging on the payload, as applicable. The same analysis should be utilized to verify acceptable payload separation velocity and tip-off rates if required

Analysis shall also be performed to verify adequate critical dynamic clearances between members within the payload during ground vibration and acoustic testing, and flight. Additionally, a deployment analysis shall be used to verify adequate clearances during payload appendage deployment. Refer to 2.4.5.2 regarding mechanical function clearances.

For all of the above clearance analyses and conditions, adequate clearances shall be verified assuming worst-case static clearances due to manufacturing, assembly and vehicle integration tolerances (unless measured on the launch stand), and quasi-static and dynamic deflections due to 1.4 times the applicable flight limit loads or flight-level ground test levels. Depending on the available static clearance, the clearance analysis requirements may be satisfied in many cases by simple worst-case estimates and/or similarity.

2.4.1.2 Modal Survey - A modal survey test will be required for payloads and subsystems, including instruments, that do not meet requirements on minimum fundamental frequency. The minimum fundamental frequency requirement is dependent on the launch vehicle and is discussed below for STS and ELV launch vehicles. In order to determine if the hardware meets the frequency requirement, an appropriate test, or tests, shall be performed to identify the fundamental frequency. A low level sine survey is generally an appropriate method for determining the fundamental frequency.

For STS, a modal test is required if the subsystem/instrument resonances are not above 50 Hz. For an ELV, the frequency below which a modal test is required is dependent on the specific launch vehicle. The determination will be made on a case-by-case basis and specified in the design and test requirements. Modal tests are generally performed at the subsystem/instrument level of assembly, but may be required at other levels of assembly such as the payload or component level depending on project requirements.

In general, the support of the hardware during the test shall duplicate the boundary conditions expected during launch. When that is not feasible, other boundary conditions are employed and the frequency limits of the test are adjusted accordingly. The effects of interface flexibilities should be considered when other than normal boundary conditions are used.

The results of the modal survey are required to identify any inaccuracies in the mathematical model used in the payload analysis program so that modifications can be made if needed. Such an experimental verification is required because a degree of uncertainty exists in unverified models owing to assumptions inherent in the modeling process. These lead to uncertainties in the results of the flight dynamic loads analysis, thereby reducing confidence in the accuracy of the set of limit loads derived therefrom.

If a modal survey test is required, all significant modes up to the required frequency must be determined both in terms of frequency and mode shape. Cross-orthogonality checks of the test and analytical mode shapes, with respect to the analytical mass matrix, shall be performed with the goal of obtaining at least 0.9 on the diagonal and no greater than 0.1 off-diagonal. Any test method that is capable of meeting the test objectives with the necessary accuracy may be used to perform the modal survey. The input forcing function may be transient, fixed frequency, swept sinewave, or random in nature.

When a satisfactory modal survey has been conducted on a representative structural model, a modal survey of the protoflight unit may be unnecessary. A representative structural model
is defined as one that duplicates the structure as to materials, configuration, fabrication, and assembly methods and that satisfactorily simulates other items that mount on the structure as to location, method of attachment, weight, mass properties, and dynamic characteristics.

2.4.1.3 Design Strength Qualification - The preferred method of verifying adequate strength is to apply a set of loads equal to 1.25 times the limit loads, after which the hardware must be capable of meeting its performance criteria (see 2.4.1.3.1 for special requirements for beryllium structure). As many test conditions shall be applied as necessary to subject the hardware to the worst-case loads. No detrimental permanent deformation shall be allowed to occur as a result of applying the loads, and all applicable alignment requirements must be met following the test.

The strength qualification test must be accompanied by a stress analysis that demonstrates a positive margin on ultimate at loads equal to 1.4 times the limit load for all ultimate failure modes such as fracture or buckling. See 2.4.1.3.1 for special requirements for beryllium structure.

In addition, the analysis shall show that at stresses equal to the limit load, the maximum allowable loads at the launch vehicle interface points are not exceeded and that no excessive deformations occur that might constitute a hazard to the mission. This analysis shall be performed prior to the start of the strength qualification tests to provide minimal risk of damage to hardware. When satisfactory qualification tests have been conducted on a representative structural model, the strength qualification testing of the protoflight unit may not be necessary.

a. Selection of Test Method - The qualification load conditions may be applied by acceleration testing, static load testing, or vibration testing (either transient, fixed frequency or swept sinusoidal excitation). Random vibration is generally not acceptable for loads testing.

The following questions shall be considered when the method to be employed for verification tests is selected:

(1) Which method most closely approximates the flight-imposed load distribution?

(2) Which can be applied with the greatest accuracy?

(3) Which best provides information for design verification and for predicting design capability for future payload or launch vehicle modifications?

(4) Which poses the least risk to the hardware in terms of handling and test equipment?

(5) Which best stays within cost, time, and facility limitations?

b. Test Setup - The subsystem/instrument shall be attached to the test equipment by a fixture whose mechanical interface simulates the mounting of the subsystem/instrument into the payload with particular attention paid to duplicating the actual mounting contact area. In mating the subsystem to the fixture, a flight-type mounting (including vibration isolators or kinematic mounts if part of the design) and fasteners shall be used.

Components that are normally sealed shall be pressurized during the test to their prelaunch pressure. In cases when significant changes in strength, stiffness, or applied load result from variations in internal and external pressure during the launch phase, a special test shall be considered to cover those effects.

When acceleration testing is performed, the centrifuge shall be large enough so that the applied load at the extreme ends of the test item does not differ by more than 10 percent from that applied to the center of gravity. In addition, when the proper orientation for the applied acceleration vector is computed, ambient gravity effects shall be considered.

c. Performance - Before and after the strength qualification test, the subsystem/ instrument shall be examined and functionally tested to verify compliance with all performance criteria. During the tests, performance shall be monitored in accordance with the verification specification and procedures.

If appropriate development tests are performed to verify accuracy of the stress model, stringent quality control procedures are invoked to ensure conformance of the structure (materials, fasteners, welds, processes, etc.) to the design, and the structure has well-defined load paths, then strength qualification may (with payload project concurrence) be accomplished by a stress analysis that demonstrates that the hardware has positive margins on yield at loads equal to 2.0 times the limit load, and positive margin on ultimate at loads equal to 2.6 times the limit load. Factors of safety lower than 2.0 on yield and 2.6 on ultimate will be considered when they can be shown to be warranted. Justification for the lower factors of safety must be based on the merits of a particular combination of test and analysis and a correlation of the two. Such alternative approaches shall be reviewed and approved on a case-by-case basis. In addition, at stresses equal to the limit load, the analysis shall show that the maximum allowable loads at the launch vehicle interface points are not exceeded and that no excessive deformations occur.

Structural elements fabricated from composite materials or beryllium shall not be qualified by analysis alone.

2.4.1.3.1 Strength Qualification - Beryllium - All beryllium primary and secondary structural elements shall undergo a strength test to 1.4 times limit load. No detrimental permanent deformation shall be allowed to occur as a result of applying the loads, and applicable alignment requirements must be met following the test. In addition:

a. When using cross-rolled sheet, the design shall preclude out-of-plane loads and displacements during assembly, testing, or service life.

b. In order to account for uncertainties in material properties and local stress levels, a design factor of safety of 1.6 on ultimate material strength shall be used.

c. Stress analysis shall properly account for the lack of ductility of the material by rigorous treatment of applied loads, boundary conditions, assembly stresses, stress concentrations, thermal cycling, and possible material anisotropy. The stress analysis shall take into account worst-case tolerance conditions.

d. All machined and/or mechanically disturbed surfaces shall be chemically milled to ensure removal of surface damage and residual stresses.

e. All parts shall undergo penetrant inspection for surface cracks and crack-like flaws per MIL-STD-6866.

2.4.1.4 Structural Reliability (Residual Strength Verification) - Structural reliability requirements are intended to provide a high probability of the structural integrity of all flight hardware. They are generally covered by the selection of materials, process controls, selected analyses (stress, and fracture mechanics/crack growth), and loads/proof tests.

All structural materials contain defects such as inclusions, porosity, and cracks. To ensure that adequate residual strength (strength remaining after the flaws are accounted for) is present for structural reliability at launch, a fracture control program, or a combination of fracture control and specific loads tests, shall be performed on all flight hardware as specified below.

The use of materials that are susceptible to brittle fracture or stress-corrosion cracking require development of, and strict adherence to, special procedures to prevent problems. If materials are used for structural application that are not listed in Table 1 of MSFC-SPEC-522, a Materials Usage Agreement (MUA) must be negotiated with the project office. Refer to project Materials and Processes Control Requirements for applicable requirements.

2.4.1.4.1 Primary and Secondary Structure:

STS and ELV Payloads - The following requirements regarding beryllium, nonmetallic-composite, and metallic-honeycomb structural elements (both primary and secondary), and bonded structural joints apply to both STS and ELV payloads:

a. Beryllium Primary and Secondary Structure: The requirements of section 2.4.1.3.1, Strength Verification-Beryllium, apply for structural reliability.

b. Nonmetallic Composite Structural Elements (including metal matrix): All flight structural elements shall be proof tested to 1.25 times limit load (even if previously qualified on valid prototype hardware). In addition:

(1) A process control plan shall be developed and implemented to ensure uniformity of processing among test coupons, test articles, and flight hardware as required by the project Materials and Processes Control Requirements.

(2) A damage control plan shall be implemented to establish procedures and controls to prevent and/or identify nonvisible impact damage which may cause premature failure of composite elements.

c. Metallic Honeycomb (both facesheets and core) Structural Elements:

(1) Appropriate process controls and coupon testing shall be implemented to demonstrate that the honeycomb structure is acceptable for use as payload flight structure as required by the project Materials and Processes Control Requirements.

(2) Metallic honeycomb is not considered to be a composite material.

d. Bonded Structural Joints (either metal-metal or metal-nonmetal):

(1) Every bonded structural joint in a flight article shall be proof tested (by static loads test) to 1.25 times limit load. For example, proof loads testing shall be performed to demonstrate that inserts will not tear out from honeycomb under protoflight loads.

(2) A process control plan shall be developed and implemented as required by applicable project Materials and Processes Control Requirements to ensure uniformity of processing among test coupons, test articles, and flight hardware.

STS Payloads - For payloads to be launched, serviced and/or retrieved by the STS, structural reliability requirements are completely covered by the STS safety and materials process control requirements. A mandatory fracture control program is instituted as part of the system safety requirements and is implemented in accordance with the following documents:

a. GSFC 731-0005-83, General Fracture Control Plan for Payloads Using the STS.

b. JSC letter TA-92-013 (dated June 29, 1992) regarding "low risk fracture parts" in STS 18798A, "Interpretations of NSTS Payload Safety Requirements."

Each STS payload organization must submit certification to the STS safety review board that beryllium is not used in a safety-critical application. NSTS reviews the project's structural certification plan for all beryllium structure flown on the orbiter. All safety provisions apply in accordance with the appropriate NSTS safety requirements documentation.

Also, metallic honeycomb (both facesheets and core) flight structure shall be proof tested to 1.25 times limit load. Metallic honeycomb is not considered to be composite structure. This requirement does not apply to solar array panels which do not support any significant mounted component weight.

ELV Payloads - If the payload is to be placed in orbit by an ELV, fracture control requirements (per GSFC 731-0005-83) shall apply to the following elements only:

a. Pressure vessels, dewars, lines, and fittings (per NHB-8071.1),

b. Castings (unless hot isostatically pressed and the flight article is proof tested to 1.25 times limit load),

c. Weldments,

d. Parts made of materials on Tables II or III of MSFC-SPEC-522B if under sustained tensile stress. (Note: All structural applications of these materials requires that a Materials Usage Agreement (MUA) must be negotiated with the project office; refer to project Materials and Processes Control Requirements,

e. Parts made of materials susceptible to cracking during quenching,

f. Nonredundant, mission-critical preloaded springs loaded to greater than 25 percent of ultimate strength.

All glass elements, that are stressed above 10% of their ultimate tensile strength, shall also be shown by fracture analysis to satisfy "Safe-life" or "Fail-safe" conditions or be subjected to a proof loads test at 1.0 times limit level.

2.4.1.5 Acceptance Requirements - All of the structural reliability requirements of 2.4.1.4 (as specified for STS and/or ELV payloads) apply for the acceptance of all flight hardware.

Generally, structural design loads testing is not required for flight structure that has been previously qualified for the current mission as part of a valid prototype or protoflight test. However, the following acceptance/proof loads tests are required unless equivalent load-level testing was performed on the actual flight hardware as part of a protoflight test program:

a. For Both STS and ELV Payloads

(1) Beryllium structure (primary and secondary) shall be proof tested to 1.4 times limit load.

(2) Nonmetallic composites (including metal matrix) structural elements shall be proof tested to 1.25 times limit load.

(3) Bonded structural joints shall be proof tested (by static loads test) to 1.25 times limit load.

b. For STS Payloads Only

(1) Any proof loads testing imposed by STS safety shall be performed

(2) Metallic honeycomb shall be proof tested to 1.25 times limit load.

If a follow-on spacecraft receives structural modifications or a new complement of instruments, it must be requalified for the loads environment if analysis so indicates.

2.4.2 Vibroacoustic Qualification

Qualification for the vibroacoustics environment generally requires an acoustics test at the payload level of assembly and random vibration tests on all components, instruments, and on the payload, when appropriate, to better simulate the structure borne inputs. In addition, random vibration tests shall be performed on all subsystems unless an assessment of the expected environment indicates that the subsystem will not be exposed to any significant vibration input. Similarly, an acoustic test shall be performed on subsystems/instruments and components unless an assessment of the hardware indicates that they are not susceptible to the expected acoustic environment or that testing at higher levels of assembly provides sufficient exposure at an acceptable level of risk to the program. Irrespective of the above stated conditions, these additional tests may be required to satisfy delivery requirements.

It is understood that for some payload projects, the vibroacoustic qualification program may have to be modified. For example, for very large payloads it may be impracticable because of test facility limitations to perform testing at the required level of assembly. In that case, testing at the highest practicable level of assembly should be performed, and additional tests and/or analyses added to the verification program if appropriate. Also, the risk to the program associated with the modified test program shall be assessed and documented in the System Verification Plan.

Similarly, for very large components, the random vibration tests may have to be supplemented or replaced by an acoustic test. If the component level tests are not capable of inducing sufficient excitation to internal electric, electronic, and electromechanical devices to provide adequate workmanship verification, it is recommended that an environmental stress screening test program be conducted at lower levels of assembly (subassembly or board level).

For the vibroacoustic environment, limit levels shall be used which are consistent with the minimum probability levels of Table 2.4-2. The protoflight qualification level is defined as the flight limit level plus 3 dB. When random vibration levels are determined, responses to the acoustic inputs plus the effects of vibration transmitted through the structure shall be considered. The random vibration test levels to be used for hardware containing delicate optics, sensors/detectors, etc., may be notched in frequency bands known to be destructive to the hardware with project concurrence. A force-limiting control strategy is recommended. This requires a dual control system which will automatically notch the input so as not to exceed design/expected forces in the area of rigid, shaker mounted resonances while maintaining acceleration control over the remainder of the frequency band. The control methodology must be approved by the GSFC project.

As a minimum, the vibroacoustic test levels shall be sufficient to demonstrate acceptable workmanship.

During test, the test item should be in an operational configuration, both electrically and mechanically, representative of its configuration at lift-off.

The vibroacoustic (acoustics plus random vibration) environmental test program shall be included in the environmental verification plan and environmental verification specification, which are reviewed by the Office of Flight Assurance.

2.4.2.1 Fatigue Life Considerations - The nature of the protoflight test program prevents a demonstration of hardware lifetime because the same hardware is both tested and flown. When hardware reliability considerations demand the demonstration of a specific hardware lifetime, a prototype verification program must be employed, and the test durations must be modified accordingly.

Specifically, the duration of the vibroacoustic exposures shall be extended to account for the life that the flight hardware will experience during its mission. In order to account for the scatter factor associated with the demonstration of fatigue life, the duration of prototype exposures shall be at least four times the intended life of the flight hardware. For ELV payloads, the duration of the exposure shall be based on both the vibroacoustic and sine vibration environments.

If there is the possibility of thermally induced structural fatigue (examples include solar arrays, antennas, etc.), thermal cycle testing shall be performed on prototype hardware. For large solar arrays, a representative smaller qualification panel may be used for test provided that it contains all of the full scale design details (including at least 100 solar cells) susceptible to thermal fatigue. The life test should normally be performed at the worst case (limit level) predicted temperature extremes for a number of thermal cycles corresponding to the required mission life. However, if required by schedule considerations, the test program may be accelerated by increasing the temperature cycle range (and possibly the temperature transition rate) provided that stress analysis shows no unrealistic failure modes are produced by the accelerated testing.

2.4.2.2 Payload Acoustic Test - At the payload level of assembly, protoflight hardware shall be subjected to an acoustic test in a reverberant sound pressure field to verify its ability to survive the lift-off acoustic environment and to provide a final workmanship acoustic test. The test specification is dependent on the payload-launch vehicle configuration and must be determined on a case-by-case basis. Guideline specifications are given in the appendices. The minimum overall test level should be at least 138 dB. If the test specification derived from the launch vehicle expected environment, including fill-factor, is less than 138 dB, the test profile should be raised to provide a 138 dB test level. The planned test and specification levels shall be confirmed by the launch vehicle program office.

a. Facilities and Test Control - The acoustic test shall be conducted in a reverberant chamber large enough to maintain a uniform sound field at all points surrounding the test item. 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 shall 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 l/4 the wavelength of the lowest frequency of interest is recommended. It is recognized that this cannot be achieved in some facilities, particularly when noise levels are specified to frequencies as low as 25 Hz. In such cases, the microphones shall 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.

b. Test Setup - 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) and 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 lift-off.

c. Performance - Before and after the acoustic exposure, the payload shall be examined and functionally tested. During the test, performance shall be monitored in accordance with the verification specification.

2.4.2.3 Payload Random Vibration Tests - At the payload level of assembly, protoflight hardware shall, when practicable, be subjected to a random vibration test to verify its ability to survive the lift-off environment and also to provide a final workmanship vibration test. For small payloads (<454 kg or 1000 lb), the test is required; for larger payloads the need to perform a random vibration test shall be assessed on a case-by-case basis. Additional qualification tests may be required if expected environments are not enveloped by this test. The acoustic environment at lift-off is usually the primary source of random vibration; however, other sources of random vibration must be considered. The sources include transonic aerodynamic fluctuating pressures and the firing of retro/apogee motors.

a. Lift-Off Random Vibration - Protoflight hardware shall be subjected to a random vibration test to verify flightworthiness and workmanship. The test level shall represent the qualification level (flight limit level plus 3 dB).

The test is intended for payloads (spacecraft) of low to moderate weight and size. For small payloads, such as Pegasus-launched spacecraft, small attached STS payloads, GAS experiments, etc., the test should cover the full 20-2000 Hz frequency range. In such cases, the project should assess and recommend a random vibration test, acoustic test, or both, depending on the payload. For larger STS payloads, the test is intended to verify the hardware in the frequency range where acoustic tests do not excite the payload to the levels it will encounter during launch. The test can therefore be limited to this frequency range, reducing the drive requirements of the vibration exciter and easing the design requirements for the "head expander" that is used to adapt the payload to the shaker. For larger ELV payloads, the test is not required unless there is a close-coupled, direct structural load path to the launch vehicle external skin. In that case, both lift-off and transonic random vibration must be considered.

The payload in its launch configuration shall be attached to a vibration fixture by use of a flight-type launch-vehicle adapter and attachment hardware. Vibration shall be applied at the base of the adapter in each of three orthogonal axes, one of which is parallel to the thrust axis. The excitation spectrum as measured by the control accelerometer(s) shall be equalized such that the acceleration spectral density is maintained within $\pm$ 3 dB of the specified level at all frequencies within the test range and the overall RMS level is within $\pm$ 10% of the specified level.

Prior to the payload test, a survey of the test fixture/exciter combination shall be performed to evaluate the fixture dynamics, the proposed choice of control accelerometer locations, and the control strategy. If a mechanical test model of the payload is available it should be included in the survey to evaluate the need for limiting.

If a random vibration test is not performed at the payload level of assembly, the feasibility of doing the test at the next lower level of assembly shall be assessed.

b. Performance - Before and after each vibration test, the payload shall be examined and functionally tested. During the tests, performance shall be monitored in accordance with the verification specification.

2.4.2.4 Subsystem/Instrument Vibroacoustic Tests - If subsystems are expected to be significantly excited by structureborne random vibration, a random vibration test shall be performed. Specific test levels are determined on a case-by-case basis. The levels shall be equal to the qualification level as predicted at the location where the input will be controlled. Subsystem acoustic tests may also be required if the subsystem is judged to be sensitive to this environment or if it is necessary to meet delivery specifications. A random vibration test is generally required for instruments.

2.4.2.5 Component/Unit Vibroacoustic Tests - As a screen for design and workmanship defects, components/units shall be subjected to a random vibration test along each of three mutually perpendicular axes. In addition, when components are particularly sensitive to the acoustic environment, an acoustic test shall be considered.

a. Random Vibration - The test item is subjected to random vibration along each of three mutually perpendicular axes for one minute each. When possible, the component random vibration spectrum shall be based on levels measured at the component mounting locations during previous subsystem or payload testing. When such measurements are not available, the levels shall be based on statistically estimated responses of similar components on similar structures or on analysis of the payload. Actual measurements shall then be used if and when they become available. In the absence of any knowledge of the expected level, the generalized vibration test specification of Table 2.4-4 may be used.

As a minimum, all components shall be subjected to the levels of Table 2.4-5, which represent a workmanship screening test. The minimum workmanship test levels are primarily intended for use on electrical, electronic, and electromechanical hardware.

The test item shall be attached to the test equipment by a rigid fixture. The mounting shall 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 or kinematic mounts, if part of the design) and fasteners should be used. Normally sealed items shall be pressurized during test to their prelaunch pressure.

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 shall be considered to cover those effects.

Prior to the test, a survey of the test fixture/exciter combination shall be performed to evaluate the fixture dynamics, the proposed choice of control accelerometer locations, and the control strategy. The evaluation shall include consideration of cross-axis responses. If a mechanical test or engineering model of the test article is available it should be included in the survey.

For very large components the random vibration tests may have to be supplemented or replaced by an acoustic test if the vibration test levels are insufficient to excite internal hardware. If neither the acoustic nor vibration excitation is sufficient to provide an adequate workmanship test, a screening program should be initiated at lower levels of assembly; down to the board level, if necessary. The need for the screening program must be evaluated by the project. The evaluation is based on mission reliability requirements and hardware criticality, as well as budgetary and schedule constraints.

If testing is performed below the component level of assembly, the workmanship test levels of Table 2.4-5 can be used as a starting point for test tailoring. The intent of testing at this level of assembly is to uncover design and workmanship flaws. The test input levels do not represent expected environments, but are intended to induce failure in weak parts and to expose workmanship errors. The susceptibility of the test item to vibration must be evaluated and the test level tailored so as not to induce unnecessary failures.

b. Acoustic Test - If a component-level acoustic test is required, the test set-up and control shall be in accordance with the requirements for payload testing.

c. Performance - Before and after test exposure, the test item shall be examined and functionally tested. During the test, performance shall be monitored in accordance with the verification specification.

2.4.2.6 Acceptance Requirements - Vibroacoustic testing for the acceptance of previously qualified hardware shall be conducted at flight limit levels using the same duration as recommended for protoflight hardware. As a minimum, the acoustic test level shall be 138 dB, and the random vibration levels shall represent the workmanship test levels.

The payload is subjected to an acoustic test and/or a random vibration test in three axes. Components shall be subjected to random vibration tests in the three axes. Additional vibroacoustic tests at subsystem/instrument and component levels of assembly are performed in accordance with the environmental verification plan or as required for delivery.

Before and after test exposure, the test item shall be examined and functionally tested. During the test, performance shall be monitored in accordance with the verification specification.

Table 2.4-4
Generalized Random Vibration Test Levels
Components (STS or ELV)
22.7-kg (50-lb) or less

Frequency (Hz)	ASD Level (G^2/Hz) Qualification	ASD Level (G^2/Hz) Acceptance
20	0.026	0.013
20-50	+6 dB/oct	+6 dB/oct
50-800	0.16	0.08
800-2000	-6 dB/oct	-6 dB/oct
2000	0.026	0.013
Overall	14.1 G_{rms}	10.0 G_{rms}

The acceleration spectral density level may be reduced for components weighing more than 22.7-kg (50 lb) according to:

	Weight in kg	Weight in lb	
dB reduction	= 10 log(W/22.7)	10 log(W/50)	
$ASD_{(50\text{-}800\ Hz)}$	= 0.16•(22.7/W)	0.16•(50/W)	for protoflight
$ASD_{(50\text{-}800\ Hz)}$	= 0.08•(22.7/W)	0.08•(50/W)	for acceptance

Where W = component weight.

The slopes shall be maintained at + and - 6dB/oct for components weighing up to 59-kg (130-lb). Above that weight, the slopes shall be adjusted to maintain an ASD level of 0.01 G^2/Hz at 20 and 2000 Hz.

For components weighing over 182-kg (400-lb), the test specification will be maintained at the level for 182-kg (400 pounds).

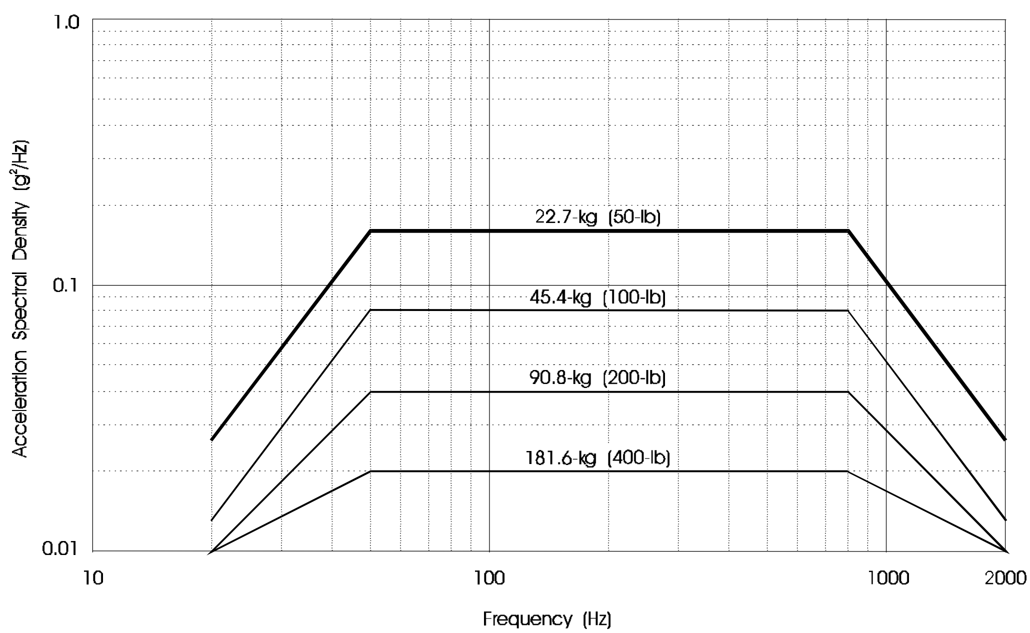


Table 2.4-5
Component Minimum Workmanship
Random Vibration Test Levels
45.4-kg (100-lb) or less

Frequency (Hz)	ASD Level (G^2/Hz)
20	0.01
20-80	+3 dB/oct
80-500	0.04
500-2000	-3 dB/oct
2000	0.01
Overall	6.8 G_{rms}

The plateau acceleration spectral density level (ASD) may be reduced for components weighing between 45.4 and 182 kg, or 100 and 400 pounds according to the component weight (W) up to a maximum of 6 dB as follows:

		Weight in kg	Weight in lb
dB reduction	=	10 log(W/45.4)	10 log(W/100)
$ASD_{(plateau)}$ level	=	0.04•(45.4/W)	0.04•(100/W)

The sloped portions of the spectrum shall be maintained at plus and minus 3 dB/oct. Therefore, the lower and upper break points, or frequencies at the ends of the plateau become:

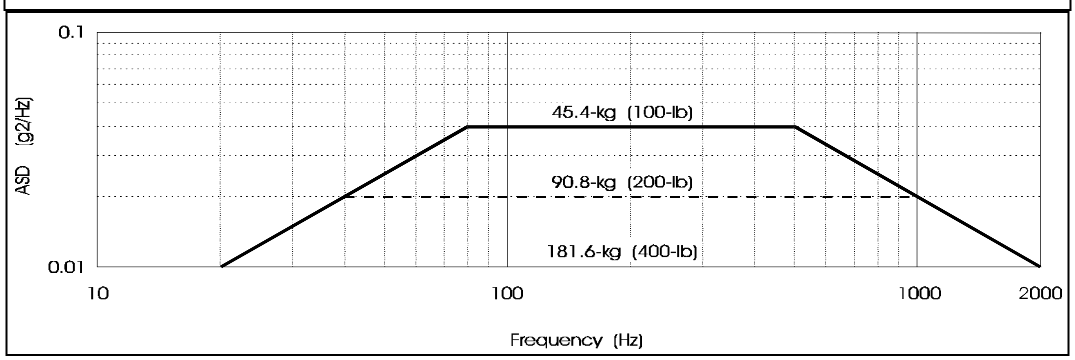
F_L = 80 (45.4/W) [kg]
= 80 (100/W) [lb]

F_L = frequency break point low end of plateau

F_H = 500 (W/45.4) [kg]
= 500 (W/100) [lb]

F_H = frequency break point high end of plateau

The test spectrum shall not go below 0.01 G^2/Hz. For components whose weight is greater than 182-kg or 400 pounds, the workmanship test spectrum is 0.01 G^2/Hz from 20 to 2000 Hz with an overall level of 4.4 G_{rms}.



2.4.2.7 Retest of Reflight Hardware - For reflight hardware, the amount of retest that is needed is determined by considering the amount of rework done after flight and by comparing the stresses of the upcoming flight with those of the previous flight. The principal objective is to verify the workmanship. If no disassembly and rework was done, the test may not be necessary. The effects of storage, elapsed time since last exposure, etc. shall be considered in determining the need for retest. Subsystems that have been taken apart and reassembled shall, as a minimum, be subjected to an acoustic test (levels shall be equal to the limit levels) and a random vibration test in at least one axis. More comprehensive exposures shall be considered if the rework has been extensive.

2.4.3 Sinusoidal Sweep Vibration Qualification

Sine sweep vibration tests are performed to qualify prototype/protoflight hardware for the low-frequency sine transient or sustained sine environments when they are present in flight, and to provide a workmanship test for all payload hardware which is exposed to such environments and normally does not respond significantly to the vibroacoustic environment at frequencies below 50 Hz, such as wiring harnesses and stowed appendages.

For STS payloads, sine vibration is required only to qualify the flight hardware for inputs from sources such as retro/apogee motor resonant burning or ignition/burnout transients, or control-jet firings if they occur in flight. Each payload shall be assessed for such applicable sine test requirements. Qualification for these environments requires swept sine vibration tests at the payload, instrument, and component levels of assembly. Test levels shall be developed on a mission-specific basis as addressed in 2.4.3.1 and 2.4.3.2.

For a payload level test, the payload shall be in a configuration representative of the time the stress occurs during flight, with appropriate flight type hardware used for attachment. For example, if the test is intended to simulate the vibration environment produced by the firing of retro/apogee motors, the vibration source shall be attached at the retro/apogee motor adapter, and the payload shall be in a configuration representative of the retro/apogee motor burning mode of operation.

The above requirement also applies to ELV payloads. In addition, all ELV payloads shall be subjected to swept sine vibration testing to simulate low-frequency sine transient vibration and sustained, pogo-like sine vibration (if expected) induced by the launch vehicle. Qualification for these environments requires swept sine vibration tests at the payload, instrument, and component levels of assembly.

It is understood that, for some payload projects, the sinusoidal sweep vibration qualification program may have to be modified. For example, for very large ELV payloads (with very large masses, extreme lengths, or large c.g. offsets) it may be impracticable because of test facility limitations to perform a swept sine vibration test at the payload level of assembly. In that case, testing at the highest level of assembly practicable is required.

For the sinusoidal vibration environment, limit levels shall be used which are consistent with the minimum probability level given in Table 2.4-2. The qualification level is then defined as the limit level times 1.25. The test input frequency range shall be limited to the band from 5 to 50 Hz. The fatigue life considerations of 2.4.2.1 apply where hardware reliability goals demand the demonstration of a specific hardware lifetime. The sine sweep environmental test program shall be included in the environmental verification plan and environmental verification specification which are reviewed by the Office of Flight Assurance.

2.4.3.1 ELV Payload Sine Sweep Vibration Tests - At the payload level of assembly, ELV prototype/protoflight hardware shall, when practicable, be subjected to a sine sweep vibration design qualification test to verify its ability to survive the low-frequency launch environment. The test also provides a workmanship vibration test for payload hardware which normally does not respond significantly to the vibroacoustic environment at frequencies below 50 Hz, but can experience significant responses from the ELV low-frequency sine transient vibration and any sustained, pogo-like sine vibration. Guidelines for developing mission-specific test levels are given in 2.4.3.1.b.

a. Vibration Test Requirements - Protoflight hardware shall be subjected to a sine sweep vibration test to verify flightworthiness and workmanship. The test shall represent the qualification level (flight limit level times 1.25).

The test is intended for all ELV payloads (spacecraft) except those with very large masses, extreme lengths and/or large c.g. offsets, where it is impracticable because of test facility limitations.

Note: The GSFC vibration test facility, including shaker and auxiliary support equipment, is currently designed to test 10,000 lb (4,540 kg) payloads and has been calibrated by sine sweep vibration of an 8,000 lb. (3630 kg) test item. A math model of the shaker system is available for pre-test dynamic analysis of the combined shaker, fixture, and payload as part of the operational hazards control.

If the sine sweep vibration test is not performed at the payload level of assembly, it shall be performed at the next lowest practicable level of assembly.

The payload in its launch configuration shall be attached to a vibration fixture by use of a flight-type launch-vehicle attach fitting (adapter) and attachment (separation system) hardware. Sine sweep vibration shall be applied at the base of the adapter in each of three orthogonal axes, one of which is parallel to the thrust axis. The test sweep rate shall be 4 octaves per minute to simulate the flight sine transient vibration; lower sweep rates shall be used in the appropriate frequency bands as required to match the duration and rate of change of frequency of any flight sustained, pogo-like vibration. The test shall be performed by sweeping the applied vibration once through the 5 to 50 Hz frequency range in each test axis. Mission-specific sine sweep test levels shall be developed for each ELV payload. Guidelines for developing the test levels are given in 2.4.3.1.b.

Prior to the payload test, a survey of the test fixture/exciter combination shall be performed to evaluate the fixture dynamics, the proposed choice of control accelerometer locations, and the control strategy. The evaluation shall include consideration of cross-axis responses. If a mechanical test model of the payload is available it should be included in the survey to evaluate the need for limiting (or notching).

During the protoflight hardware sine sweep vibration test to the specified test levels, loads induced in the payload and/or adapter structure while sweeping through resonance shall not exceed 1.25 times flight limit loads. If required, test levels shall be reduced ("notched") at critical frequencies. Acceleration responses of specific critical items may also be limited to 1.25 times flight limit levels if required to preclude unrealistic levels, provided that the spacecraft model used for the coupled loads analysis has sufficient detail and that the specific responses are recovered (using the acceleration transformation matrix) from the coupled loads analysis results. The minimum controlled input test level shall be ± 0.1 g to facilitate shaker control.

A low-level sine sweep shall be performed prior to the protoflight-level sine sweep test in each test axis. Data from the low-level sweeps measured at locations identified by a notching analysis shall be examined to determine if there are any significant test response deviations from analytical predictions. The data utilized shall include cross-axis response levels. Based on the results of the low-level tests, the predetermined notch levels shall be verified prior to the protoflight-level test. The flight limit loads used for notching analysis shall be based on the final verification cycle coupled loads analysis (including a test-verified payload model).

b. Mission-Specific Test Level Development - Sinusoidal vibration test levels required to simulate the flight environment for ELV spacecraft vary with the payload attach fitting (adapter) and spacecraft configuration, including overall weight and length, mass and stiffness distributions, and axial-to-lateral coupling. It therefore is impracticable to specify generalized sine sweep vibration test levels applicable to all spacecraft, and mission-specific test levels must be developed for each ELV spacecraft based on the coupled loads analysis. The ELV loading conditions of Table 2.4-3 shall be considered in developing the sine test levels, as addressed in 2.4.1.1.1

Prior to the availability of coupled loads analysis results, preliminary sine test levels may be estimated by using the ELV "user manual" sine vibration levels, provided in the appendices (truncated at 50 Hz) for spacecraft base drive analysis, with notching levels based on net loads equivalent to the user manual cg load factor loads. Alternatively, spacecraft interface dynamic response data from flight measurements or coupled loads analysis for similar spacecraft may be used for the base drive input in conjunction with a suitable uncertainty factor.

c. Performance - Before and after each vibration test, the payload shall be examined and functionally tested. During the tests, performance shall be monitored in accordance with the verification specification.

2.4.3.2 ELV Payload Subsystem (including Instruments) and Component Sine Sweep Vibration Tests - As a screen for design and workmanship defects, these items (per Table 2.4-1) shall be subjected to a sine sweep vibration test along each of three mutually perpendicular axes. For the sinusoidal vibration environment, limit levels shall be defined to be consistent with the minimum probability level of Table 2.4-2. The protoflight qualification level is then defined as the limit level times 1.25. The test input frequency range shall be limited to the band from 5 to 50 Hz. The fatigue life considerations of 2.4.2.1 apply where hardware reliability goals demand the demonstration of a specific hardware lifetime.

a. Vibration Test Requirements - The test item in its launch configuration shall be attached to the test equipment by a rigid fixture. The mounting shall simulate, insofar as practicable, the actual mounting of the item in the payload, with particular attention given to duplicating the mounting interface. All connections to the item (connectors and harnesses, plumbing, etc.) should be simulated with lengths at least to the first tie-down point. In mating the test item to the fixture, a flight-type mounting (including vibration isolators or kinematic mounts, if part of the design) and fasteners, including torque levels and locking features, shall be used. Normally-sealed items shall be pressurized during test to their prelaunch pressure.

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 shall be considered to cover those effects.

Sine sweep vibration shall be applied at the base of the test item in each of three mutually perpendicular axes. The test sweep rate shall be consistent with the payload-level sweep rate, i.e., 4 octaves per minute to simulate the flight sine transient vibration, and (if required) lower sweep rates in the appropriate frequency bands to match the duration and rate of change of frequency of any flight sustained, pogo-like vibration. The test shall be performed by sweeping the applied vibration once through the 5 to 50 Hz frequency range in each test axis.

Spacecraft subsystem, including instrument, and component levels depend on the type of structure to which the item is attached, the local attachment stiffness, the distance from the spacecraft separation plane, and the item's mass, size, and stiffness. It therefore is impracticable to specify generalized sine sweep vibration test levels applicable to all subsystems/instruments, and components, and mission-specific test levels shall be developed for each payload. Guidelines for developing the specific test levels are given in 2.4.3.2.b.

Prior to the test, a survey of the test fixture/exciter combination shall be performed to evaluate the fixture dynamics, the proposed choice of control accelerometer locations, and the control strategy. The evaluation shall include consideration of cross-axis responses. If a mechanical test or engineering model of the test article is available it should be included in the survey.

A low-level sine sweep shall be performed prior to the protoflight level sine sweep test in each test axis (with particular emphasis on cross-axis responses) to verify the control strategy and check test fixture dynamics.

b. Mission Specific Test Level Development - The mission-specific sine sweep test levels for spacecraft subsystems/components should be based on test data from structural model spacecraft sine sweep tests if available. If not available, the test levels should be based on an envelope of two sets of responses:

(1) Coupled loads analysis dynamic responses should be utilized if acceleration-response time histories are available at the test article location for all significant flight event loading conditions. Equivalent sine sweep vibration test input levels should be developed using shock response spectra (SRS) techniques for transient flight events. It should be noted that, in developing equivalent test input levels by dividing the SRS by Q (where $Q=C_c/2C$), assumption of a lower Q is more conservative. In the absence of test data, typical assumed values of Q for subsystems/components are from 10 to 20. For pogo-like flight events, the use of SRS techniques is not generally required.

(2) Subsystem/component responses from a base drive analysis of the spacecraft and adapter, using the spacecraft sine sweep test levels as input (in three axes), should be included in the test level envelope. The base drive responses of the test article should be corrected for effects of the spacecraft test sweep rates if the sweep rates are not included in the base drive analysis input. Subsystem/component test sweep rates should match spacecraft test sweep rates.

Since the shaker can only apply translational (but not rotational) accelerations, for test articles with predicted large rotational responses it may be necessary to increase the test levels based on analysis to assure adequate response levels.

Also, for certain cases such as large items mounted on kinematic mount flexures, which experience both significant rotations and translations, it may be necessary to use the test article c.g. rotational and translational acceleration response levels as not-to-exceed test levels in conjunction with appropriate notching or limiting.

c. Performance - Before and after test exposure, the test item shall be examined and functionally tested. During the test, performance shall be monitored in accordance with the verification specification.

2.4.3.3 Acceptance Requirements - Sine sweep vibration testing for the acceptance of previously qualified hardware shall be conducted at the flight limit levels using the same sweep rates as used for protoflight hardware.

2.4.4 Mechanical Shock Qualification

Both self-induced and externally induced shocks shall be considered in defining the mechanical shock environment.

2.4.4.1 Subsystem Mechanical Shock Tests - All subsystems, including instruments, shall be qualified for the mechanical shock environment.

a. Self-Induced Shock - The subsystem shall be exposed to self-induced shocks by actuation of all shock-producing devices. Self-induced shocks occur principally when pyrotechnic and pneumatic devices are actuated to release booms, solar arrays, protective covers, etc. Also the impact on deployable devices as they reach their operational position at the "end of travel" is a likely source of significant shock. When hardware contains such devices, it shall be exposed to each shock source twice to account for the scatter associated with the actuation of the same device. The internal spacecraft flight firing circuits should be used to trigger the event rather than external test firing circuits. At the project's discretion, this testing may be deferred to the payload level of assembly.

b. Externally Induced Shock - Mechanical shocks originating from other subsystems, payloads, or launch vehicle operations must be assessed. When the most severe shock is externally induced, a suitable simulation of that shock shall be applied at the subsystem interface. When it is feasible to apply this shock with a controllable shock-generating device, the qualification level shall be 1.4 times the maximum expected value at the subsystem interface, applied once in each of the three axes. A pulse or complex transient (whose positive and negative shock spectrum matches the desired spectrum within +25% and -10%) is applied to the test item interface once along each of the three axes. Equalization of the shock spectrum is performed at a maximum resolution of one-third octave. The critical damping ratio (c/c_c) used in the shock spectral analysis of the test pulse should equal the damping ratio used in the analysis of the data from which the test specification was derived. In the absence of a strong rationale for some other value, a damping ratio equivalent to a Q of 10 shall be used for shock spectrum analysis.

If the project so chooses or if it is not feasible to apply the shock with a controllable shock-generating device (e.g. the subsystem is too large for the device), the test may be conducted at the payload level by actuating the devices in the payload that produce the shocks external to the subsystem to be tested. The shock-producing device(s) must be actuated a minimum of two times for this test.

It will not be necessary to conduct a test for externally induced shocks if it can be demonstrated that the shock spectrum of the self-induced environment is greater at all
frequencies than the envelope of the spectra created by the external events at all locations within the subsystem.

c. <u>The STS Shock Environment</u> - Mechanical shock occurring in a payload as a result of STS operations or the activities of other payloads within the cargo bay are estimated to be negligible. Therefore, when the self-induced shock test is conducted at the payload level of assembly, the externally induced mechanical shock environment may be disregarded. When the self-induced shock test is conducted at the subsystem level of assembly, the shock simulation will be that induced by the other subsystems of the same payload. An envelope of such shocks as defined at the subsystem interface with the payload constitutes the externally induced mechanical shock environment.

d. <u>Test Setup</u> - During test, the test item should be in the electrical and mechanical operational modes appropriate to the phase of mission operations when the shock will occur.

e. <u>Performance</u> - Before and after the mechanical shock test, the test item shall be examined and functionally tested. During the tests, performance shall be monitored in accordance with the verification specification.

2.4.4.2 <u>Payload (Spacecraft) Mechanical Shock Tests</u> - The payload must be qualified for the shock induced during payload separation (when applicable) and for any other externally induced shocks whose levels are not enveloped at the payload interface by the separation shock level. The payload separation shock is usually higher than other launch vehicle-induced shocks; however that is not always the case. For instance, the shocks induced at the payload interface during inertial upper stage (IUS) actuation can be greater. In addition, mechanical shock testing may be performed at the payload level of assembly to satisfy the subsystem mechanical shock requirements of 2.4.4.1.

a. <u>Other Payload (Spacecraft) Shocks</u> - If launch vehicle induced shocks or shocks from other sources are not enveloped by the separation test, the spacecraft must be subjected to a test designed to simulate the greater environment. If a controllable source is used, the qualification level shall be 1.4 x the maximum expected level at the payload interface applied once in each of the three axes. The tolerance band on the simulated level of response is +25% and -10%. The analysis should be performed with a critical damping corresponding to a Q of 10 or, if other than 10, with the Q for which the shock being simulated was analyzed. The subsystem mechanical shock requirements may be satisfied by testing at the payload level of assembly as described above.

b. Performance - Before and after the mechanical shock test, the test item shall be examined and functionally tested. During the tests, performance shall be monitored in accordance with the verification test plan and specification.

2.4.4.3 Acceptance Requirements - The need to perform mechanical shock tests for the acceptance of previously qualified hardware shall be considered on a case-by-case basis. Testing should be given careful consideration evaluating mission reliability goals, shock severity, hardware susceptibility, design changes from the previous qualification configuration including proximity to the shock source, and previous history.

2.4.5 Mechanical Function Verification

A kinematic analysis of all payload mechanical operations is required (a) to ensure that each mechanism can perform satisfactorily and has adequate margins under worst-case conditions, (b) to ensure that satisfactory clearances exist for both the stowed and operational configurations as well as during any mechanical operation, and (c) to ensure that all mechanical elements are capable of withstanding the worst-case loads that may be encountered. Payload qualification tests are required to demonstrate that the installation of each mechanical device is correct and that no problems exist that will prevent proper operation of the mechanism during mission life.

Subsystem qualification tests are required for each mechanical operation at nominal-, low-, and high-energy levels. To establish that functioning is proper for normal operations, the nominal test shall be conducted under the most probable conditions expected during normal flight. A high-energy test and a low-energy test shall also be conducted to prove positive margins of strength and function. The levels of these tests shall demonstrate margins beyond the nominal conditions by considering adverse interaction of potential extremes of parameters such as temperature, friction, spring forces, stiffness of electrical cabling or thermal insulation, and, when applicable, spin rate. Parameters to be varied during the high- and low-energy tests shall include, to the maximum extent practicable, all those that could substantively affect the operation of the mechanism as determined by the results of analytic predictions or development tests. As a minimum, successful operation at temperature extremes 10°C beyond the range of expected flight temperatures shall be demonstrated.

2.4.5.1 Life Testing

A life test program shall be implemented for mechanical elements that move repetitively as part of their normal function and whose useful life must be determined in order to verify their adequacy for the mission. The verification plan and the verification specification shall address the life test program, identifying the mechanical elements that require such testing, describing the test hardware that will be used, and the test methods that will be employed.

Life test planning should be initiated as early as possible in the development phase to allow enough time to complete the life test and thoroughly disassemble and inspect the mechanism, while retaining enough time to react to any anomalous findings.

The life test mechanism shall be fabricated and assembled such that it is as nearly identical as possible to the actual flight mechanism, with special attention to the development and implementation of detailed assembly procedures and certification logs. In fact, it is preferable that the life test mechanism actually be a flight spare. Careful attention should be given to properly simulating the flight interfaces, especially the perhaps less obvious details, such as the method of mounting of the mechanism, the preloading and/or clamping of bearings or other tribological interfaces, the routing of harnesses, the attachment of thermal blankets, and any other items that could have an influence on the performance of the mechanism.

Prior to the start of life testing, mechanisms should be subjected to the same ground testing environments that are anticipated for the flight units (protoflight or acceptance, as appropriate). These environments may have a significant influence on the life test performance of the mechanism.

Consideration should be given to the geometry of the test set-up and the effects of gravity on the performance of the life test mechanism, including the effects on lubrication and external loads. For example, gravity may cause lubrication to puddle at the bottom of a bearing race or run out of the bearing. In some cases, the effects of gravity may cause abnormally high loads on the mechanism.

The thermal environment of the mechanism during the life test should be representative of the on-orbit environment. If expected bulk temperature changes are significant, then the life test should include a number of transitions from the hot on-orbit predictions to the cold on-orbit predictions, and vice versa. Depending on the thermal design, significant temperature gradients may be developed which could have a profound influence on the life of the mechanism and, therefore, should be factored into the thermal profile for the life test.

Consideration should be given to including in the life test the effects of vacuum on the performance of the mechanism with particular attention to its effects on the thermal environment (i.e., no convective heat transfer) and potentially adverse effects on lubrication and materials.

Life testing of electrically powered devices should be conducted with nominal supply voltage.

The selection of the proper instrumentation for the life test is very important. Physical parameters that are an indication of the health of the mechanism should be closely monitored and trended during the life test. These parameters may include in-rush and steady-state currents, electrical opens or shorts, threshold voltages, temperatures (both steady-state and rate of change), torques, angular or linear positions, vibration, and times of actuation.

The life test should be designed to "fail safe" in the event of any failure of the test setup, ground support equipment, or test article. There may be a severe impact to the life test results if it is necessary to stop a life test to replace or repair ground support equipment. Uninteruptable power supplies should be considered when required for autonomous shutdown without damage to the test article or loss of test data. Redundant sensors should be provided for all critical test data. If used, the vacuum pumping station should be designed to maintain the integrity of the vacuum in the event of a sudden loss of power.

The test spectrum for the life test shall represent the required mission life for the flight mechanism, including both ground and on-orbit mechanism operations. In order to reduce test time and cost, the test spectrum should be simplified as much as possible while retaining an appropriate balance between realism and conservatism. It should include, if applicable, a representative range of velocities, number of direction reversals, and number of dead times or stop/start sequences between movements. Direction reversals or stop/start operations could have a significant effect on lubrication life, internal stresses, and, ultimately, the long term performance of the mechanism.

The minimum requirement for demonstrated life test operation without failure shall be 1.0 times the mission life. However, due to the uncertainties and simplifications inherent in the test, a marginally successful test requires post-test inspections and characterizations to extrapolate the remaining useful life. Because this can be difficult and uncertain, the recommended goal for the life test is to achieve a 25% margin on mission life. Even higher margins should be considered if time permits in order to establish greater confidence. Pre and post-life test baseline performance tests shall be conducted with clear requirements established for determining minimum acceptable performance at end-of-life.

When it is necessary to accelerate the life test in order to achieve the required life demonstration in the time available, caution must be exercised in increasing the speed or duty cycle of the mechanism. Mechanisms may survive a life test at a certain speed or duty cycle, but fail if the speed is increased or decreased, or if the duty cycle is increased significantly. There are three lubrication regimes to consider when considering whether to accelerate a life test, "boundary lubrication", "mixed lubrication", and "full elastohydrodynamic (EHD) lubrication".

For boundary and mixed lubrication regimes, the most likely failure mechanism will be wear, not fatigue. Unfortunately failure by wear is not an exact science and no formulas yet exist to accurately predict life available in these lubrication regimes. Therefore, life test acceleration by increasing speed should not be considered.

In the EHD regime, no appreciable wear should occur and the failure mechanism should be material fatigue rather than wear. Therefore, while life test acceleration by increasing speed may be considered, other speed limiting factors must also be considered. For example, at the speed at which EHD lubrication is attained, one must be concerned with bearing retainer imbalances which may produce excessive wear of the retainer, which would in turn produce contaminants which could degrade the performance of the bearings.

If there are significant downtimes associated with the operation of an intermittent mechanism, the life test can be accelerated by reducing this downtime, as long as this does not adversely affect temperatures and leaves enough "settle time" for the lubricant film to "squish out" of the contact area to simulate a full stop condition. If the mechanism runs continuously, it may still be possible to accelerate the speed somewhat by increasing the temperature (higher temperature will reduce the film thickness) to mimic the film thickness at the lower speed and lower temperature. Caution must be used, however, that the higher temperature will not cause any chemical differences in the lubricant which could effect the outcome of the test.

For all these reasons, the life test should be run as nearly as possible using the on-orbit speeds and duty cycles. In some cases it may not be possible to accelerate the test at all.

Upon completion of the life test, it is imperative that careful disassembly procedures are followed and that the proper level of inspections are conducted. Successful tests will not have any anomalous conditions such as abnormal wear, significant lubrication breakdown, or excessive debris generation. These or other anomalous conditions may be cause for declaring the life test a failure despite completion of the required test spectrum. A thorough investigation of all moving components and wear surfaces should be conducted. This may include physical dimensional inspection of components, high magnification photography, lubricant analysis, Scanning Electron Microscope (SEM) analysis, etc.

For those items determined not to require life testing, the rationale for eliminating the test shall be provided along with a description of the analyses that will be done to verify the validity of the rationale. Caution should be exercised when citing heritage as a reason for not conducting a life test. Many factors such as assembly personnel, environments, changes to previously used processes, or "improvements" to the design may lead to subtle differences in the mechanism that in turn could affect the outcome of a life test. For example, environmental testing of the heritage mechanism may not actually have enveloped the predicted flight environment of the mechanism under consideration.

2.4.5.2 Demonstration - Compliance with the mechanical function qualification requirements is demonstrated by a combination of analysis and test. The functional qualification aspects of the demonstration are discussed below. The life test demonstrations are peculiar to the design and cannot be described here. Rather, they must be described in detail in an approved verification plan and verification specification.

a. Analysis - An analysis of the payload shall be conducted to ensure that satisfactory clearances exist for both the stowed and operational configurations. Therefore, in conjunction with the flight-loads analysis, an assessment of the relative displacements of the various payload elements with respect to other payloads and various elements of the STS, or ELV payload fairing, shall be made for potentially critical events. During analysis, the following effects shall be considered: an adverse build-up of tolerances, thermal distortions, and mechanical misalignments, as well as the effects of static and dynamic displacements induced by particular mission events.

In addition, a kinematic analysis of all deployment and retraction sequences shall be conducted to ensure that each mechanism has adequate torque margin under worst-case friction conditions and is capable of withstanding the worst-case loads that may be encountered during unlatching, deployment, retraction, relatching, or ejection

sequences. In addition, the analysis shall verify that sufficient clearance exists during the motion of the mechanisms to avoid any interference.

b. Payload Testing - A series of mechanical function tests shall be performed on the payload to demonstrate "freedom-of-motion" of all appendages and other mechanical devices whose operation may be affected by the process of integrating them with the payload. The tests shall demonstrate proper release, motion, and lock-in of each device, as appropriate, in order to ensure that no tolerance buildup, assembly error, or other problem will prevent proper operation of the mechanism during mission life. Unless the design of the device dictates otherwise, mechanical testing may be conducted in ambient laboratory conditions. The testing shall be performed at an appropriate time in the payload environmental test sequence and, if any device is subsequently removed from the payload, the testing shall be repeated after final reinstallation of the device.

c. Subsystem Testing - Each subsystem, and instrument, that performs a mechanical operation shall undergo functional qualification testing. At the project's discretion, however, such testing may be performed at the payload level of assembly. The test is conducted after any other testing that may affect mechanical operation. The purpose is to confirm proper performance and to ensure that no degradation has occurred during the previous tests.

During the test, the electrical and mechanical components of the subsystem shall be in the appropriate operational mode. The subsystem is also exposed to pertinent environmental effects that may occur before and during mechanical operation. The verification specification shall stipulate the tests to be conducted, the necessary environmental conditioning, and the range of required operations.

It is desirable that preliminary mechanical function tests and exploratory design development tests shall have been performed with a structural model prior to qualification testing of the subsystem. Such tests uncover weaknesses, detect failure modes, and allow time before protoflight testing to develop and institute quality control procedures and corrective redesign.

(1) Information Requirements - The following information is necessary to define the series of functional qualification tests:

- o A description of mission requirements, how the mechanism is intended to operate, and when operation occurs during the mission;

- o The required range of acceptable operation and criteria for acceptable performance;

- o The anticipated variation of all pertinent flight conditions or other parameters that may affect performance.

(2) Test Levels and Margins - For each mechanical operation, such as appendage deployment, tests at nominal-, low-, and high-energy levels shall be performed. One test shall be conducted at the most probable level that will occur during a normal mission (the nominal level). The test will establish that functioning is proper for nominal operating conditions and baseline measurements will be obtained for subsequent tests.

Other tests shall be conducted to prove positive margins of strength and function, including a high-energy test and a low-energy test. The levels of these tests shall demonstrate margins beyond the nominal operational limits. The margins shall not be selected arbitrarily, but shall take into account all the uncertainties of operation, strength, and test.

While in an appropriate functional configuration the hardware shall be subjected to events such as separation, appendage deployment, retromotor ejection, or other mechanical operations, such as spin-up or despin that are associated with the particular mission.

Gravity compensation shall be provided to the extent necessary to achieve the test objectives. As a guide, the uncompensated gravity effects should be less than 10 percent of the operational loads. Uncompensated gravity of 0.l g is usually achievable and acceptable for separation tests and for comparative measurements of appendage positioning if the direction is correct, i.e., the net shear and moment imposed during measurements acts in the same direction as it would in flight, thereby causing any mechanism with backlash to assume the correct extreme positions. For testing of certain mechanical functions, however, more stringent uncompensated gravity constraints may be required. When appropriate, the subsystem shall be preconditioned before test or conditioned during test to pertinent environmental levels. This can include vibration, high- and low-temperature cycling, pressure-time profiles, transportation and handling.

(3) Performance - Before and after test, the subsystem shall be examined and electrically tested. During the test, the subsystem performance shall be monitored in accordance with the verification specification.

2.4.5.3 Torque Ratio - The torque ratio shall be determined by test to demonstrate the minimum requirements.

The torque ratio (TR) is a measure of the degree to which the torque available to accomplish a mechanical function exceeds the torque required. The torque ratio is simply the ratio of the driving or available torque to the required or resistive torque. Numerically, the torque margin is the torque ratio minus one. The torque ratio requirement defined below applies to all mechanical functions, those driven by motors as well as springs, etc. at beginning of life (BOL) only; end of life (EOL) mechanism performance is determined by life testing as discussed in paragraph 2.4.5.1, and/or by analysis. Positive margin must be shown for worst case conditions EOL. For linear devices, the term "force" shall replace "torque" throughout the section.

For final design verification, the torque ratio shall be verified by testing the qualification unit both before and after exposure to qualification level environmental testing. The torque ratio shall also be verified by testing all flight units both before and after exposure to acceptance level environmental testing. All torque ratio testing shall be performed at the highest possible level of assembly, throughout the mechanism's range of travel, under worst-case BOL environmental conditions, representing the worst-case combination of maximum and/or minimum predicted (not qualification) temperatures, gradients, positions, acceleration/ deceleration of load, voltage, vacuum, etc.

Along with system level test, available torque (T_{avail}) and resistive torque (T_{res}) under worst case conditions should be determined, whenever possible, through system and subsystem level tests. Torque ratios for gear driven systems should be verified, using subsystem level results, on both sides of the geartrain. The minimum available torque for these types of systems shall never be less than 1 in-oz at the motor. Kick-off springs which do not operate over the entire range of the mechanical function shall be neglected when computing available torque.

For systems that include (velocity dependent) dampers, and are deployable rate independent, it is allowable to characterize (as nearly as possible) only the frictional resistive torque. For systems that include dampers, and are deployable rate dependent, appropriate measures shall be taken to properly account for (as nearly as possible) the resistive torque produced by the dampers.

The torque ratio is then given by:

$$TR = T_{avail}/T_{res}$$

The minimum required test-verified torque ratios for various types of mechanism systems prior to environmental testing are shown below. The system type should be determined and agreed to by the project early in the design phase.

System Type	TR_{min}
Systems which are dominated by resistive torques due to inertia, such as momentum and reaction wheels	1.5
Systems which are dominated by resistive torques due to a combination of both inertia and friction, such as large pointing platforms and heavy deployable systems	2.25
Systems which are dominated by resistive torques due to friction, such as deployment mechanisms, solar array drives, cable wraps, and despun platforms	3.0

After exposure to environmental testing, the reduction (if any) in test-verified torque ratio shall be no greater than 10%, after appropriate consideration has been given to the error inherent in the test methods used to measure the torque ratio.

It is important to note that this torque ratio requirement relates to the verification phase of the hardware in question. Conservative decisions must be made during the design phase to ensure adequate margins will be realized.

The required torque ratios should be appropriately higher than given above if:

a. The designs involve an unusually large degree of uncertainty in the characterization of resistive torques.

b. The torque ratio testing is not performed in the required environmental conditions or is not repeatable.

c. The torque ratio testing is performed at the component level.

2.4.5.4 Acceptance Requirements - For the acceptance testing of previously qualified hardware, the payload and subsystem tests described in 2.4.5.2.b and 2.4.5.2.c shall be performed, except that the subsystem tests need be performed only at the nominal energy level. Adequate torque ratio (margin) shall be demonstrated for all flight mechanisms.

2.4.6 Pressure Profile Qualification

The need for a pressure profile test shall be assessed for all subsystems. A qualification test shall be required if analysis does not indicate a positive margin at loads equal to twice those induced by the maximum expected pressure differential during launch. If a test is required, the limit pressure profile is determined by the predicted pressure-time profile for the nominal trajectory of the particular mission.

Because pressure-induced loads vary with the square of the rate of change, the qualification pressure profile is determined by multiplying the predicted pressure rate of change by a factor of 1.12 (the square root of 1.25, the required qualification factor on load).

2.4.6.1 Demonstration - The hardware is qualified for the pressure profile environment by analysis and/or test. An analysis shall be performed to estimate the pressure differential induced by the nominal launch and reentry trajectories, as appropriate, across elements susceptible to such loading (e.g. thermal blankets, contamination enclosures, and housings of components). If analysis does not indicate a positive margin at loads equal to twice those induced by the maximum expected pressure differential, testing is required. Although testing at the subsystem level is usually appropriate, the project may elect to test at the payload level of assembly.

a. Test Profile - The flight pressure profile shall be determined by the analytically predicted pressure-time history inside the cargo bay (or payload fairing) for the nominal launch trajectory for the mission (including reentry if appropriate). Because pressure-induced loads vary as the square of the pressure rate, the pressure profile for qualification is determined by increasing the predicted flight rate by a factor of 1.12 (square root of 1.25, the required test factor for loads). The pressure profile shall be applied once.

b. Facility Considerations - Loads induced by the changing pressure environment are affected both by the pressure change rate and the venting area. Because the exact times of occurrence of the maximum pressure differential is not always coincident with the maximum rate of change, the pumping capacity of the facility must be capable of matching the desired pressure profile within ± 5% at all times.

c. Test Setup - During the test, the subsystem shall be in the electrical and mechanical operational modes that are appropriate for the event being simulated.

d. Performance - Before and after the pressure profile test, the subsystem shall be examined and functionally tested. During the tests, performance shall be monitored in accordance with the verification specification.

2.4.6.2 Acceptance Requirements - Pressure profile test requirements do not apply for the acceptance testing of previously qualified hardware.

2.4.7 Mass Properties Verification

Hardware mass property requirements are mission-dependent and, therefore, are determined on a case-by-case basis. The mass properties program shall include an analytic assessment of the payload's ability to comply with the mission requirements, supplemented as necessary by measurement.

2.4.7.1 Demonstration - The mass properties of the payload are 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 in order to attain the accuracy required for the mission and to ensure that analytical determination of payload mass properties is feasible. Determination of the various subsystem properties 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 must be determined:

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

b. Balance - Hardware is balanced in accordance with mission requirements. Balance may be achieved analytically, if necessary, with the aid of direct measurements.

   (1) Procedure for Direct Measurement - The usual procedure for direct measurement is to perform an initial balance before beginning the environmental verification program and a final balance after completing 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 made during the verification program such as hardware replacement or redesign.

   (2) Maintaining Balance - It is recommended that changes to the hardware that may affect weight distribution be minimized after completion of final balance. The effects of such changes (including any disassembly, hardware substitution, etc.) on the residual unbalance of the hardware should be assessed. That 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.

(3) 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 of the verification specification. Balance operations include interface, fit, and alignment checks as 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.

2.4.7.2 Acceptance Requirements - The mass property requirements cited above apply to all flight hardware.

SECTION 2.5

EMC

2.5 ELECTROMAGNETIC COMPATIBILITY (EMC) REQUIREMENTS

The general requirements for electromagnetic compatibility are as follows:

a. The payload (spacecraft) and its elements shall not generate electromagnetic interference that could adversely affect its own subsystems and components, other payloads, or the safety and operation of the launch vehicle (STS or ELV) and launch site.

b. The payload (spacecraft) and its subsystems and components shall not be susceptible to emissions that could adversely affect their safety and performance. This applies whether the emissions are self-generated or emanate from other sources, or whether they are intentional or unintentional.

2.5.1 Requirements Summary

The EMC test requirements herein when performed as a set are intended to provide an adequate measure of hardware quality and workmanship. The tests are performed to fixed levels which are intended to envelope those that may be expected during a typical mission and allow for some degradation of the hardware during the mission. The levels should be tailored to meet mission specific requirements, such as, the enveloping of launch vehicle and launch site environments, or the inclusion of very sensitive detectors or instruments in the payload.

Thus tailored, the requirements envelope the environments usually encountered during integration and ground testing. However, because some payloads may have sensors and devices that are particularly sensitive to the low-level EMI ground environment, special work-around procedures may have to be developed to meet individual payload needs.

2.5.1.1 The Range of Requirements - Table 2.5-1 is a matrix of EMC tests that apply to a wide range of hardware intended for launch either by the STS or an expendable launch vehicle (ELV). Tests are prescribed at the component, subsystem, and payload levels of assembly. Not all tests apply to all levels of assembly or to all types of payloads. The project must select the requirements that fit the characteristics of the mission and hardware, e.g. a transmitter would require a different group of EMC tests than a receiver. Symbols in the hardware levels of assembly columns will assist in the selection of an appropriate EMC test program.

Once the program is selected, all flight hardware shall be tested. The EMC test program is meant to uncover workmanship defects and unit-to-unit variations in electromagnetic characteristics, as well as design flaws. The qualification and flight acceptance EMC programs are the same. Performance of both will provide a margin of hardware reliability.

A specific group of EMC requirements are imposed by Johnson Space Center (JSC) on STS payloads that operate on orbiter power or that operate on their own power within or near the orbiter. Those requirements, which are defined in the ICD 2-19001 document (1.7.), are partially included here for the convenience of the user; however the user is responsible for obtaining those requirements from ICD 2-19001, which is the controlling document.

Table 2.5-1
EMC Requirements per Level of Assembly

Type	Test Spacecraft	Paragraph Number	STS	ELV	Component	Subsystem/ Instrument	Payload*
CE	Dc power leads	2.5.2.1.a&c	X	X	Sb,Rb,R	Sb,Rb,R	Sb
CE	Ac power leads	2.5.2.1.a&c	X		Sb,Rb	Sb,Rb	Sb
CE	Power Leads	2.5.2.1.b	X	X	Rb,R	Rb,R	-
CE	Transients on orbiter dc power lines	2.5.2.1.d	X		Sb	Sb	Sb
CE	Spikes on orbiter ac power lines	2.5.2.1.e	X		Sb	Sb	Sb
CE	Antenna terminals	2.5.2.1.f	X	X	R	-	-
RE	Magnetic field (STS payloads)	2.5.2.2.a	X		-	-	Sd
RE	Ac magnetic field	2.5.2.2.b	X	X	Rb,R	Rb,R	Rb,R
RE	E-fields	2.5.2.2.c&d	X	X	Rb,R	Rb,R	Sd,Rb,R
RE	Payload transmitters	2.5.2.2.e	X	X	-	-	Sd,**
RE	Spurious (transmitter antenna)	2.5.2.2.f	X	X	-	Rb,R	-
CS	Power line	2.5.3.1.a	X	X	Rb,R	Rb,R	Rb
CS	Intermodulation products	2.5.3.1.b	X	X	Rb,R	-	-
CS	Signal rejection	2.5.3.1.c	X	X	Rb,R	-	-
CS	Cross modulation	2.5.3.1.d	X	X	Rb,R	-	-
CS	Power line transients	2.5.3.1.e	X	X	Rb,R	Rb,R	Rb
RS	E-field (general compatibility)	2.5.3.2.a	X	X	Rb,R	Rb,R	Rb,R
RS	Compatibility with orbiter transmitters	2.5.3.2.b	X		-	-	Rb
RS	Orbiter unintentional E-field	2.5.3.2.c	X		-	-	Rb
RS	Magnetic-field susceptibility	2.5.3.2.d	X	X	Rb,R	Rb,R	Rb,R
	Magnetic properties	2.5.4	X	X	R	R	R

CE - Conducted Emission
CS - Conducted Susceptibility
R - Test to ensure reliable operation of payload, and to help ensure compatibility with the launch vehicle and launch site
Rb - Test to ensure reliable operation of orbiter attached payloads
RE - Radiated Emission
RS - Radiated Susceptibility
Sb - Items interfacing with orbiter power in payload bay or in the cabin; required by ICD 2-19001
Sd - Items operating on or near orbiter; required by ICD 2-19001
* - Payload, Mission, or highest level of assembly
** - Must meet any unique requirements of launch vehicle and launch site for transmitters that are on during launch

A wide range of EMC test requirements are provided to cover a variety of free flyer and shuttle-attached payload operating modes. For example, some free flyers will be operated with the orbiter during prerelease and checkout procedures and must be tested to ensure EMC with the orbiter. The more stringent EMC environment occurs after the free flyer moves away from the orbiter when it becomes more susceptible to the operations of its own subsystems and sensitive instruments. Because some free flyers will not be operated or checked out before release from the orbiter, they will not have to meet the JSC EMC requirements and the tests need only ensure self-compatibility and survival after exposure to the high-level emissions from the orbiter's transmitters. Requirements are also provided for attached payloads that may be subjected throughout the mission to EMI from the orbiter and from other attached payloads.

The EMC tests are intended to verify that:

(1) The hardware will operate properly if subjected to conducted or radiated emissions from other sources that could occur during launch or in orbit (susceptibility tests).

(2) The hardware does not generate either conducted or radiated signals that could hinder the operation of other systems (emissions tests).

2.5.1.2 Testing at Lower Levels of Assembly - It is recommended that testing be performed at the component, subsystem, and payload levels of assembly. Testing at lower levels of assembly has many advantages: it uncovers problems early in the program when they are less costly to correct and less disruptive to the program schedule; it uncovers problems that cannot be detected or traced at higher levels of assembly; it characterizes box-to-box EMI performance, providing a baseline that can be used to alert the project to potential problems at higher levels of assembly; and it aids in troubleshooting.

2.5.1.3 Basis of the Tests - A description of the individual EMC tests listed in Table 2.5-1, including their requirement limits and test procedures, are provided in paragraphs 2.5.2 through 2.5.4.7 Most of the tests are based on the requirements of MIL-STD-461C and 462, as amended by Notice 1, and MIL-STD-463A (1.7.8). Note: all references in this document to MIL-STD-462 assume reference to Notice 1.

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 GSFC project approval is obtained.

The MIL-STD limits have been modified as appropriate to meet the EMC requirements for STS payloads as defined by ICD 2-19001 and also to meet the STS reliability requirements specified herein.

For ELV launch, additional EMC requirements may be placed on the spacecraft by the launch vehicle or launch site or in consideration of the mission launch radiation environment. Those requirements shall be established during coordination between the spacecraft project and the launch vehicle program office.

More stringent requirements may be needed for payloads with very sensitive electric field or magnetic field measurement systems. The tests and their limits shall be documented in the verification plan, specification, and procedures.

2.5.l.4 Safety and Controls - During prelaunch and prerelease checkout, sensitive detectors and hardware may require special procedures to protect them from the damage of high-level radiated emissions. If such procedures are needed, they should also be applied during EMC testing. Operational control procedures should also be instituted for EMC testing during prerelease checkout to minimize interference with the orbiter and other payloads as appropriate.

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

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

2.5.2 Emission Requirements

The following paragraphs on emission tests shall be used to implement the emission requirements of Table 2.5-1.

2.5.2.l Conducted Emission Limits - Conducted emission limits and requirements on power leads, as well as on antenna terminals, shall be applied to payload hardware as defined below. The requirements do not apply to secondary power leads to subunits within the level of assembly under test unless they are specifically included in a hardware specification.

a. Narrowband conducted emissions on power, and power-return leads (both dc and ac for STS) shall be limited to the levels specified in Figure 2.5-1.

   Testing shall be in accordance with MIL-STD-461C and 462, test numbers CE01 and CE03, as applicable, with limits as shown in Figure 2.5-1.

b. A Conducted Emissions (CE) test to control Common Mode Noise (CMN) shall be required at the subsystem/component level. This frequency domain current test shall be performed on all non-passive components which receive or generate spacecraft primary power.

   The purpose of the test is to limit CMN emissions that flow through the spacecraft structure and flight harness which result in the generation of undesirable electrical currents, and electro-magnetic fields at the integrated system level.

   Specific CMN requirements must be determined carefully from spacecraft hardware designs or mission scenario. Spacecraft which have analog or low level signal interfaces, low level detectors, and instruments that measure electromagnetic fields may be particularly sensitive to CMN. If mission requirements do not place stricter control on CMN, the limits of Figure 2.5-1a are suggested.

   The CMN test procedure is the same as narrowband CE01/03 except that the current probe is placed around both the plus and return primary wires together.

c. Broadband conducted emissions on power, and power-return leads (both dc and ac for STS) shall be limited to the levels specified in Figure 2.5-2. Testing shall be in accordance with MIL-STD-461C and 462, test number CE03, with limits as shown in Figure 2.5-2.

d. Transients produced by orbiter payloads on dc powerlines interconnecting to the orbiter, caused by switching or other operations, shall not exceed the limits defined in Figure 2.5-3 when fed from a source impedance close to but not less than the values defined in Figure 2.5-4 (The use of a battery cart is preferable to regulated dc power supplies). Each non-overlapping transient is considered independent of prior or post transients. Rise and fall times shall be greater than 1.0 microsecond. The steady state ripple voltage in the time domain (starting approximately one second after the transient) shall not exceed 28.45 volts nor go below 27.55 volts (28 ± 0.45 volts). A network for simulating the orbiter power source impedance is shown in Figure 2.5-5.

e. Transient spikes produced by orbiter payloads on ac powerlines from the orbiter to the payloads shall not exceed the limits defined in Figure 2.5-6 when they are fed from a source impedance not greater than 10 ohms. Peak spikes below 10 microseconds duration shall be limited to 60 volts superimposed on the 400 Hz sine wave. Rise and fall times shall be greater than 1.0 microsecond.

f. Conducted emissions on the antenna terminals of payload receivers, and transmitters in key-up modes shall not exceed 34 dB μV for narrowband emissions and 40 dB μV/MHz for broadband emissions.

Harmonics (greater than the third) and all other spurious emissions from transmitters in the key-down mode shall have peak powers 80 dB down from the power at the fundamental. Power at the second and third harmonics shall be suppressed by {50 + 10 Log(Peak Power in watts at the fundamental) dB}, or 80 dB whichever requires less suppression.

Testing shall be in accordance with MIL-STD-462, test number CE06. The test is conducted on receivers and transmitters before they are integrated with their antenna systems. Refer to MIL-STD-461C and MIL-STD-462 for additional details concerning this requirement.

2.5.2.2 Radiated Emission Limits - Radiated emission limits and requirements shall be applied to payload hardware as defined in sections 2.5.2.2.a through 2.5.2.2.f below. Additional tests or test conditions should be considered by the project if it appears that this may be necessary, for example, if the spacecraft receives at frequencies other than S-band (1.77 - 2.3 GHz).

a. Radiated ac magnetic field levels produced by orbiter payloads at distances of 1 meter from the payload shall not exceed 130 dB above 1 pico-tesla over the frequency range of 20 Hz to 2 kHz, then falling 40 dB per decade to 50 kHz as shown in Figure 2.5-8. Testing shall be in accordance with MIL-STD-462 test number RE04.

The dc magnetic field generated by orbiter payloads shall not exceed 170 dB pT at the payload envelope. This limit applies to electromagnetic and permanent magnetic devices.

b. Radiated ac magnetic field levels produced by STS free flyer (or ELV-launched) payloads and their subsystems shall be limited to 60 dB pT from 20 Hz to 50 kHz. This requirement may be deleted with project approval if subsystems or instruments are not inherently susceptible to ac magnetic fields; however, the requirements in paragraph a, above, still apply for STS payloads.

If the free flyer payloads or their instruments contain sensitive magnetic field detectors or devices with high sensitivities to magnetic fields, more stringent limits on magnetic field emission may be required. Testing shall be in accordance with MIL-STD-462, test number RE04, with limits as defined above.

c. Unintentional radiated narrowband electric field levels produced by payloads shall not exceed the levels specified in Figure 2.5-9. Testing shall be in accordance with MIL-STD 461C and 462, test number RE02, with the test frequency range and limits revised as defined in Figure 2.5-9. In addition, STS payloads shall not exceed the limits of Figure 2.5-9a.

d. Unintentional radiated broadband electric field levels produced by payloads shall not exceed the levels specified in Figure 2.5-10. Testing shall be in accordance with MIL-STD-461C and 462, test number RE02, with the test frequency range and the limits revised as defined in Figure 2.5-10.

e. Allowable levels of radiation from payload transmitter antenna systems depend on the launch vehicle and launch site.

For an ELV launch, any unique requirements of the launch vehicle and launch site for transmitters that will be on during launch must be met.

For STS applications, the allowable levels of radiation from orbiter payload transmitter antenna systems are shown in Figure 2.5-11. The radiation limits apply at surfaces defined as follows:

(1) The allowable payload-to-payload (cargo element-to-cargo element) limit is defined as the radiation impinging upon imaginary planes (orbiter y, z) located at the smallest and largest X_o allocated to the radiating payload, or upon the imaginary planes (orbiter x, z) located at the smallest and largest $\pm Y_o$ allocated to the radiating payload. The limits have been established to permit flexibility in manifesting payloads. However, the limits can be waived by JSC for individual payloads (cargo elements) with selective mixing of payloads in flight manifesting.

(2) The allowable payload-to-orbiter limit is defined as the radiation impinging upon an imaginary surface 7.6 cm (3 inches) beyond the payload allowable envelope for envelope Z_o of 410 or less. This does not limit radiation at higher levels with a directional antenna through open cargo bay doors (Z_o 410).

(3) The allowable payload-to-remote manipulator system (RMS) limit for payloads attached to the RMS is defined as the radiation impinging upon an imaginary plane containing the RMS wrist roll joint end face, which is the mating interface for the standard end effector to the RMS.

(4) The allowable payload-to-RMS limit for payloads intentionally producing radiated fields while mounted in the cargo bay is defined as the radiation impinging on an imaginary surface 7.6 cm (3 inches) beyond the envelope of the actual surface of the payload in the ± X, ± Y, and +Z direction during RMS operation.

The above is in reference to radiation with the cargo bay doors open. No intentional radiation will be permitted with the doors closed.

Allowable levels of radiation from orbiter cabin payload or experiment transmitter systems are specified in section 10.7.3.2 of ICD 2-19001.

f. Radiated spurious and harmonic emissions from payload transmitter antennas shall have peak powers 80 dB down from the power at the fundamental (for harmonics greater than the third). Power at the second and third harmonics shall be suppressed by {50 + 10 Log(Peak Power in watts at the fundamental) dB}, or 80 dB whichever requires less suppression. These are the same limits as those for conducted spurious and harmonic emissions on antenna terminals in paragraph 2.5.2.1.f. When the MIL-STD-462 test CE06 for conducted emissions on antenna terminals cannot be applied, test RE03 for radiated spurious and harmonic emissions shall be used as an alternative test. Refer to MIL-STD-461C and 462 for details.

2.5.2.3 Acceptance Requirements - The emission requirements of 2.5.2 shall also apply to all previously qualified hardware.

2.5.3 Susceptibility Requirements

The following paragraphs on susceptibility tests shall be used to implement the susceptibility requirements of Table 2.5-1. Additional tests or test conditions should be considered by the project if the operational scenario, the launch site environment, or the design suggests such additions may be necessary. The worst-case levels of shuttle-produced emissions in the payload bay, as defined in ICD 2-19001, have been incorporated into the following requirements where applicable.

2.5.3.1 Conducted Susceptibility Requirements - The following conducted susceptibility design and test requirements shall be applied to power leads (both dc and ac for STS) and to antenna terminals of payload hardware:

a. Conducted Susceptibility CS01-CS02 (Powerlines) - The tests should be conducted over the frequency range of 30 Hz to 400 MHz in accordance with the limit requirements and test procedures of MIL-STD-461C and 462. If degraded performance is observed, the signal level should be decreased to determine the threshold of interference. Above 50 KHz, modulation of the applied susceptibility signal is required if appropriate. If the appropriate modulation has not been established by component design or mission application, the following guidelines for selecting an appropriate modulation will apply:

(1) AM Receivers - Modulate 50 percent with 1000-Hz tone.

(2) FM Receivers - While monitoring signal-to-noise ratio, modulate with 1000-Hz signal using 10-kHz deviation. When testing for receiver quieting, use no modulation.

(3) SSB Receivers - Use no modulation.

(4) Components With Video Channels Other Than Receivers - Modulate 90 to 100 percent with pulse of duration 2/BW and repetition rate equal to BW/1000 where BW is the video bandwidth.

(5) Digital Components - Use pulse modulation with pulse duration and repetition rate equal to that used in the component under test.

(6) Nontuned Components - Use 1000-Hz tone for amplitude modulation of 50 percent.

For STS payloads, the conducted susceptibility tests of paragraphs 2.5.3.1.a are performed on applicable hardware in keeping with two operational requirements derived from ICD 2-19001. The first requirement applies to payload hardware that operates on +28 volt power originating from one of the orbiter's dc power buses. The requirement is met with sawtoothed transient oscillations (between 500 and 700 Hz) on the powerlines with a maximum voltage envelope shown in either Figure 2.5-12a or Figure 2.5-12b depending on which orbiter bus is supplying the power. The bus voltage transients (caused by activation of the hydraulic circulation pump connected to the bus) may occur at any time during on-orbit operations, plus activation at touchdown, and are not subjected to preflight scheduling.

The second requirement applies to equipment which operates on orbiter-supplied ac power. The requirement is met with transient spikes on the ac buses as defined in Figure 2.5-13. For payload testing purposes, the impedance into which the spikes are generated is 50 ohms minimum for significant frequency components of the spikes.

b. Conducted Susceptibility CS03 (Two-Signal Intermodulation) - This test, which determines the presence of intermodulation products from two signals, should be conducted on receivers operating in the frequency range of 30 Hz to 18 GHz where this test is appropriate for that type of receiver. The items should perform in accordance with the limit requirements and the test procedures of MIL-STD-461C and 462 except that the operational frequency range of equipment subject to this test should be increased to 18 GHz and the highest frequency used in the test procedure should be increased to 40 GHz.

c. Conducted Susceptibility CS04 (Rejection of Undesired Signals) - Receivers operating in the frequency range from 30 Hz to 18 GHz should be tested for rejection of spurious signals where this test is appropriate for that type of receiver. The items should perform in accordance with the limit requirements and the test procedures of MIL-STD-461C and 462 except that the frequency range should be increased to 40 GHz.

d. Conducted Susceptibility CS05 (Cross Modulation) - Receivers of amplitude-modulated RF signals operating in the frequency range of 30 Hz to 18 GHz should be tested to determine the presence of products of cross modulation where this test is appropriate for that type of receiver. The items should perform in accordance with the limit requirements and test procedures of MIL-STD-461C and 462 except that the operational frequency range of equipment subject to this test should be increased to 18 GHz and the highest frequency used in the test procedure should be increased to 40 GHz.

e. Conducted Susceptibility CS06 (Powerline Transient) - A transient signal should be applied to powerlines in accordance with the procedures of MIL-STD-461C and 462. Because the applied transient signal should equal the powerline voltage, the resulting total voltage is twice the powerline level. The transient should be applied for a duration of 5 minutes at a repetition rate of 60 pps. The test should be applied to the input power leads of all payloads.

Changes in the method of describing powerline transients (line-to-line in lieu of line-to-structure) in JSC ICD-2-19001 reveal that STS payloads could be exposed to powerline transient voltages in excess of these levels. (Refer to paragraphs 7.3.7.2 and 7.3.7.4 of ICD-2-19001.) Payloads should be designed with this in mind, and tested to these ICD levels at the STS interface.

2.5.3.2 Radiated Susceptibility Requirements - The following tests shall be applied to individual payloads and payload subsystems. The tests are based on MIL-STD-461C and 462, as supplemented.

a. Radiated Susceptibility Test RS03 (E-field) - The payload shall be exposed to external electromagnetic signals in accordance with the requirements and test methods of test RS03. Intentional E-field sensors on payloads that operate within the frequency range of the test shall be removed or disabled without otherwise disabling the payload during the test. The test shall demonstrate that spacecraft (exclusive of E-field sensors) can meet their performance objectives while exposed to the specified levels. Modulation of the applied susceptibility signal is required. If the appropriate modulation has not been established by hardware design or mission scenario, then 50% amplitude modulation by a 100 Hz square wave should be considered. When performing additional testing at discrete frequencies of known emitters, the modulation characteristics of the emitter should be simulated as closely as possible.

(1) ELV-launched spacecraft or STS payloads not operated or checked out before release from the orbiter:

o 2 V/m over the frequency range of 14 kHz to 2 GHz.

o 5 V/m over the frequency range of 2 to 12 GHz.

o 10 V/m over the frequency range of 12 to 18 GHz; applicable only to spacecraft with a Ku band telemetry system.

(2) Orbiter attached payloads and free flyers operated or checked out before release from the orbiter:

o 2 V/m over the frequency range of 14 KHz to 2 GHz. (Other payloads are permitted to radiate in excess of these levels after the payload bay doors are opened. Refer to 2.5.2.2.e and Figure 2.5-11. If it is determined that the payload will be exposed to higher levels than 2 V/m, the requirements should be revised to reflect those higher levels at the specific frequencies involved.)

o 20 V/m over the frequency range of 2 to 18 GHz. The 20 V/m level is required since other payloads are permitted to radiate these levels after the payload bay doors are opened; refer to 2.5.2.2.e and Figure 2.5-11. Also, a payload element could be exposed to these levels at S-band if it is within 2 meters of the payload bay forward bulkhead; refer to Figure 2.5-14a.

For both STS and ELV payloads, the EMI test levels (or frequency range) should be increased if it is determined that onboard telemetry systems, another payload, or other signals in space could expose a payload to higher levels than the above test levels. Systems such as ground based radars are known to produce signals in space in excess of 2 V/m at frequencies at least as low as 400 MHz.

STS Applications

Payloads not operated or checked out before release from the orbiter shall be tested to ensure proper performance after a 6-minute minimum exposure to E-field levels of 20 V/m during which the frequency is uniformly swept from 2 to 18 GHz. This test shall be conducted with the payload powered down.

Free flyers could be exposed to the main beam of the orbiter's Ku-band transmitter after being released from the orbiter, or an attached payload could be exposed after deployment. Refer to paragraph 10.7.2.2 of the ICD 2-19001 for the Ku-band levels. Projects may choose to negotiate operational constraints with JSC to avoid exposure of the payload rather then design and test to those high Ku-band levels. Any agreements with JSC shall be defined in the payload integration plan.

During deployment, after release, or during retrieval, payloads could be exposed to levels greater than 20 V/m from the orbiter's S-band transmitters or the ERPCL S-band transmitter. Refer to paragraph 10.7.2.2 of the ICD-2-19001 and Figures 2.5-14a through 2.5-14e.

The maximum field intensities associated with the transmitters supporting an EVA crewman are 6.5 volts per meter at one meter from the TV antenna of the EMU and 3.8 volts per meter at one meter from the EMU EVA voice antenna. Transmitter characteristics associated with EVA activities are given in Table 2.5-2. Payloads that could be exposed to these EVA emissions shall be designed to meet these induced environments. [Note: The TV antenna and the voice antenna are both located on the man.]

There is also a Wireless Crew Communications System (WCCS) operating in the orbiter crew compartment at frequencies between 338.0 MHz and 392.0 MHz. (Refer to paragraph 10.7.2.2 of the ICD-2-19001).

b. Operational Compatibility of Attached Payloads with the Orbiter's Intentional (Transmitter) Emissions - Payloads designed to operate in the orbiter bay that contain sensors or devices that are inherently susceptible to EMI shall be tested to demonstrate that they can meet their performance requirements while exposed to the radiated emissions from the orbiter's transmitters. The levels, defined in Figure 2.5-14a, are worst-case values in the upper (+Z) quadrant of the payload envelope with the bay doors open. Although reduced levels can usually be expected in the lower levels of the bay, the levels are dependent on the geometry of the payload. Table 2.5-2 gives the frequency range and modulation associated with the orbiter transmitter field strengths, which are given in Figure 2.5-14a.

Testing shall be in accordance with test RS03 utilizing the actual orbiter, adjacent payload, and EVA transmitter frequencies and levels as applicable. All payload sensory devices shall be connected and operating. Appropriate modulation of the test signals shall be based on the modulation types defined in Table 2.5-2. The test signal antenna shall be positioned to provide appropriate simulation of the operation of the payload while it is exposed to intended emissions from the orbiter's transmitters.

Table 2.5-2
Frequency Range and Modulation Associated
With Orbiter Transmitters

Transmitter	Frequency	Modulation
S-Band Hemi	2000-2300 MHz	FM
S-Band Quad	2200-2300 MHz	PSK,PM
S-Band Payload	2000-2200 MHz	PSK,PM,FM/P
Ku-Band	13-15 GHz	PSK,FM,Pulse
UHF-(EVA)	259.7, 279.0 MHz	AM Voice and Data

c. Operational Compatibility of Attached Payloads With the Orbiter's Unintentional Emissions - Payloads that are designed to operate in the payload bay of the orbiter and that contain sensors or devices that are inherently susceptible to EMI shall be tested with their sensors operating in order to demonstrate that they can meet performance requirements while exposed to unintentional radiated emissions from the orbiter. The test levels shall be in accordance with the orbiter's radiated narrowband E-field limits given in Figure 2.5-15 and the orbiter's broadband emission limits given in Figure 2.5-16.

The tests shall be in accordance with RS03. The test signal antenna shall be located so as to simulate payload operation while it is exposed to the orbiter's radiated emissions.

d. Magnetic Field Susceptibility - Payloads that could be susceptible to the magnetic field levels generated by their own subsystems and components, or STS payloads that could be susceptible to the magnetic fields generated by the STS, shall be tested for susceptibility in a suitable test facility. The tests shall be performed to expected/acceptable levels from 30 Hz to 50 KHz and/or in a static (dc) field.

This requirement may be deleted with project approval for payloads that do not include subsystems or instruments that are inherently susceptible to magnetic fields.

The minimum test levels to satisfy STS requirements are given in paragraph 10.7.2.2. of ICD 2-19001. The magnetic field susceptibility portion may be deleted with project approval.

2.5.3.3 Acceptance Requirements - The susceptibility requirements of 2.5.3 shall apply to all previously qualified hardware.

2.5.4 Magnetic Properties*

A spacecraft whose magnetic properties or fields must be controlled to satisfy operational or scientific requirements, shall be tested at the component, subsystem, and spacecraft levels of assembly, as appropriate, and shall meet the following magnetic requirements (spacecraft with magnetic sensors, e.g., magnetometers, may have more stringent requirements):

2.5.4.l Initial Perm Test - The maximum dc dipole moment produced by a spacecraft and by each of its components following manufacture shall not exceed 3.0 and 0.2 AM^2 (dipole moment), respectively.

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

2.5.4.3 Perm Levels After Exposures to Deperm Test - The maximum dipole moment produced by a spacecraft and each of its components after exposures to magnetic field deperm levels of 30 X 10^{-4} tesla for spacecraft and 50 X 10^{-4} tesla for components shall not exceed 2.0 and 0.l AM^2, respectively.

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

2.5.4.5 Stray Magnetic Field Measurements - A spacecraft and each of its components shall not produce dipole moments due to internal current flows in excess of 0.5 and 0.05 AM^2, respectively.

2.5.4.6 Subsystem Requirements - Subsystems shall also be tested in accordance with the above requirements; however, the requirement limits shall be determined on a per case basis. The limits shall be designated between the levels for the spacecraft and those for components and shall depend upon the number of components in a subsystem and the number of subsystems in the spacecraft. Subsystem limits shall be designated such that the fully integrated spacecraft can meet its magnetic requirements.

2.5.4.7 Acceptance Requirements - The provisions for magnetic testing (2.5.4) shall apply to all previously qualified hardware.

---

* Dc magnetics testing should be performed after vibration testing. This provides an opportunity to correct for any magnetization of the flight hardware caused by fields associated with the vibration test equipment.

2.5.4.8 Notes on Magnetics Terminology and Units Used In GEVS

Induced Field - If a low level magnetic field is applied to the hardware, the measured change in the magnetic field may be different from the applied field. This difference is called the induced field. The induced field disappears when the applied field is turned-off. The induced field is measured with the hardware turned-off. The low level applied field is approximately equal to the Earth field.

Stray Fields - Magnetic fields that are generated by current flowing within the spacecraft and its experiments.

Perm Levels - This is the permanent magnetic field of the hardware. This permanent magnetic field is actually a function of its history of exposure to magnetic fields.

Deperm - The process of demagnetizing the hardware with the purpose of reducing the effects of any previous environmental field exposures.

The product of the area of a plane loop of wire and the dc current flowing in the loop is called the magnetic dipole moment. At distances sufficiently removed from the hardware, the magnetic flux density (B-field) can approximately be modeled as if it were produced by such a loop. Under such conditions, the magnetic dipole moment becomes a measure of the B-field.

Comparison of the "Perm Levels After Exposure to Deperm Test" with the "Perm Levels After Exposure to Magnetic Field" gives an indication of the amount of soft magnetic material present in the s/c hardware.

Induced magnetism has historically, been the major factor preventing accurate calculation of the s/c dipole moment from the measured dipole moments of all of the major subunits of the s/c.

The 15 gauss exposure level in the GEVS is based on worst case field levels expected in the vicinity of shaker tables used during environmental testing.

The stray field measurements are designed so that it is possible to differentiate between the power-on vs. power-off conditions of operation as well as shifts in the stray-field levels during operation of the equipment.

The magnetic flux density (B) is expressed in units of Tesla (Weber/meter-squared) in the mks system.

The magnetization M of a material is defined as the magnetic (dipole) moment per unit volume. In the mks system, the units of M are ampere/meter.

The magnetic field strength (H) is often expressed in units of ampere/meter; this is the same units as M. But it is also often expressed in the units of B in lieu of the units of M; this is one of the sources of ambiguity in magnetics units.

Historically "magnetic charge" was defined as an analog to "electric charge." The magnetic "pole" is a unit of "magnetic charge." Even the existence of magnetic charges has not been established, but this mathematical analog sometimes proves useful.

Examples Of Considerations And Situations That Occur

Measurement of hysteresis and eddy current losses can be performed in a test facility that can produce a rotating magnetic field.

Hysteresis effects - The (irreversible) magnetic field characteristics of ferromagnetic materials (hysteresis) result in energy dissipation in the materials under conditions of spacecraft hardware spinning in a magnetic field. The disturbance torques produced in the process can act to despin the spinning part of a spacecraft. On Transit 1B, this effect was used to despin the satellite. Eight 31 inch long rods mounted orthogonal to the spin axis were used to accomplish this. (The rods were made of a soft magnetic material).

Eddy Currents - Eddy currents in a material are caused by time-varying magnetic fields. These currents may act to despin the spinning part of a spacecraft. Eddy currents would be possible even in the absence of spacecraft generated magnetic fields.

Disturbance torques can result from spacecraft hardware that rotates relative to other hardware on the spacecraft.

The magnetic disturbance torque acting on a spacecraft is equal to the cross product of the magnetic dipole moment of the spacecraft and the magnetic flux density.

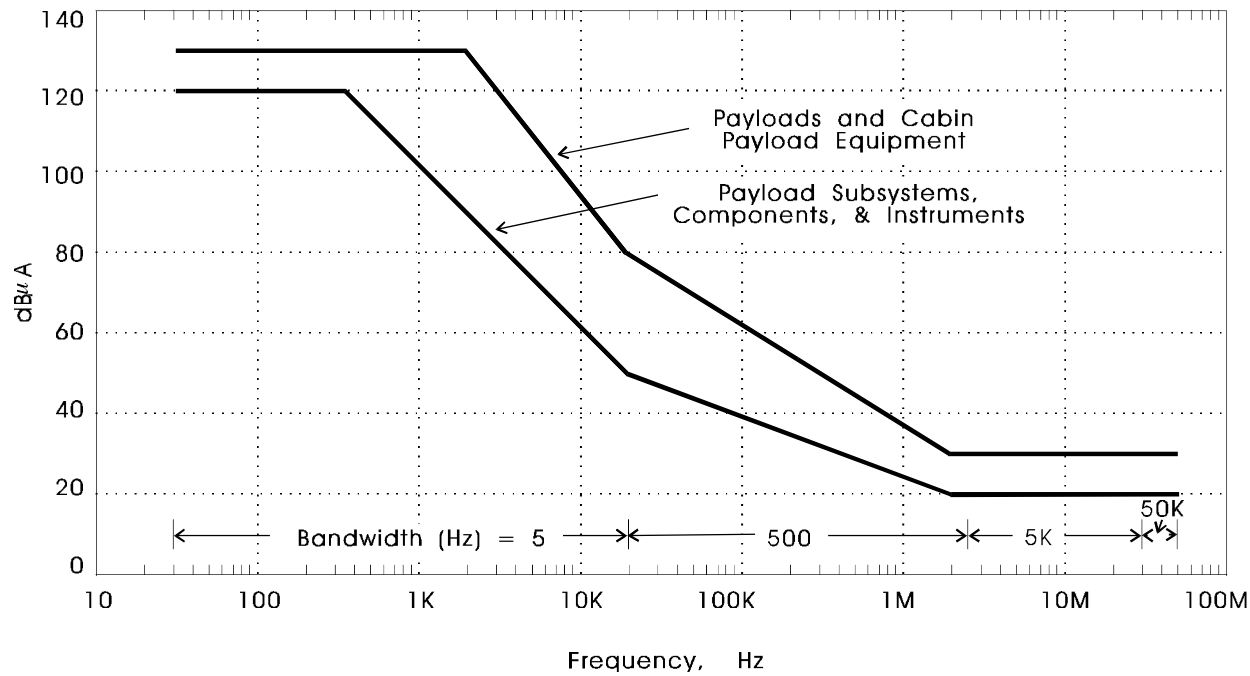


Figure 2.5-1 Narrowband Conducted Emission Limits on Payload Power Lines

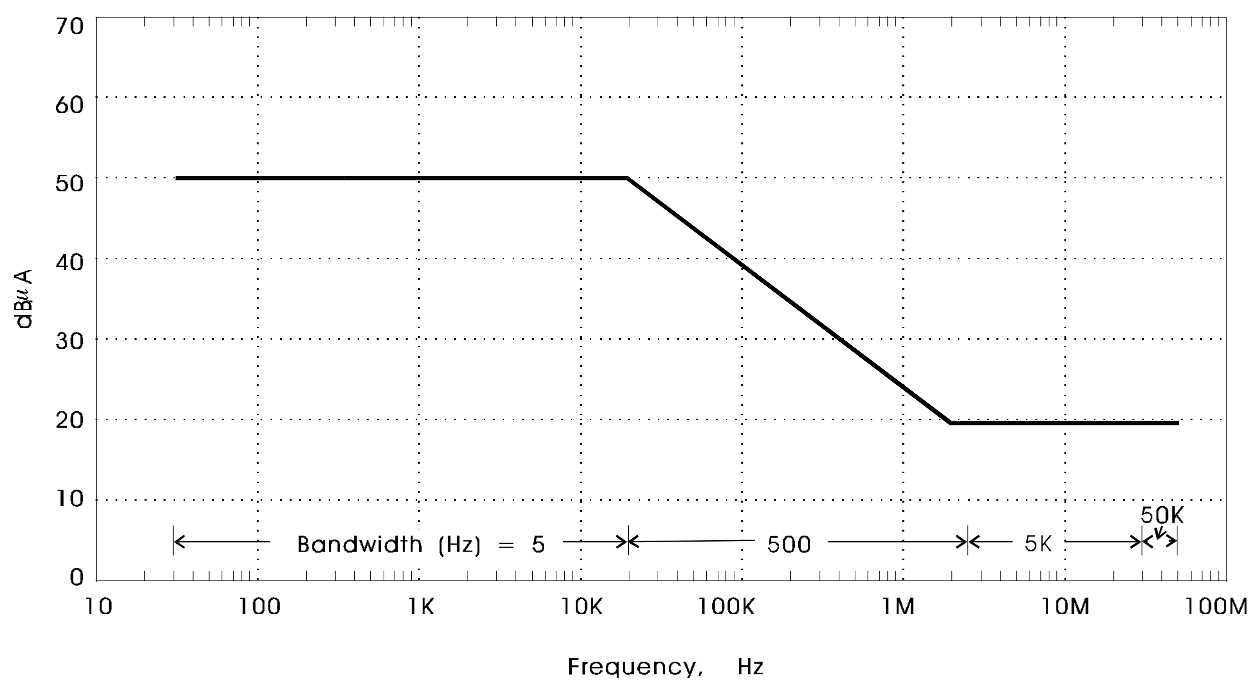


Figure 2.5-1a Common Mode Conducted Emission Limits on Primary Power Lines

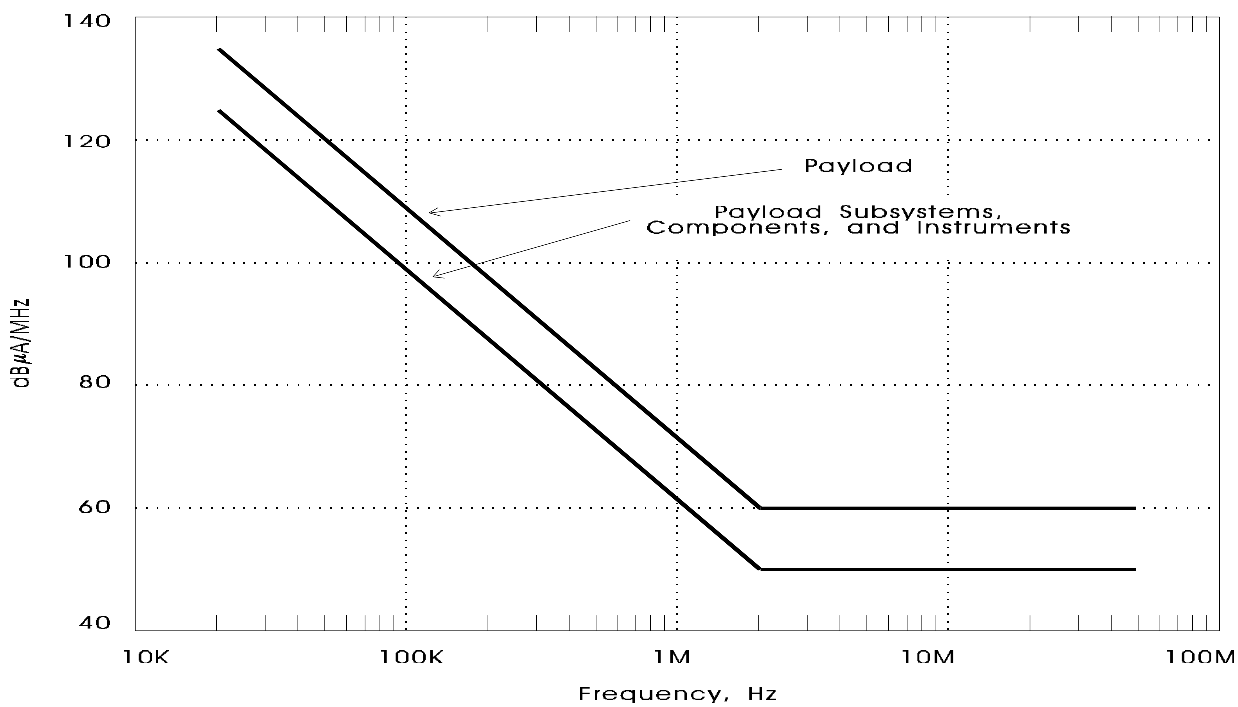


Figure 2.5-2 Broadband Conducted Emission Limits on Payload Power Lines

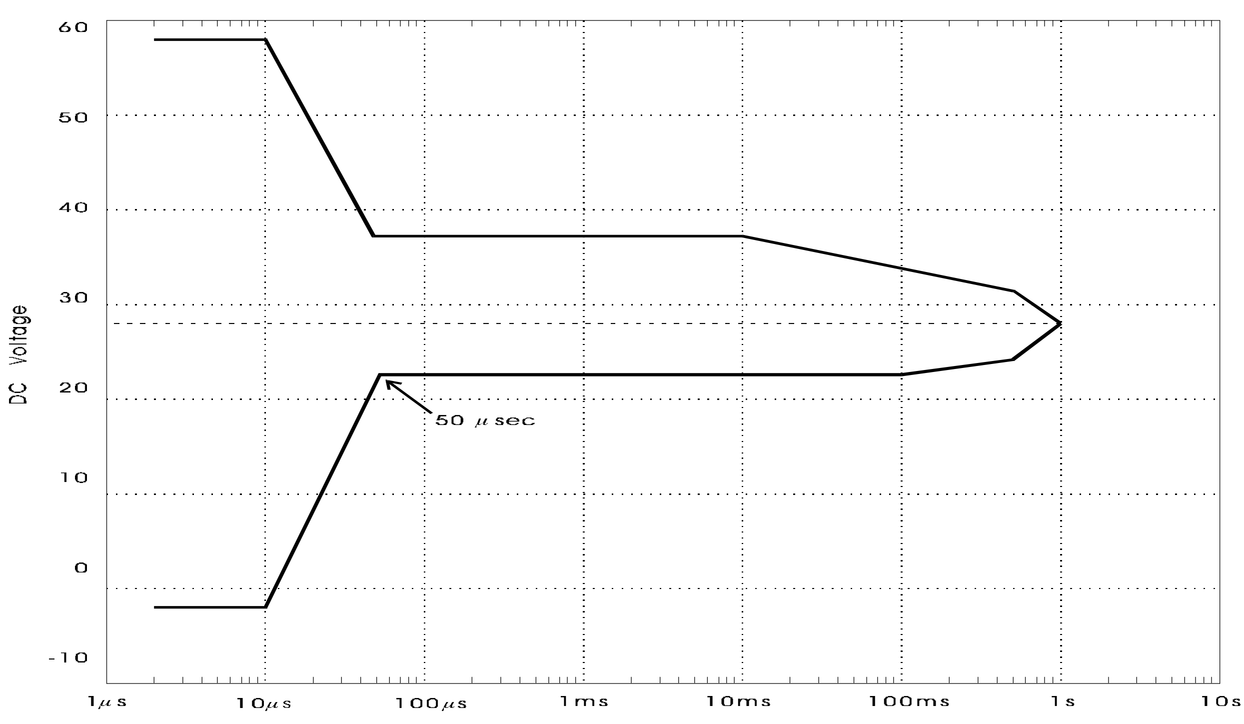


Figure 2.5-3 Limit Envelope of Cargo-Generated Transients (Line-to-Line) on DC Power Busses for Normal Electrical System

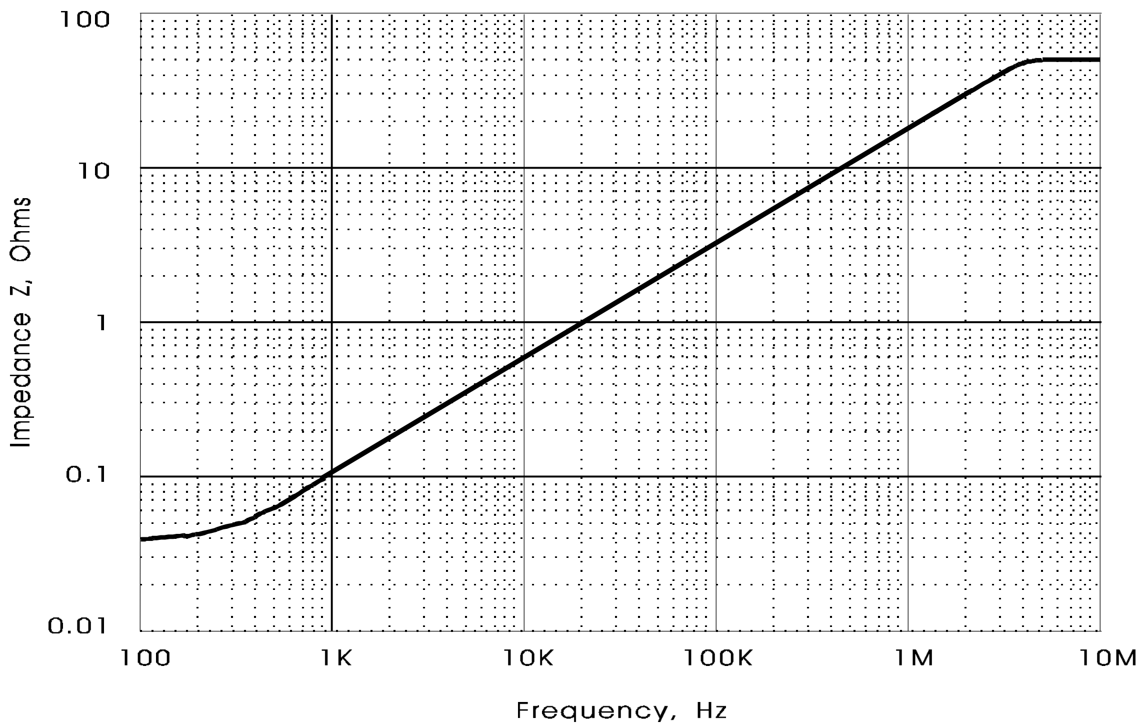
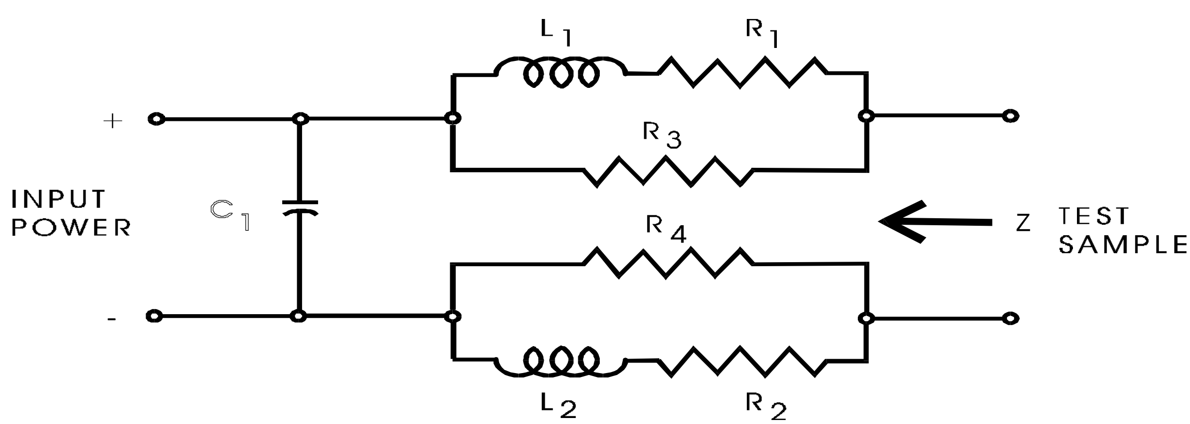


Figure 2.5-4 Orbiter DC Powerline Impedance



R1,R2 = 0.25 ohm*
R3,R4 = 25 ohm
C1 = 19,000 μ F (75 V ELECTROLYTIC)
L1,L2 = 4 μ H

* Value of resistors may be reduced to 0.025 ohms or lower for hardware requiring high levels of power currents.

Figure 2.5-5 Network Schematic for Simulating Impedance of Orbiter Power System

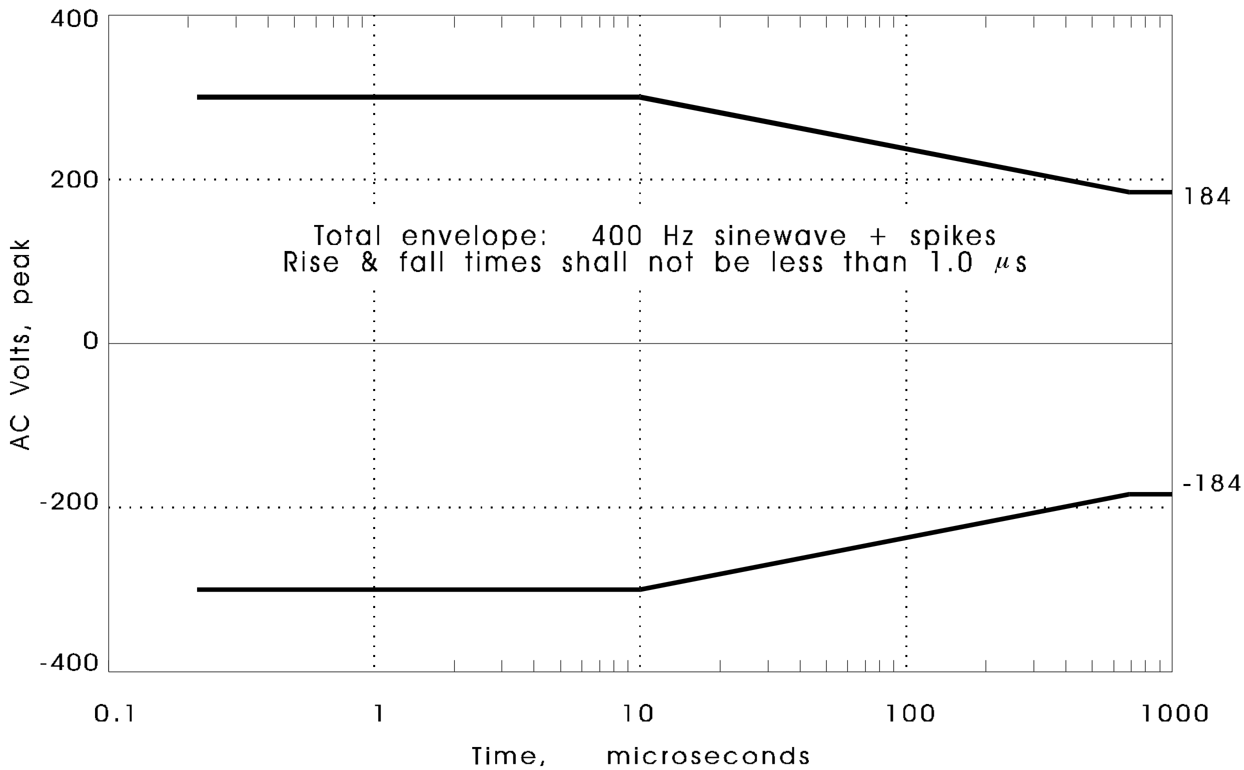


Figure 2.5-6 Limits of Payload-Produced Spikes on Orbiter AC Power Leads

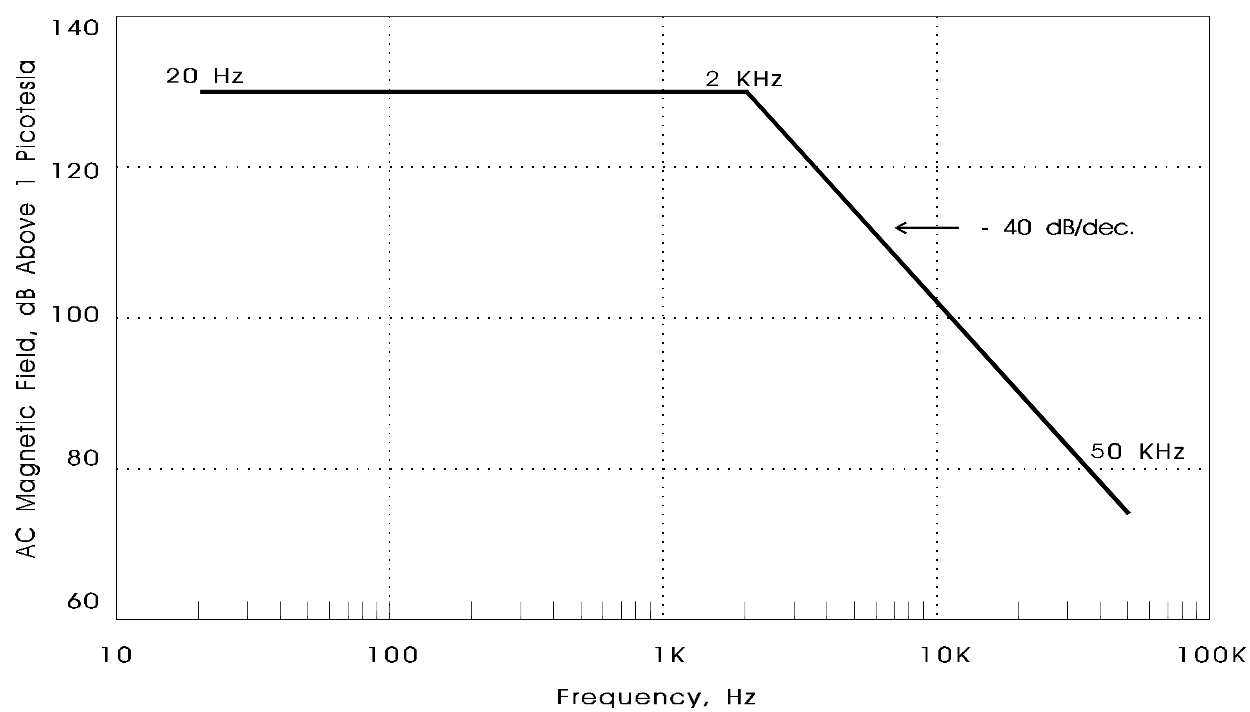


Figure 2.5-8 Limits of Radiated AC Magnetic Field at 1 Meter From Orbiter Payload

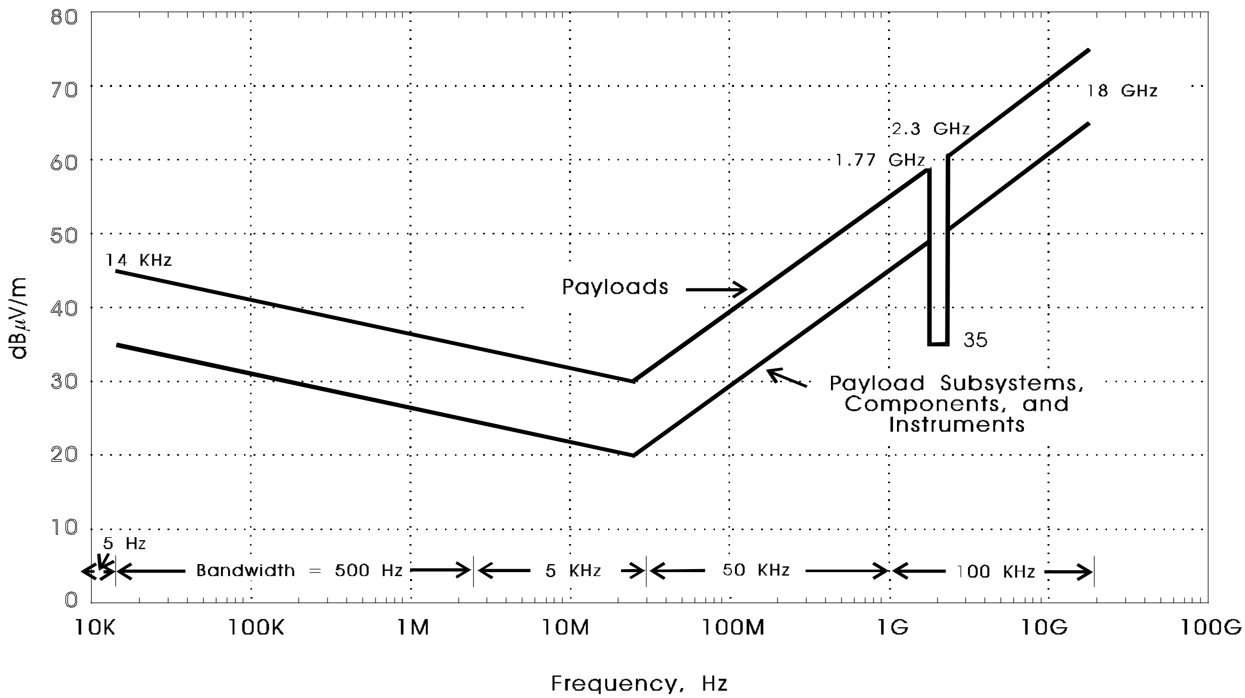
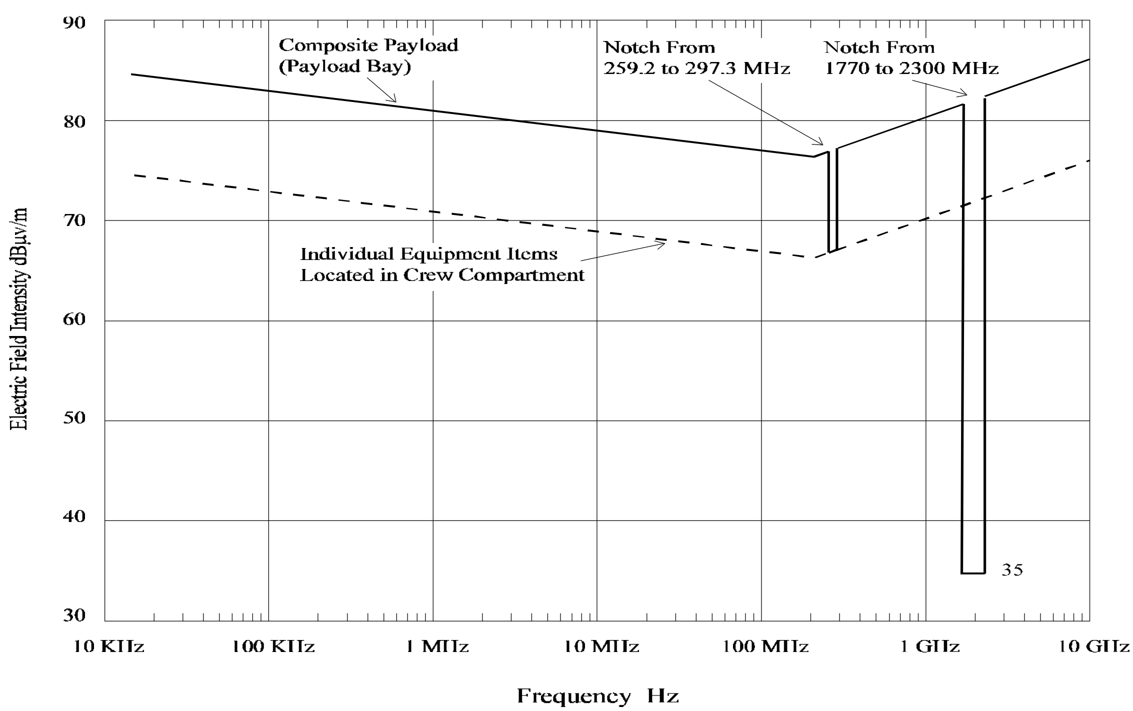


Figure 2.5-9 Unintentional Radiated Narrowband Limits for Electric Field Emission Produced by Payloads and Payload Subsystems



.Figure 2.5-9a Allowable Unintentional Radiated Narrowband Emissions Limits in Orbiter Cargo Bay

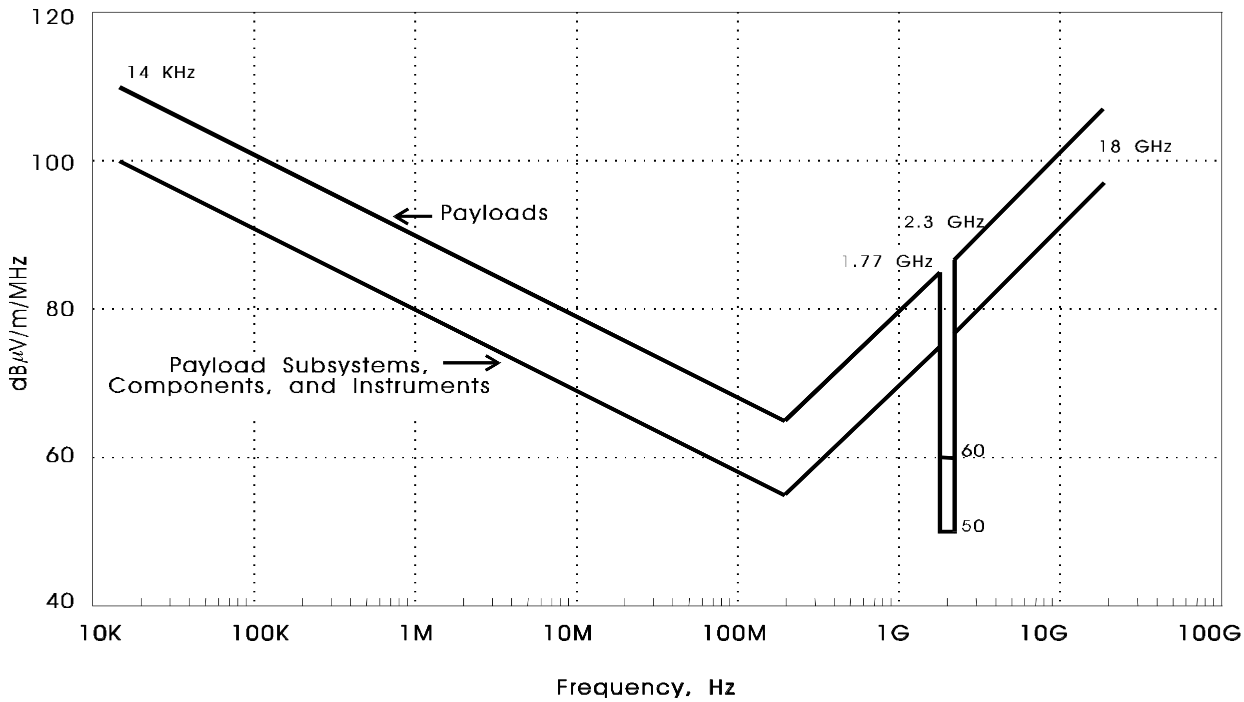


Figure 2.5-10 Unintentional Radiated Broadband Limits for Electric Field Emissions Produced by Payloads and Payload Subsystems

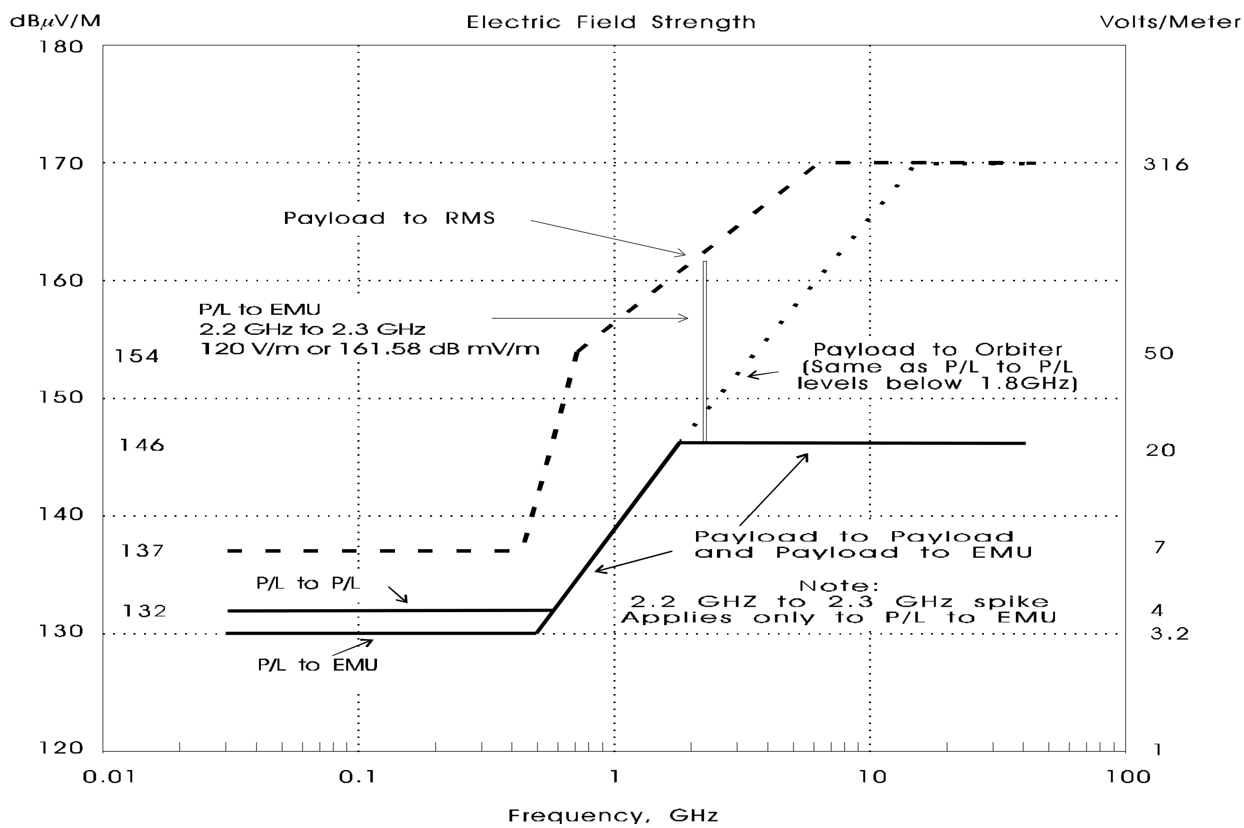


Figure 2.5-11 Allowable Intentional Field Strength in Orbiter Cargo Bay

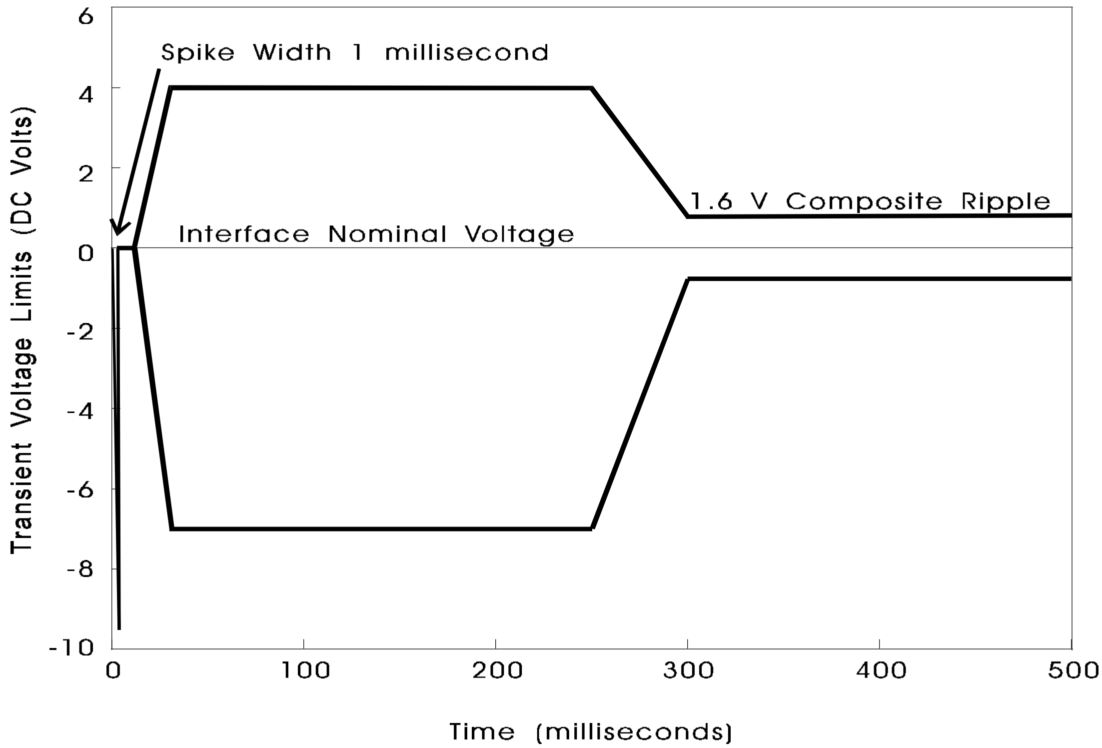


Figure 2.5-12a Transient Voltage on the Aft Payload B and C DC Buses Produced by Operation of the Hydraulic Circulation Pump

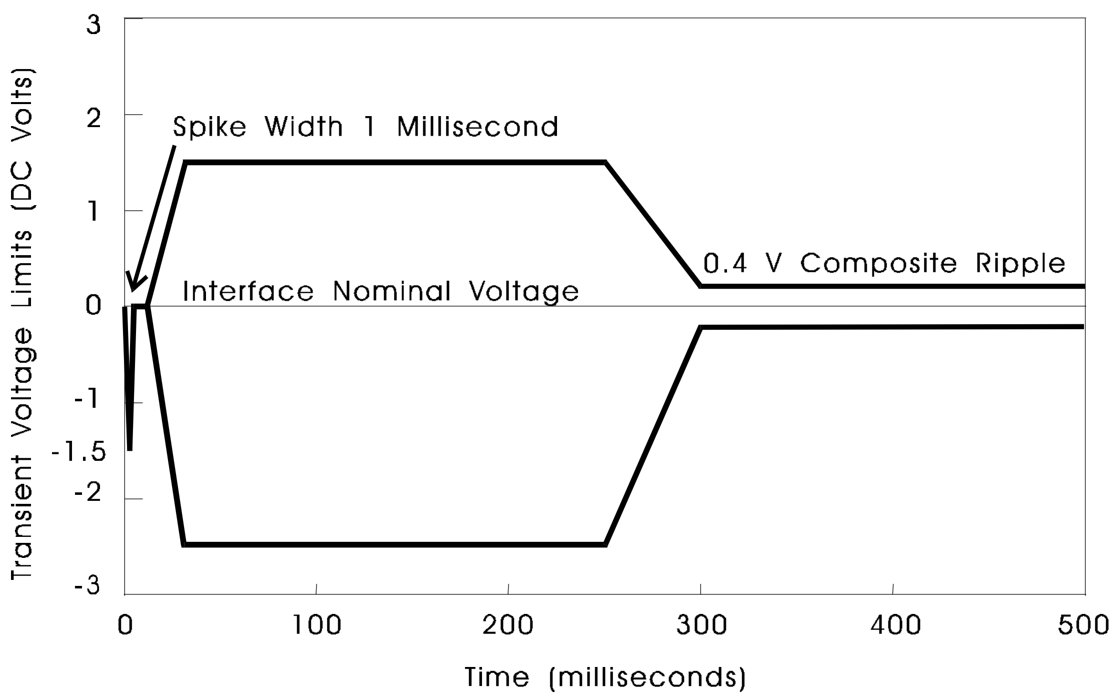


Figure 2.5-12b Transient Voltage on the Primary P/L Bus, Aux P/L A, AUX P/L B, and the Cabin P/L Bus at the Cargo Element Interface Produced by Operation of the Hydraulic Circulation Pump

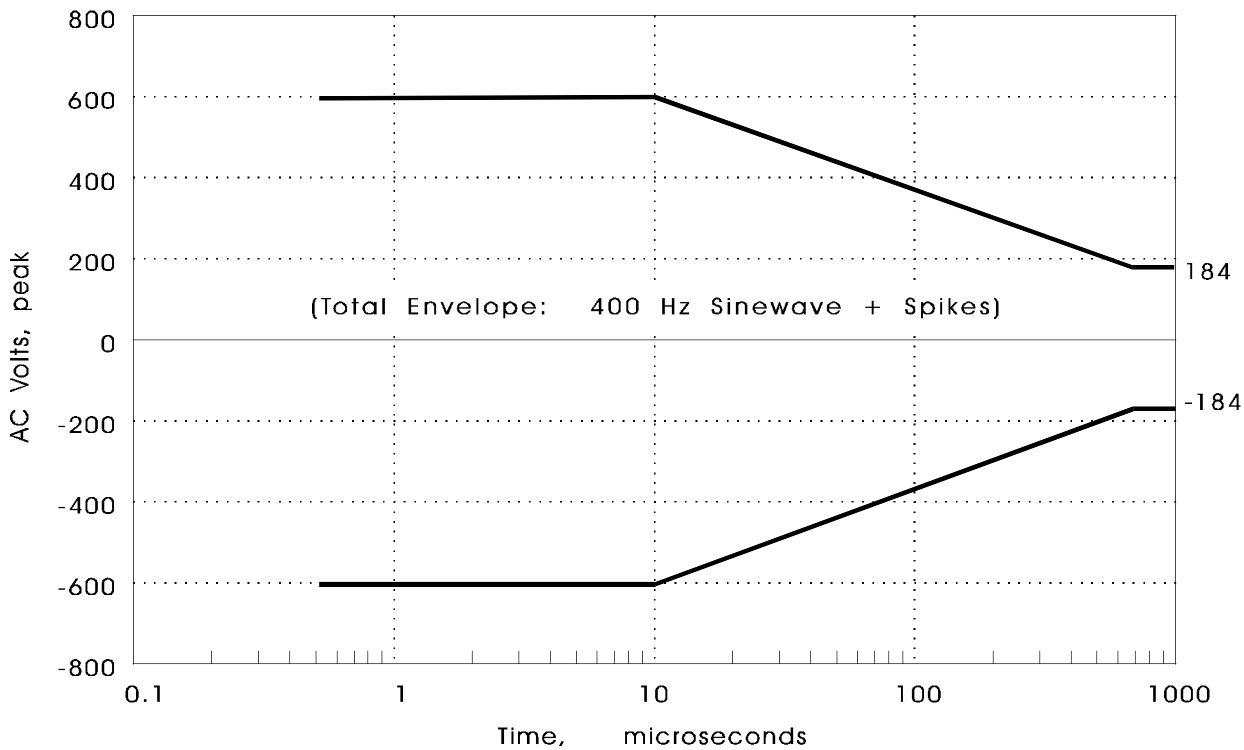


Figure 2.5-13 Envelope of Spikes on the Orbiter AC Power Bus

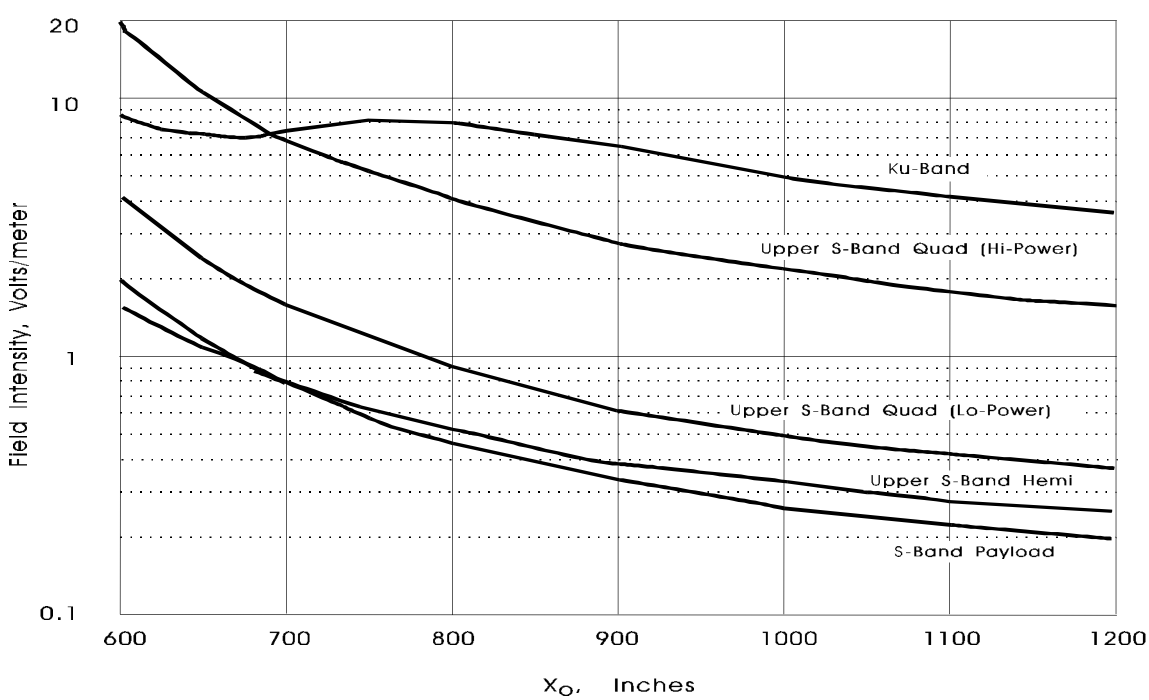
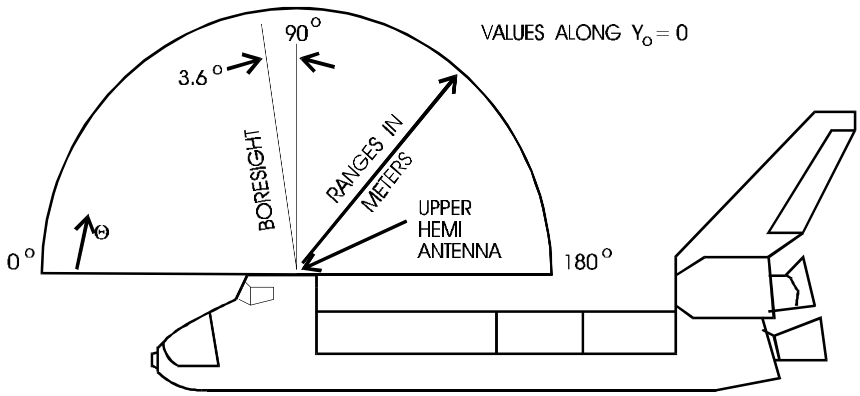


Figure 2.5-14a Maximum Field Intensities on Payload Envelope Produced by Orbiter Transmitters



For ranges greater than 1 meter:

$$\text{Volts/meter @ Desired Range} = \frac{\text{Volts/meter @ 1 meter}}{\text{Range in meters}}$$

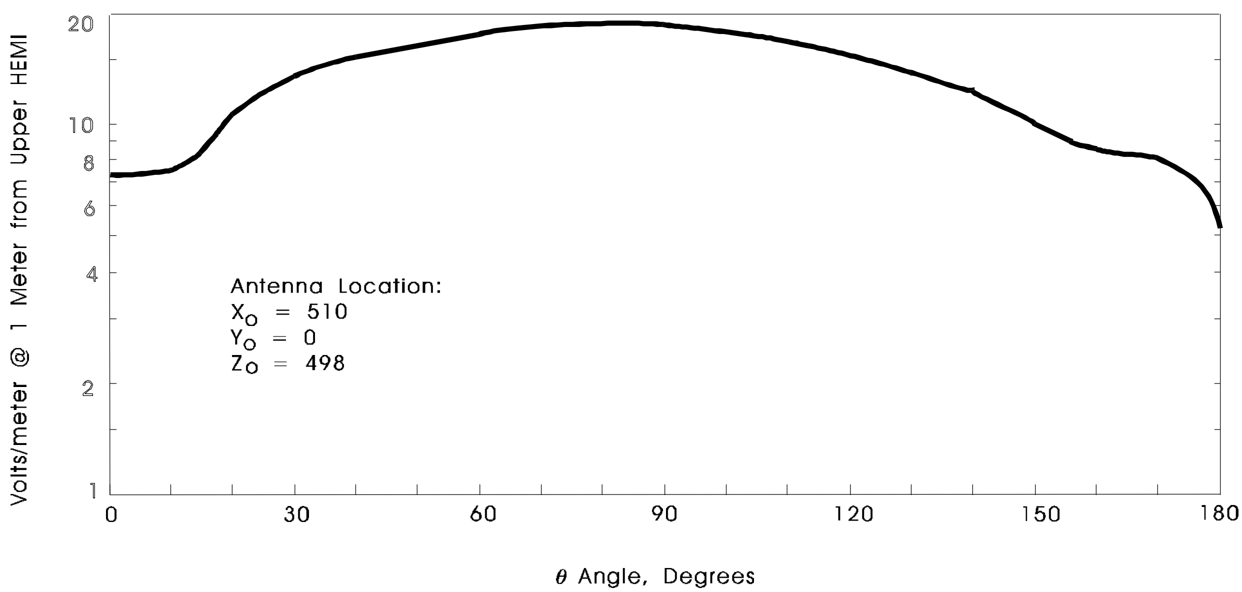
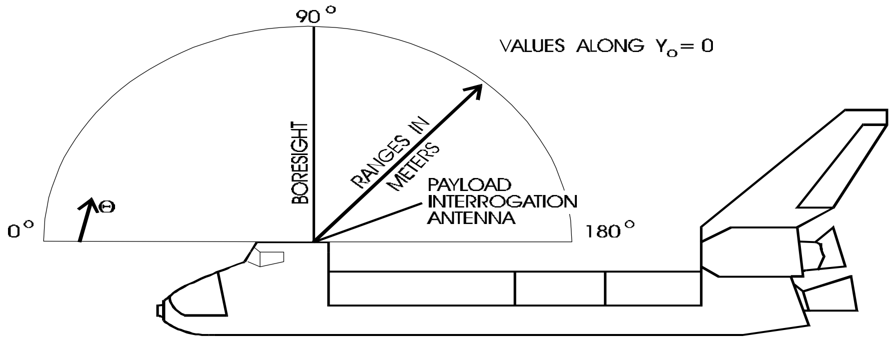


Figure 2.5-14b S-Band FM Transmitter, Upper HEMI Antenna, Maximum Field Intensities



For ranges greater than 1 meter:

$$\text{Volts/meter @ Desired Range} = \frac{\text{Volts/meter @ 1 meter}}{\text{Range in meters}}$$

For the medium power mode multiply volts/meter by 0.316 (-10 dB)

For the low power mode multiply volts/meter by 0.071 (-23 dB)

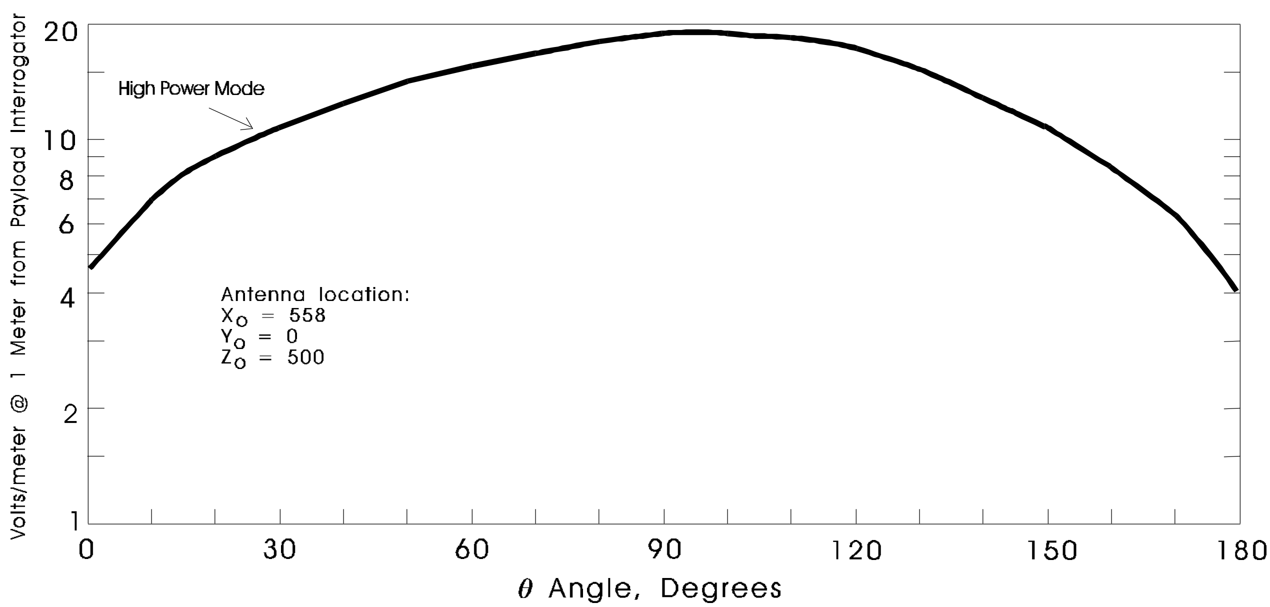


Figure 2.5-14c S-Band Payload Interrogator, Maximum Field Intensities

For ranges greater than 1 meter:

$$\text{Volts/meter @ Desired Range} = \frac{\text{Volts/meter @ 1 meter}}{\text{Range in meters}}$$

For the low power mode multiply Volts/meter by 0.158 (-16 dB)

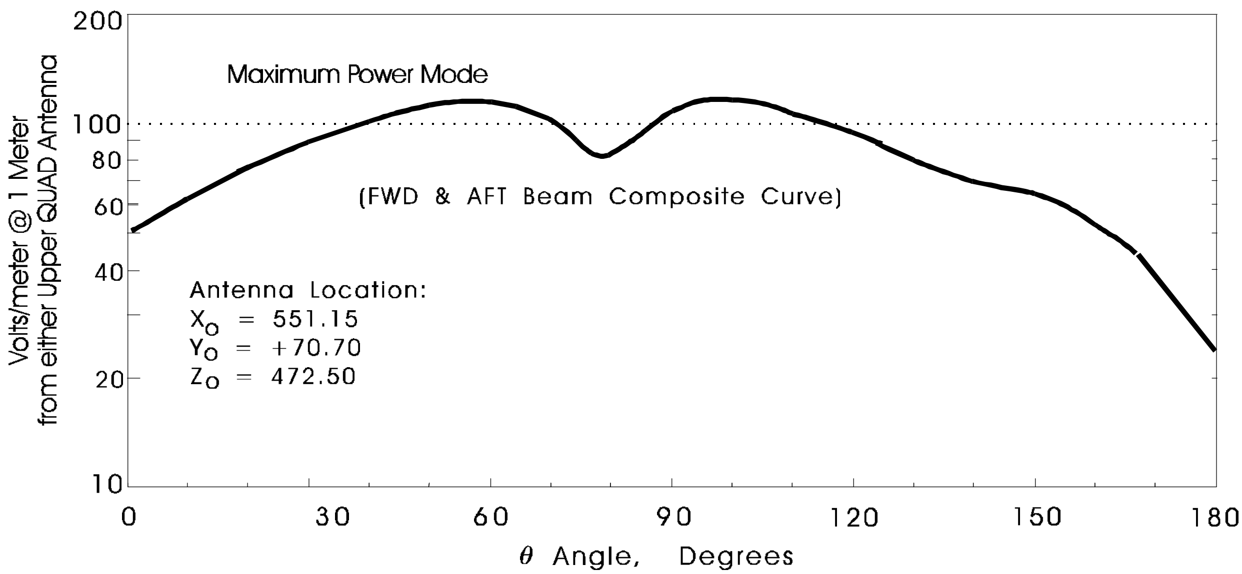


Figure 2.5-14d S-Band Network Transponder, Upper Quad Antennas, Maximum Field Intensities

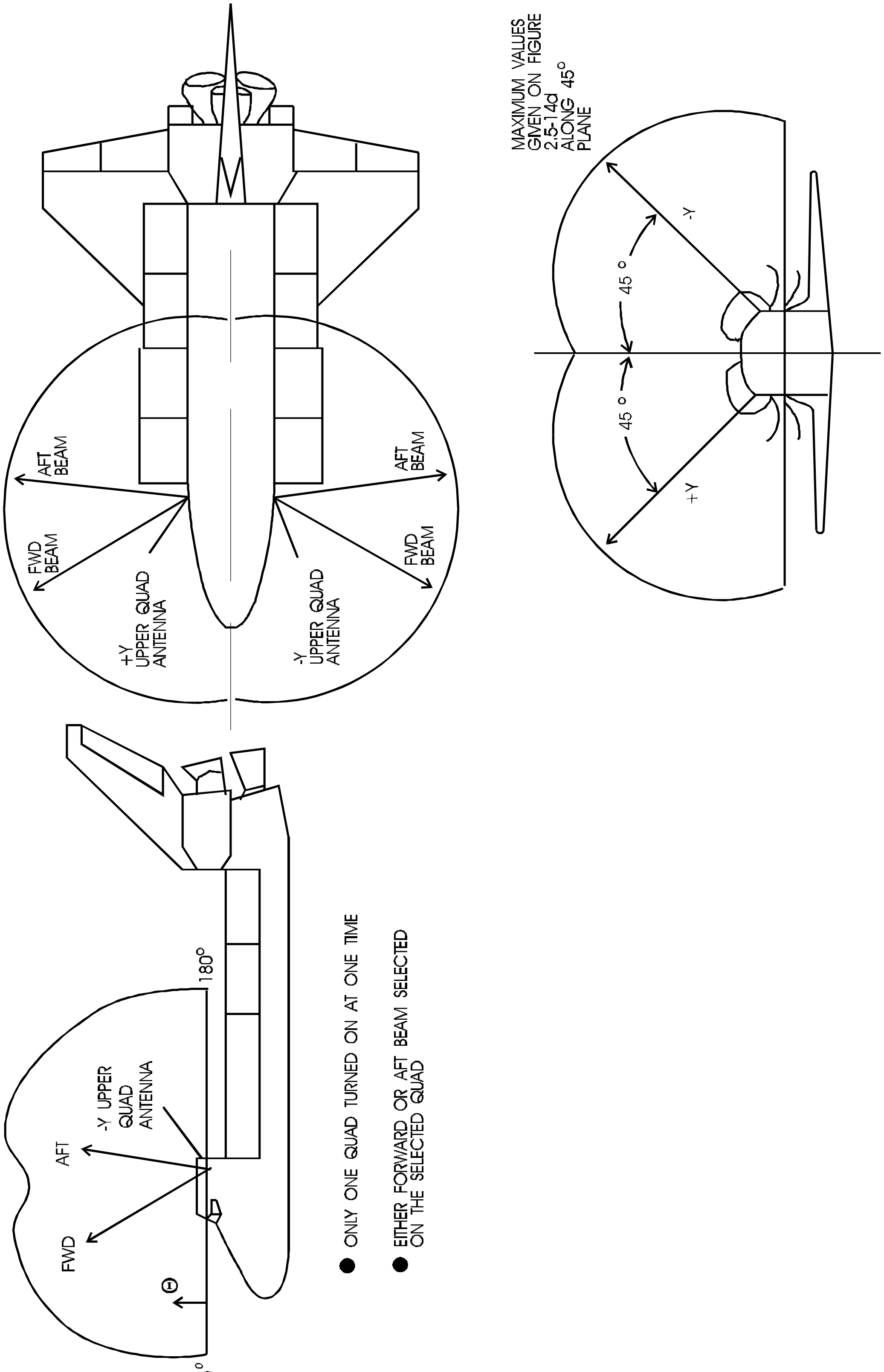


Figure 2.5-14e. S-Band Network Transponder, Upper Quad Antennas, Beam Configuration

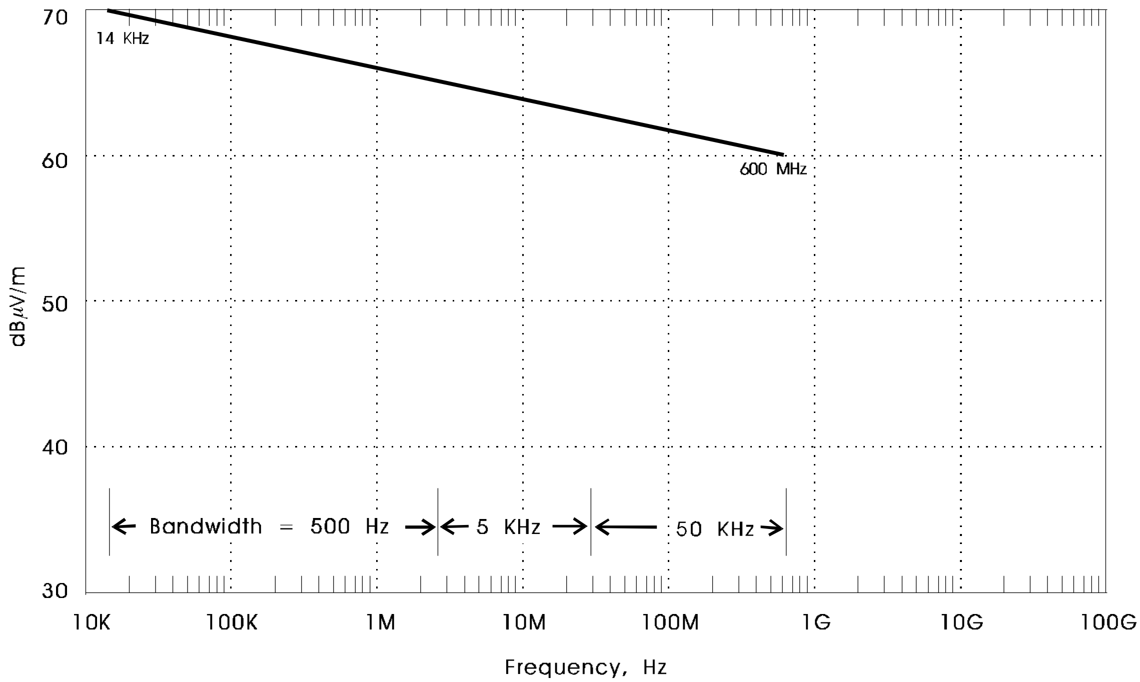


Figure 2.5-15 Orbiter Produced Radiated Narrowband Emissions in Payload Bay

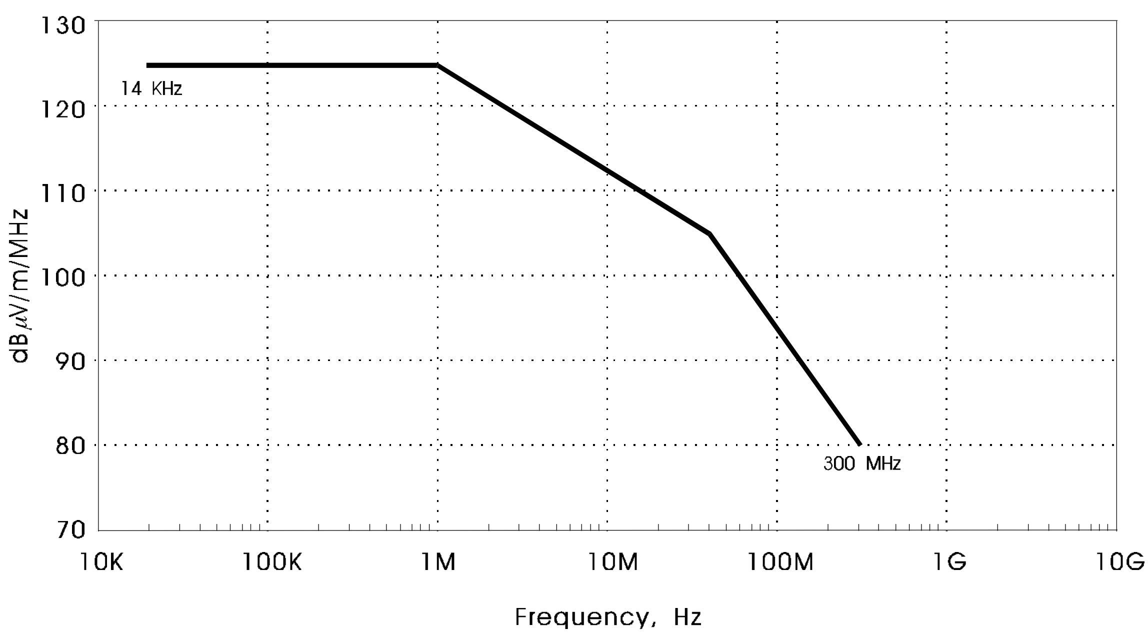


Figure 2.5-16 Orbiter Produced Radiated Broadband Emissions in Payload Bay

SECTION 2.6

THERMAL

2.6 VACUUM, THERMAL, AND HUMIDITY VERIFICATION REQUIREMENTS

The vacuum, thermal, and humidity requirements herein apply to STS and ELV payloads (spacecraft). An appropriate set of tests and analyses shall be selected to demonstrate the following payload or payload equipment capabilities.

a. The payload shall perform satisfactorily within the vacuum and thermal mission limits (including launch and return as applicable).

b. The thermal design and the thermal control system shall maintain the affected hardware within the established mission thermal limits during planned mission phases.

c. The hardware shall withstand, as necessary, the temperature and humidity conditions of transportation, storage, the orbiter cargo bay, and the orbiter manned spaces.

d. The quality of workmanship and materials of the hardware shall be sufficient to pass thermal cycle test screening in vacuum, or under ambient pressure if appropriate.

2.6.l Summary of Requirements

Table 2.6-1 summarizes the tests and analyses that collectively will fulfill the general requirements of 2.6. Tests noted in the table may require supporting analyses. The order in which tests or analyses are conducted shall be determined by the project and set down in the environmental verification plan, specification, and procedures (2.1.1.1.1 and 2.1.1.4). It is recommended, however, that mechanical testing occur before thermal testing at the systems level.

Payloads mounted in pressurized compartments of the orbiter need not be qualified for the vacuum environment, but the thermal cycling requirements of paragraph 2.6.2 do apply. These payloads must also be qualified for proper thermal performance.

The thermal cycle fatigue life test requirements of 2.4.2.1 also apply for hardware (e.g., solar arrays) susceptible to thermally induced mechanical fatigue.

The qualification and acceptance thermal-vacuum verification programs are the same except that a 10°C temperature margin is added in the thermal-vacuum test to qualify prototype or protoflight hardware.

2.6.2 Thermal-Vacuum Qualification

The thermal-vacuum qualification program shall ensure that the payload operates satisfactorily in a simulated space environment more severe than expected during the mission.

2.6.2.l Applicability - All flight hardware shall be subjected to thermal-vacuum testing in order to demonstrate satisfactory operation in modes representative of mission functions at the nominal operating temperatures, at temperatures in excess of the extremes predicted for the mission, and during temperature transitions. The tests shall demonstrate satisfactory operation over the range of possible flight voltages. In addition, hot and cold turn-on shall be demonstrated where applicable.

TABLE 2.6-1

VACUUM, THERMAL, AND HUMIDITY REQUIREMENTS

Requirement	Payload or Highest Practicable Level of Assembly	Subsystem including Instruments	Unit/ Component
Thermal-Vacuum[1,2]	T	T	T
Thermal Balance[1,3]	T and A	T,A	T,A
Temperature-Humidity[3] (Manned Spaces)	T/A	T/A	T,A
Temperature-Humidity[4] (Descent & Landing)	T/A	T/A	T,A
Temperature-Humidity[5] (Transportation & Storage)	A	A	A
Leakage[6]	T	T	T

1 Applies to hardware carried in the unpressurized cargo bay of the orbiter, and to ELV-launched hardware.

2 Temperature cycling at ambient pressure may be substituted for thermal-vacuum temperature cycling if it can be shown analytically to be acceptable.

3 Applies to flight hardware located in pressurized area which support payloads in the cargo bay.

4 Applies to hardware that must retain a specified performance after return from orbit and is carried in the unpressurized cargo bay.

5 Consideration should be given to environmental control of the enclosure.

6 Hardware that passes this test at a lower level of assembly need not be retested at a higher level unless there is reason to suspect its integrity.

T = Test required.

A = Analysis required; tests may be required to substantiate the analysis.

T/A = Test required if analysis indicates possible condensation.

T,A = Test is not required at all levels of assembly if analysis verification is established for non-tested elements.

Note: Card level thermal analysis is required to insure temperature limits, for example, junction temperatures, are not exceeded.

Spare components shall undergo a test program in which the number of thermal cycles is equivalent to the total number of cycles other flight components are subjected to at the component, subsystem, and payload levels of assembly. As a minimum, spare components shall be subjected to eight thermal cycles prior to integration onto payload/spacecraft. Likewise, the durations of the tests at the upper and lower temperatures shall be the same as those for flight components.

Redundant components shall be exercised sufficiently during the test program, including cold and hot starts, to verify proper orbital operations. Testing to validate cross-strapping shall also be performed if applicable. The method of conducting the tests shall be described in the environmental verification test specification and procedures (2.1.1.1.1 and 2.1.1.4).

Consideration should be given to conducting the thermal balance verification test in conjunction with the thermal-vacuum test program. A combined test is often technically and economically advantageous. It must, however, satisfy the requirements of both tests. The approach that is chosen shall be described in the environmental verification specification and procedures.

2.6.2.2 Special Considerations -

a. Unrealistic Failure Modes - Care shall be taken during the test to prevent unrealistic environmental conditions that could induce test failure modes. For instance, maximum rates of temperature changes shall not exceed acceptable limits. The limits are based on hardware characteristics or orbital predictions.

b. Avoiding Contamination - Elements of a test item can be sensitive to contamination arising from test operations or from the test item itself. If the test item contains sensitive elements, the test chamber and all test support equipment shall be examined and certified prior to placement of the item in the chamber to ensure that it is not a significant source of contamination. Particular care shall be taken that potential contaminants emanating from the test item are not masked by contaminants from the chamber or the test equipment. Chamber bakeout and certification may be necessary for contamination sensitive hardware.

   The level of contamination present during thermal vacuum testing should be monitored using, as a minimum, a Temperature-controlled Quartz Crystal Microbalance (TQCM) to measure the accretion rate and a cold finger to obtain a measure of the content and relative amount of the contamination. The use of additional contamination monitors such as a Residual Gas Analyzer (RGA), mirrors, and chamber wipes shall also be considered. When using TQCMs, RGAs, or mirrors, the locations of the sensors must be carefully selected so that they will adequately measure outgassing from the desired source.

   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 shall be conducted at rates sufficiently slow to prevent that from occurring. Testing shall start with a hot soak and end with a hot soak to minimize this risk. However, if it is necessary that the last exposure be a cold one, the test procedure shall include a phase to warm the test item before the chamber is returned to ambient conditions so that the item will remain the warmest in the test chamber, thus decreasing the likelihood of its contamination during thecritical period. In all cases, every effort should be made to keep the test article warmer than its surroundings during testing.

2.6.2.3 Level of Testing - The demonstrations described below apply to component, subsystem/instrument, and payload level tests. If it is impracticable to test an entire integrated payload, the test may be conducted at the highest practicable level of assembly and ancillary testing and analyses shall be conducted to verify the flightworthiness of the integrated payload. In cases where testing is compromised, for example the inability to drive temperatures of the all-up assembly to the qualification limits, testing at lower levels of assembly may be warranted.

2.6.2.4 Test Parameters - The following parameters define key environmental conditions of the test:

a. Thermal Margins - Thermal margins shall be established to provide allowances to compensate for uncertainties in the thermal parameters and to induce stress conditions to detect unsatisfactory performance that would not otherwise be uncovered before flight.

When a thermal balance test precedes the thermal-vacuum test, results from that test shall be used to refine the thermal-vacuum test criteria, presuming that the thermal analysis model has been test verified.

The maximum and minimum temperatures to be imposed during the thermal-vacuum test shall be based on either program requirements or predicted temperatures derived analytically, using a verified model, that each component will undergo during the mission, and shall represent a temperature range, including margins, large enough to induce workmanship stressing during temperature cycling.

A temperature margin of no less than 10°C above the predicted maximum operating conditions and 10°C below the minimum operating conditions (and if appropriate, nonoperating conditions) shall be used in establishing test temperatures. Where the temperature of an area is controlled by a verified active thermal control system, the margin may be reduced to no less than 5°C. Verification may be shown by establishing that the heater will be on no more than 70 percent of the time at the lower operating limit with worst case cold environment (cold environmental fluxes, low biased power, minimum orbit average voltage, cold case thermal properties, etc.) thereby providing a positive heater control margin of 30 percent. This demonstration must be accomplished by test.

Test temperatures for a thermal vacuum soak shall be based on the temperatures at selected locations or average temperature of a group of locations. The locations shall be selected in accordance with an assessment to ensure that components or critical parts of the payload achieve the desired temperature for the required time during the testing cycle. As an example, the temperature sensors shall be attached to the component base plate or to the heat sink on which the component is mounted. Temperature soaks and dwells shall begin when the “control” temperature is within ± 2° C of the proposed test temperature.

b. Temperature Cycling - Cycling between temperature extremes has the purpose of checking performance at other than stabilized conditions and of causing temperature gradient shifts, thus inducing stresses intended to uncover incipient problems. The minimum number of thermal-vacuum temperature cycles for the payload, subsystem/instrument, and component levels of assembly are as follows:

1. Payload/Spacecraft - Four (4) thermal-vacuum temperature cycles shall be performed at the payload level of assembly. If the expected mission temperature excursions are small (less than 10° C) or the transition times are long (greater than 72 hours), the minimum number of thermal-vacuum test cycles may be reduced to two (2) with project approval; however, in these cases the durations for the hot and cold temperature dwells shall be doubled.

2. Subsystem/Instrument - A minimum of four (4) thermal-vacuum temperature cycles shall be performed at the subsystem/instrument level of assembly. During the cycling, the hardware shall be operating and its performance shall be monitored.

3. Component/Unit - All space hardware shall be subjected to a minimum of eight (8) thermal-vacuum temperature cycles before being installed into the payload; these may include test cycles performed at the subsystem/instrument level of assembly. During the cycling, the hardware shall be operating and its performance shall be monitored.

   For components that are determined by analysis to be insensitive to vacuum effects relative to temperature levels and temperature gradients, the requirements may be satisfied by temperature cycling at normal room pressure in an air or gaseous-nitrogen environment. If this approach is used, the cycling at ambient pressure should be increased (both the temperature range and the number of cycles) to account for possible analytical uncertainties and to heighten the probability of detecting workmanship defects. It is recommended that the qualification margin of ± 10°C (in vacuum) be increased to ±15°C if testing at ambient pressure is performed. Likewise, the number of thermal cycles should be increased by fifty (50) percent if testing at ambient pressure (i.e., if 4 cycles would be performed in vacuum, then 6 cycles should be performed at ambient pressure).

   The recommended approach is to test in the expected environment (vacuum). If testing at ambient pressure is implemented, GSFC project approval is required.

c. Duration - The total test duration shall 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. Minimum temperature dwell times are as follows:

1. Payloads/Spacecraft - Payloads shall be exposed for a minimum of twenty-four (24) hours at each extreme of each temperature cycle. The thermal soaks must be of sufficient duration to allow time for performance tests. For small payloads (Scout class), the durations may be shortened, if appropriate, for mission simulation. For large payloads that elect to perform only two (2) thermal-vacuum cycles at the payload level of assembly, the dwell times shall be doubled to a minimum of forty-eight (48) hours.

2. Subsystem/Instrument - Subsystems and instruments shall be exposed for a minimum of twelve (12) hours at each extreme of each temperature cycle. The thermal soaks must be of sufficient duration to allow time for performance tests.

3. Unit/Component - Components shall be exposed for a minimum of four (4) hours at each extreme of each temperature cycle.

d. Functional Test - Because of the length of time involved, it may be impracticable to conduct a comprehensive electrical functional test during thermal-vacuum verification. With project approval, a limited functional test may be substituted if satisfactory performance is demonstrated for the major mission-critical modes of operation. Otherwise, the requirements of 2.3 apply.

e. Pressure - The chamber pressure after the electrical discharge checks are conducted shall be less than 1.33 X 10^{-3} Pa. (1 X 10^{-5} torr).

f. Turn-on Demonstration - Turn-on capability shall be demonstrated under vacuum at least twice at both the low and high temperatures, as applicable. Turn-on temperatures are defined by the expected mission operations; that is, temperatures should be in slight, 2°C, excess of either safe-hold or survival conditions. The ability to function through the voltage breakdown region shall be demonstrated if applicable to mission requirements (all elements that are operational during launch).

2.6.2.5 Test Setup - The setup for the test, including any instrument stimulators, shall be reviewed to ensure that the test objectives will be achieved, and that no test induced problems are introduced. The payload test configurations shall be as described in the test plan and test procedure. The test item shall be, as nearly as practicable, in flight configuration. The components shall be thermally coated and the mounting surface shall have the same treatment as it will have during flight. Critical temperatures shall be monitored throughout the test and "alarmed" if possible. The operational modes of the payload shall be monitored in accordance with 2.3. The provisions of 2.3 apply except when modified by the considerations of 2.6.2.4 d.

2.6.2.6 Demonstration -

a. Electrical Discharge Check - Items that are electrically operational during pressure transitions shall undergo an electrical discharge check 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 (if applicable). The test shall include checks for electrical discharge during the corresponding phases of the vacuum chamber operations.

b. Outgassing Phase - If the test article is contamination sensitive (or if required by the contamination control plan) an outgassing phase must be included to permit a large portion of the volatile contaminants to be removed. The outgassing phase will be incorporated into a hot exposure that will occur during thermal-vacuum testing. The test item will be cycled hot and remain at this temperature until the contamination control monitors indicate that the outgassing has decreased to an acceptable level.

c. Hot and Cold Start Demonstrations - Start-up capability shall be demonstrated to verify that the test item will turn on after exposure to the extreme temperatures that may occur in orbit. For this check, the test item may be in one of three modes: commanded-off, undervoltage-recycle, or high-voltage.

Cold Conditions - The temperature controls shall be adjusted to cause the test item to stabilize at the lower test temperature. Cold turn-on capability shall be demonstrated as required and may be conducted at the start of the cold condition. The duration of the cold phase shall be at least sufficient to permit the performance

The duration of the cold phase shall be at least sufficient to permit the performance of the functional tests with a minimum soak time of four (4) hours for components, twelve (12) hours for subsystems and instruments, and twenty-four (24) hours for payload testing.

e. Transitions - The test item shall remain in an operational mode during the transitions between temperatures so that its functioning can be monitored under a changing environment. The requirement may be suspended when turn-on of the test item is to be demonstrated after a particular transition. In certain cases, it may be possible to remove thermal insulation to expedite cool-down rates. Caution must be taken not to violate temperature limits, or to induce test failures caused by excessive gradients.

f. Hot Conditions - The temperature controls shall be adjusted to cause the test item to stabilize at the upper test temperature. Hot turn-on capability is demonstrated as required. The duration of this phase shall be at least sufficient to permit the performance of the functional tests with a minimum soak time of four (4) hours for components, twelve (12) hours for subsystems and instruments, and twenty-four (24) hours for payload testing.

g. Return to Ambient - If the mission includes a requirement for the test item to remain in an operational mode through the descent and landing phases, the test shall include a segment to verify that capability. If possible, the test article should be kept warmer than the surroundings to protect against contamination from the test facility.

2.6.2.7 Special Tests - Special tests may be required to evaluate unique features, such as a radiation cooler, or to demonstrate the performance of external devices such as solar array hinges or experiment booms that are deployed after the payload has attained orbit.

The test configuration shall reflect, as nearly as practicable, the configuration expected in flight.

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, the facility, or personnel.

Any special tests shall be included in the environmental verification specification (1.10.2).

2.6.2.8 Trouble-Free-Performance - At least 100 trouble-free hours of functional operations at the hot conditions, and 100 trouble-free hours of functional operations at the cold conditions must be demonstrated in the thermal verification program (refer to section 2.3.4).

2.6.2.9 Acceptance Requirements - The above provisions apply for the acceptance of previously qualified hardware except that the 10°C margin may be waived.

2.6.3 Thermal Balance Qualification

The adequacy of the thermal design and the capability of the thermal control system shall be verified. It is preferable that the thermal balance test precede the thermal vacuum test so that the results of the balance test can be used to establish the temperature goals for the thermal vacuum test.

2.6.3.1 Alternative Methods - It is preferable to conduct a thermal balance test on the fully assembled payload. If that is impracticable, one of the following alternative methods may be used:

a. Test at lower levels of assembly, and compare the results with the predictions derived from the modified analytical model.

b. Test a thermally similar physical representation of the flight payload (e.g. a physical thermal model) and compare the results with predictions derived from the analytical model (modified as necessary).

If the flight equipment is not used in the tests, additional tests to verify critical thermal properties, such as thermal control coating absorptivity and emissivity, shall be conducted to demonstrate similarity between the item tested and the flight hardware.

2.6.3.2 Use of a Thermal Analytical Model - In the course of a payload program, analytical thermal models are developed of the payload, its elements, and the mission environment for the purpose of predicting the thermal performance during the mission. The models can be modified to predict the thermal performance in a known test-chamber environment. Correlation of the results of the chamber thermal balance tests with predictions derived from the modified analytical model provides a means for validating the thermal design and for improving model accuracy. Predictions derived from the modified analytical model must be based on the actual test conditions. At the same time, a thermal balance test can provide the basis for evaluating the performance of the as-built thermal control system.

2.6.3.3 Method of Thermal Simulation - A decision must be made as to the method used to simulate thermal inputs. The type of simulation to be used is generally determined by the size of the chamber, the methods available to simulate environmental conditions, and the payload. In planning the method to be used, the project test engineer should try to achieve the highest practical order of simulation; that is, the one that requires the minimum number of assumptions and calculations to determine the environmental inputs. The closer the simulation to the worst case environments, the less reliance on the thermal analytical model to verify the adequacy of the thermal design. Methods of simulation and the major assumptions for a successful test are described below:

a. Solar inputs can be simulated by mercury-xenon, xenon, or carbonarc sources and/or heaters as described below. The spectrum and uniformity of the source used to simulate the sun and planet albedo must be understood. The change in effective absorptivity caused by spectrum mismatches can be quite large; the emissivity change is quite small.

b. Planetary, or earth emissions, can be simulated with either:

(1) Skin Heaters - This is an acceptable test for simply shaped payloads. The absorbed energy from all exterior sources are simulated by the skin using I^2R heaters. The absorptivity and incident radiation are used to calculate the absorbed energy to be simulated,

(2) Heater Plates - This can be an acceptable test if the payload outer skins are not touched. The same information is needed for the plates as for the skin heaters and the exchange factor between the plates and the payload must be known. In both cases, a net balance equation considering absorptivity, emissivity, incident and rejected energies must be solved to establish accurate test conditions.

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

2.6.3.4 Extraneous Effects - 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 shall also be taken to prevent conditions, such as test configuration-induced contamination, that cause an unrealistic degradation of the test item. Devices such as a TQCM, cold finger, RGA, mirrors, witness samples, and chamber wipes shall be included as necessary to monitor contamination.

2.6.3.5 Demonstration - The number of energy balance conditions simulated during the test shall be sufficient to verify the thermal design. (If it is necessary to verify the thermal model, a minimum of two tests are required.) The duration of the thermal balance test depends on the mission, payload design, payload operating modes, and times to reach stabilization; stabilization is generally considered to have been achieved when the control sensors change less than 0.05°C per hour, for a period of not less than six hours, and exhibit a decreasing temperature slope over that period. Alternatively, a stabilization criteria may be used where the amount of energy represented by the time rate of temperature change (and the thermal mass of the test article) is a small fraction (typically 2 to 5%) of the total energy of the test article. The exposures shall be long enough for the payload to reach stabilization so that temperature distributions in the steady-state conditions may be verified. The conditions defining temperature stabilization shall be described in the environmental verification specification.

The amount of differences allowed between predicted and measured temperatures are determined by the cognizant thermal analyst. Verification of the thermal analytical model is considered accomplished if the established criteria are met.

2.6.3.6 Acceptance Requirements - The thermal balance test may be waived, but either tests shall be conducted to verify the thermal similarity to the previously qualified hardware or sufficient temperature margins exist to preclude reverification.

2.6.4 Temperature-Humidity Verification: Manned Spaces

If the environment is such that condensation can occur, tests shall be conducted to demonstrate that the hardware can function under the severest conditions that credibly can be expected.

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

2.6.4.2 Demonstration - The hardware shall 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; in that event, tests shall be conducted to demonstrate that the hardware can function properly after (or, if applicable, during) such exposure.

Temperature cycling, duration, performance tests, and other requirements (except those related to vacuum) as described for the thermal-vacuum test (2.6.2) shall apply.

2.6.5 Temperature-Humidity Verification: Descent and Landing

Hardware that is to undergo the temperature and humidity environment of the unpressurized cargo bay and that must return from orbit with a specified performance capability (e.g. throughput or reflectivity) shall be subjected to a temperature-humidity test to verify that it can survive the environmental conditions during descent and landing without experiencing unacceptable degradation.

2.6.5.1 Special Considerations - If the test would make the hardware unflightworthy, such as by rendering thermal control surfaces ineffective, then it should not be performed on the flight item. Instead, an analysis based on tests of engineering or prototype models, or other convincing methods, may be used.

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

The temperature and humidity profiles in Figure 2.6-1 set the parameters for the demonstration. The payload shall be in a configuration appropriate for the descent and landing phase.

Electrical function tests (2.3) shall be conducted after the test exposure to determine whether acceptable limits of degradation have been exceeded.

2.6.5.3 Acceptance Requirements - The above provisions apply for the acceptance of previously qualified hardware.

2.6.6 Temperature-Humidity: Transportation and Storage

Hardware that will not be maintained in a temperature-humidity environment that is controlled within acceptable limits during transportation and storage shall be subjected to a temperature-humidity test to verify satisfactory performance after (and, if applicable, during) exposure to that environment.

2.6.6.1 Applicability - The test applies to all payload equipment. It need not be conducted on equipment for which the demonstrated acceptable limits have been established during other portions of the verification program.

2.6.6.2 Demonstration - The demonstration shall be performed prior to the thermal-vacuum test. An analysis shall be made to establish the uncontrolled temperature and humidity limits to which the item will be exposed from the time of its integration at the component level through launch. The item shall be placed in a temperature-humidity chamber and electrical function tests (2.3) shall be conducted before the item is exposed to the test environment.

If an electrical function test was conducted during the post-test checkout of the preceding test, the results of that may suffice. Functional tests shall also be conducted during the test exposure if the item will be required to operate during the periods of uncontrolled environment.

The test shall include exposure of the hardware to the extremes of temperatures and humidities as follows: 10°C and 10 RH (but not greater than 95% RH) higher and lower than those predicted for the transportation and storage environments. The test item shall be exposed to each extreme for a period of six (6) hours.

Electrical function tests shall be conducted after the test exposure to demonstrate acceptable performance.

2.6.6.3 Acceptance Requirements - The above provisions apply to previously qualified hardware except that the 10°C and 10 RH margins may be waived.

2.6.7 Leakage (Integrity Verification)

Tests shall be conducted on sealed items to determine whether leakage exceeds the rate prescribed for the mission.

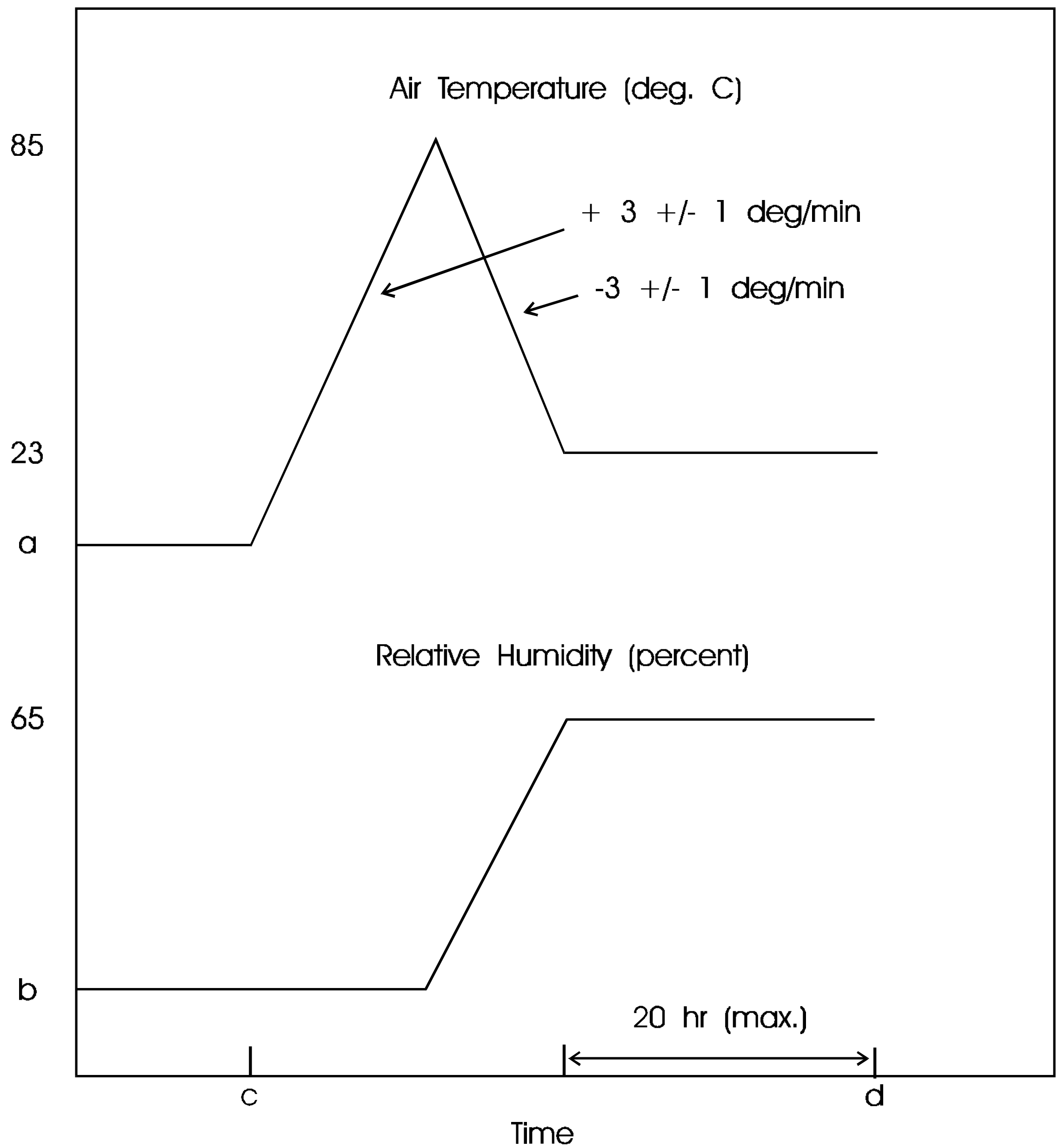
2.6.7.1 Levels of Assembly - Tests may be conducted on the component level of assembly to gain assurance that the item will function satisfactorily before tests are made at higher levels. Checks at the payload level need include only those items that have not demonstrated satisfactory performance at the lower level, are not fully assembled until the higher levels of integration, or the integrity of which is suspect.

2.6.7.2 Demonstration - Leakage rates are checked before and after stress-inducing portions of the verification program. The final check may be conducted during the final thermal-vacuum test.

A mass spectrometer may be used to detect flow out of or into a sealed item.

If dynamic seals are used, the item shall be operated during the test, otherwise operation is not required. The test should be conducted under steady-state conditions, i.e., stable pumping, pressures, temperatures, etc. If time constraints do not permit the imposition of such conditions, a special test method shall be devised.

2.6.7.3 Acceptance Requirements - The above provisions apply to the acceptance testing of previously qualified hardware.



Legend:

a.	=	Temperature of payload at deorbit
b.	=	Minimum chamber relative humidity
c.	=	Payload temperature stabilized
d.	=	Functional check-out

Figure 2.6-1 Temperature-Humidity Profile for Descent and Landing Demonstration

SECTION 2.7

CONTAMINATION CONTROL

2.7 CONTAMINATION CONTROL PROGRAM

The objective of the contamination control program is to decrease the likelihood that the performance of payloads will be unacceptably degraded by contaminants. Since contamination control programs are dependent on the specific mission goals, instrument designs, planned operating scenarios, etc. it is necessary for each program to provide an allowable contamination budget and a Contamination Control Plan (CCP) which defines the complete contamination control program to be implemented for the mission. The specific verification plans and requirements must be defined in the CCP. The procedures that follow provide an organized approach to the attainment of the objective so that the allowable contamination limit is not violated.

2.7.1 Applicability

The contamination control program is applicable to all payloads, subsystems, instruments, and components during all mission phases (fabrication, assembly, integration, testing, transport, storage, launch site, launch, and on-orbit). In the cases of payloads which are not sensitive to contamination, this program may still be required due to cross-contamination potentials to other payloads and/or orbiter systems.

2.7.2 Summary of Verification Process

The following are performed in order:

a. Determination of contamination sensitivity;
b. Determination of a contamination allowance;
c. Determination of a contamination budget;
d. Development and implementation of a contamination control plan.

Each of the above activities shall be documented and submitted to the project manager for concurrence/approval.

2.7.3 Contamination Sensitivity

An assessment shall be made early in the program to determine whether the possibility exists that the item will be unacceptably degraded by molecular or particulate contaminants, or is a source of contaminants. The assessment shall take into account all the various factors during the entire development program and flight including identification of materials (including quantity and location), manufacturing processes, integration, test, packing and packaging, transportation, and mission operations including launch and return to earth, if applicable. In addition, the assessment should identify the types of substances that may contaminate and cause unacceptable degradation of the test item.

If the assessment indicates a likelihood that contamination will degrade performance, a contamination control program should be instituted. The degree of effort applied shall be in accordance with the importance of the item's function to mission success, its sensitivity to contamination, and the likelihood of its being contaminated.

2.7.4 Contamination Allowance

The amount of degradation of science performance that is allowed for critical, contamination-sensitive items shall be established, usually by the Project Scientist. From this limit, the amount of contamination that can be tolerated, the contamination allowance, will be established. The rationale for such determination and the ways in which contaminants will cause degradation shall be described in the contamination control plan (2.7.7) The allowable degradation should also be included in a contamination budget.

2.7.5 Contamination Budget

A contamination budget shall be developed for each critical item. It shall describe the quantity of contaminant and the degradation that may be expected during the various phases in the lifetime of the item. The phases shall include the mission itself. The budget should be stated in terms (or units) that can be measured during testing. The measure of contamination shall be monitored as the program progresses to include the contamination and degradation experienced. The budget shall be monitored to ensure that, given the actual contamination, the mission performance will remain acceptable. In the event that contamination build-up predictions are not borne out, corrective action shall be taken.

A contamination-sensitive item may be cleaned periodically to reestablish a budget baseline. Contamination avoidance methods, such as cleanrooms and instrument covers, will affect the budget and a general description of their usage should be included. The contamination budget shall be negotiated among the cognizant parties (e.g., the Project Scientist, the instrument contractor and the payload integration contractor). Each contractor shall be responsible for staying within their portion of the budget; however, the budget may be redistributed, with the concurrence of the project manager, in order to improve the approach.

2.7.6 Contamination Control Plan

A contamination control plan shall be prepared that describes the procedures that will be followed to control contamination. It shall establish the implementation and describe the methods that will be used to measure and maintain the levels of cleanliness required during each of the various phases of the item's lifetime. The plan shall specifically address outgassing requirements for the flight items, test chamber, and test support equipment.

2.7.7 Other Considerations

The effects of the payload on other payloads in the orbiter cargo bay shall also be considered and addressed in the Contamination Plan. The formation or transfer of payload effluents that could jeopardize the performance of orbiter systems (e.g., radiators, windows, optics, etc.) or other payloads manifested on the same flight shall be restricted. All non-metallic materials shall be selected for low outgassing characteristics. Material selection criteria shall be consistent with those stipulated in JSC 07700 Vol. XIV. and NASA Reference Publication 1124.

Bake-outs of solar arrays, major wiring harnesses, and thermal blankets are required unless it can be satisfactorily demonstrated to the GSFC project that the contamination allowance can be met without bake-outs. Bake-outs of other components with large amounts of non-metallic material, such as batteries, electronic boxes, and painted surfaces may also be necessary.

Because they can be a source of contamination themselves, special consideration shall also be given to materials and equipment used in cleaning, handling, packaging, and purging flight hardware.

Contamination

The contamination program requirements be followed closely during the environmental test program. Non-flight materials near the flight hardware may damage or contaminate it. For example:

- Non-flight GSE wiring and connector materials can contaminate the flight hardware during thermal testing.
- Packaging material (plastic films and flexible foams) can contaminate hardware or cause corrosion during shipping and storage.
- Plastic bags without anti-static properties can allow electrostatic discharges to damage electronic components on circuit boards.
- Tygon tubing (or other non-flight tubing) used in purge systems can contaminate hardware when gasses or liquids extract plasticizers from the tubing.
- Paints, sealants, and cleaning materials used to maintain clean rooms can contaminate or corrode flight hardware.

To protect flight hardware, non-flight hardware that will be exposed to thermal vacuum testing with flight hardware (items such as cables, electronics, fixtures, etc.) should be fabricated from flight quality materials. Packaging materials should be tested to verify that they are non-corrosive, non-contaminating, and provide electrostatic protection, if required. All materials used in purge systems should be tested for cleanliness and compatibility with flight materials.

SECTION 2.8

END-TO-END TESTING

2.8 END-TO-END COMPATIBILITY TESTS AND SIMULATIONS

2.8.1 Compatibility Tests

The end-to-end compatibility test encompasses the entire chain of payload operations that will occur during all mission modes in such manner as to ensure that the system will fulfill mission requirements. The mission environment shall be simulated as realistically as possible and the instruments shall receive stimuli of the kind they will receive during the mission. The RF links, ground station operations, and software functions shall be fully exercised. When acceptable simulation facilities are available for portions of the operational systems, they may be used for the test instead of the actual system elements.

Network Directorate simulation facilities are a constrained resource and their use by the payload project must be negotiated.

The specific environments under which the end-to-end test is conducted and the stimuli, payload configuration, RF links, and other system elements to be used must be determined in accordance with the characteristics of the mission.

2.8.2 Mission Simulations

After compatibility between the network and the user facility have been demonstrated, data flow tests shall be performed that exercise as much of the total system as possible. Once the data flow paths have been verified, mission simulations are enacted to validate nominal and contingency mission operating procedures and to provide for operator training. To provide ample time for checkout of the project operating control center (POCC), it is essential that users take part in mission simulations from the early stages.

Mission simulations are the combined responsibility of the mission operations manager and the network support managers, and shall involve all participating elements and operating personnel (from project and support elements).

Information describing the network data simulation equipment capabilities can be found in PSS and SOC Guide for TDRSS and GSTDN Users, STDN No. 101.6 (see 1.7.5). Information describing DSN is contained in the Deep Space Network/Flight Project Interface Design Handbook (1.7.6).

APPENDIX A

GENERAL INFORMATION

Acoustic Fill effects

The acoustic sound pressure level in the area between the payload and the payload fairing, or orbiter side walls, increases as the gap decreases. Thus for large payloads, a fill factor is often used to adjust for this effect.

The Fill Factor recommended by the NASA Vibroacoustics Standards Panel is given by:

$$\text{Fill Factor (dB)} = 10 \text{ Log} \left\{ \frac{\left(1 + \dfrac{C_a}{2fH_{gap}}\right)}{1 + \dfrac{C_a}{2fH_{gap}}\left(1 - Vol_{ratio}\right)} \right\}$$

where: C_a is the speed of sound in air (typically 344.4 meters/second, 1130 ft/sec, or 13,560 in/sec)

f is the one-third octave band center frequency (Hz),

H_{gap} is the average distance between the payload and the fairing, or cargo bay, wall, and

Vol_{ratio} is the ratio between the payload volume and the empty fairing, or cargo bay, volume for the payload zone of interest.

This fill-factor is added to the empty fairing/cargo bay expected or test levels. However, engineering judgment must be used in the application of this fill-factor for irregular shaped payloads. Also, Many acoustic specifications are now provided with some fill-factor included.

As an example, assume a cylindrical payload section of radius R_S in a fairing of radius R_F shown in Figure A-1.

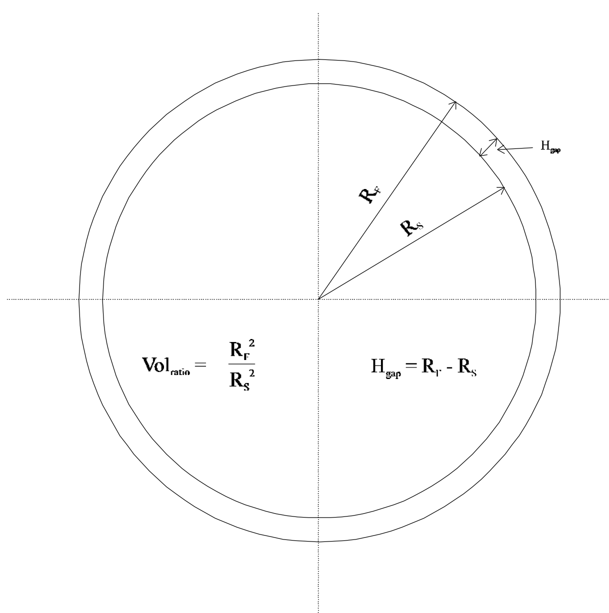


Figure A-1 Cylindrical Payload in Fairing Acoustic Fill-Factor

The fill-factor to be added to the empty fairing acoustic levels for various size payloads, assuming a fairing diameter of 3.0 meters, is given in Table A-1, and is shown in Figure A-2.

Table A-1
Acoustic Fill-Factor (dB)
3 meter Payload Fairing

1/3 Octave Band Center Freq. (Hz)	Payload Diameter (meters)/Volume Fill Ratio (%)				
	2.85/90.3	2.75/84.0	2.50/69.4	2.25/56.3	2.0044.4
25	9.7	7.6	4.8	3.3	2.3
32	9.6	7.5	4.7	3.2	2.3
40	9.5	7.4	4.6	3.2	2.2
50	9.3	7.2	4.5	3.1	2.1
63	9.2	7.1	4.4	3.0	2.0
80	8.9	6.9	4.2	2.8	1.9
100	8.7	6.6	4.0	2.7	1.8
125	8.4	6.4	3.8	2.5	1.7
160	8.1	6.1	3.6	2.3	1.6
200	7.7	5.7	3.4	2.2	1.4
250	7.3	5.4	3.1	2.0	1.3
315	6.9	5.0	2.8	1.8	1.1
400	6.4	4.6	2.5	1.6	1.0
500	5.9	4.2	2.2	1.4	0.9
630	5.3	3.7	2.0	1.2	0.7
800	4.8	3.3	1.7	1.0	0.6
1000	4.3	2.9	1.4	0.8	0.5
1250	3.8	2.5	1.2	0.7	0.4
1600	.0.	2.1	1.0	0.6	0.4
2000	2.9	1.8	0.9	0.5	0.3
2500	2.5	1.5	0.7	0.4	0.2
3150	2.1	1.3	0.6	0.3	0.2
4000	1.7	1.1	0.5	0.3	0.2
5000	1.5	0.9	0.4	0.2	0.1
6300	1.2	0.7	0.3	0.2	0.1
8000	1.0	0.6	0.2	0.1	0.1
10000	0.8	0.5	0.2	0.1	0.1

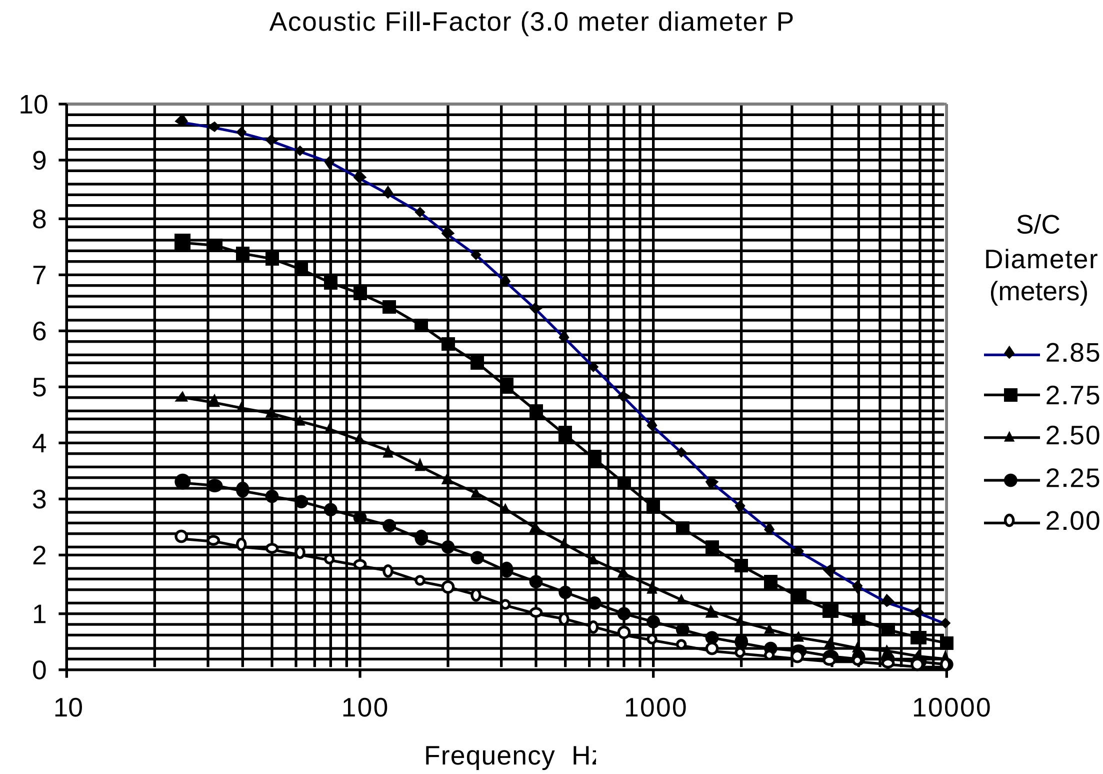


Figure A-2 Acoustic Fill-Factor for various size Payloads in a 3 meter Diameter Payload Fairing

Component Random Vibration

Component random vibration testing is one of the primary workmanship tests to uncover flaws or defects in materials and production. To the greatest extent possible, test levels should be based on knowledge of the expected environment from previous missions or tests. However, it is important to test with sufficient amplitude to uncover the defects. Therefore, as a rule, the input levels should always be greater than or equal to workmanship test levels for electronic, electrical, or electro-mechanical components. If the hardware contains delicate optics, detectors, sensors, etc., that could be damaged by the levels of the workmanship test in certain frequency bands, the test levels may, with project concurrence, be reduced in those frequency regions. A force-limiting control strategy is recommended. The control method shall be described in the Verification Test Procedure and approved by the GSFC project.

The qualification (prototype or protoflight) test level is generally 3 dB greater than the maximum expected (acceptance) test level. That is not always the case however. If the expected level is less than the workmanship level an envelope of the two is used to determine the acceptance test level. The qualification level is also an envelope of the maximum expected + 3 dB and the workmanship level. Under this condition, the qualification envelope may not exceed the acceptance level by 3 dB. Figure A-3 demonstrates this.

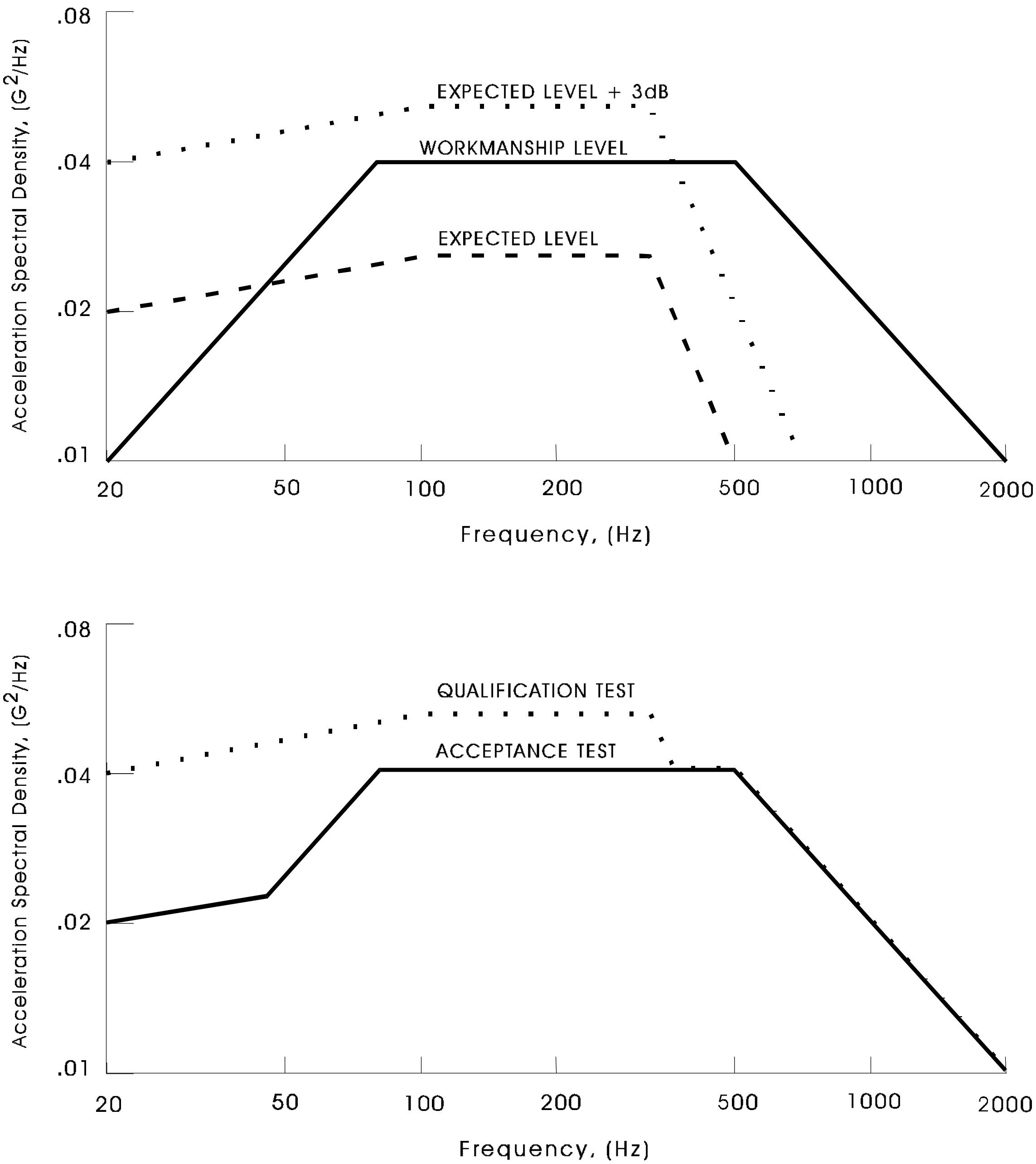


Figure A-3 Determination of Qualification and Acceptance Random Verification Test Levels

Mechanical Shock

In the following appendices, shock spectrum envelopes are provided for various launch-vehicle-induced events. The maximum shock producing event for payloads is generally the actuation of separation devices. The expected shock environment should be assessed for the device to be used, and a spacecraft separation test shall be performed if pyrotechnic devices are to be used for the separation.

A pyrotechnic shock environment is characterized as a high intensity, high frequency, and very short duration acceleration time history that resembles a summation of decaying sinusoids with very rapid rise times. In addition, it is characterized most realistically as a traveling wave response phenomenon rather than as a classical standing wave response of vibration modes. Typically, at or very near the source, the acceleration time history can have levels in the thousands of g's, have a primary frequency content from 1 kHz to 10 kHz, and decay within 3-15 milliseconds. When assessing the source pyro shock environment descriptor as given in the GEVS, the following three factors must be considered:

a. Because of the very complex waveform and very short duration of the time history, there is no accepted way for giving a unique, "explicit" description of the environment for test specification purposes. The accepted standard non-unique, "implicit" description is a "damage potential" measure produced by computing the Shock Response Spectrum (SRS) of the actual environment time history. A SRS is defined as the maximum absolute acceleration response, to the environment time history, of a series of damped, single-degree-of-freedom oscillators that have a specified range of resonant frequencies and a constant value of viscous damping (e.g., Q=10). This type of descriptor is contained in the GEVS. The resulting fundamental objective of the verification test is to create a test environment forcing time history that has nearly the same SRS as the test specification and thereby give some assurance that the test environment has approximately the same "damage potential" as the actual environment.

b. Because of the high frequency, traveling wave response like nature of the subject environment, the acceleration level will be rapidly attenuated as a function of distance from the source and as the response wave traverses discontinuities produced by joints and interfaces.

c. Because of the high frequency, short duration nature of the pyro-shock environment, "potential for damage" is essentially restricted to portions of the payload, or instrument that, for example, have very high frequency resonances (i.e., electrical/electronic elements such as relays, circuit boards, computer memory, etc.) and have high frequency sensitive electromechanical elements such as gyros, etc.

An Aerospace Systems Pyrotechnic Shock Data study was performed by the Denver Division of Martin Marietta for GSFC; The following information, extracted from the 1970 final report of this study, is provided to aid in assessing expected shock levels. The results are empirical and based on a limited amount of data, but provide insight into the characteristics of the shock response spectrum (SRS) produced by various sources, and the attenuation of the shock through various structural elements.

The study evaluated the shock produced by four general types of pyrotechnic devices

- Linear charges (MDF and FLSC);
- Separation nuts and explosive bolts;
- Pin-puller and pin-pushers;
- bolt-cutters, pin-cutters and cable-cutters.

Empirically derived expected SRS's for these four categories are given in Tables A-4 through A-7. It was found that the low-frequency region could be represented, or enveloped, by a constant velocity curve. All shock response curves are for a Q=10.

The attenuation, as a function of frequency and distance was evaluated for the following general types of structure:

- Cylindrical shell;
- Longeron or stringer of skin/ring- frame structure;
- Ring frame of skin/ring- frame structure;
- Primary truss member;
- Complex airframe;
- Complex equipment mounting structure;
- Honeycomb structure.

It was found that the attenuation of the Shock, as a function of distance from the source, could be separated into two parts; the attenuation of the low-frequency constant velocity curve, and the attenuation of the high-frequency peak levels. The attenuation of the constant velocity curve was roughly the same for all types of structure; whereas the attenuation of the higher frequency peak shock response was different for the various categories of structure. Figure A-8 gives the attenuation of the constant velocity portion of the SRS as a function of distance, and Figure A-9 gives the attenuation of the peak SRS level as a function of distance for the various general categories of structure. It must be emphasized that this information was derived empirically from a limited set of shock data.

As an example of the use of these attenuation curves, assume that the source spectrum is that for an explosive bolt given in Figure A-5, and that an estimate of the shock levels 80 inches from the source is being evaluated for complex equipment mounting structure. From Figure A-8, the constant velocity, low-frequency envelope will be attenuated to approximately 20% of the original level. From Figure A-9, the peak level will be attenuated to approximately 7.8% of the original level. The assumed source spectrum and new estimate of the SRS envelope is shown in Figure A-10.

Structural interfaces can attenuate a shock pulse; guideline levels of reduction are as follows:

Interface	Percent Reduction
Solid Joint	0
Riveted butt joint	0
Matched angle joint	30-60
Solid joint with layer of different material in joint	0-30

the attenuation due to joints and interfaces is assumed for the first three joints.

A reduction of shock levels can also be expected from intervening structure in a shell type structure. An example is shown in Figure A-11.

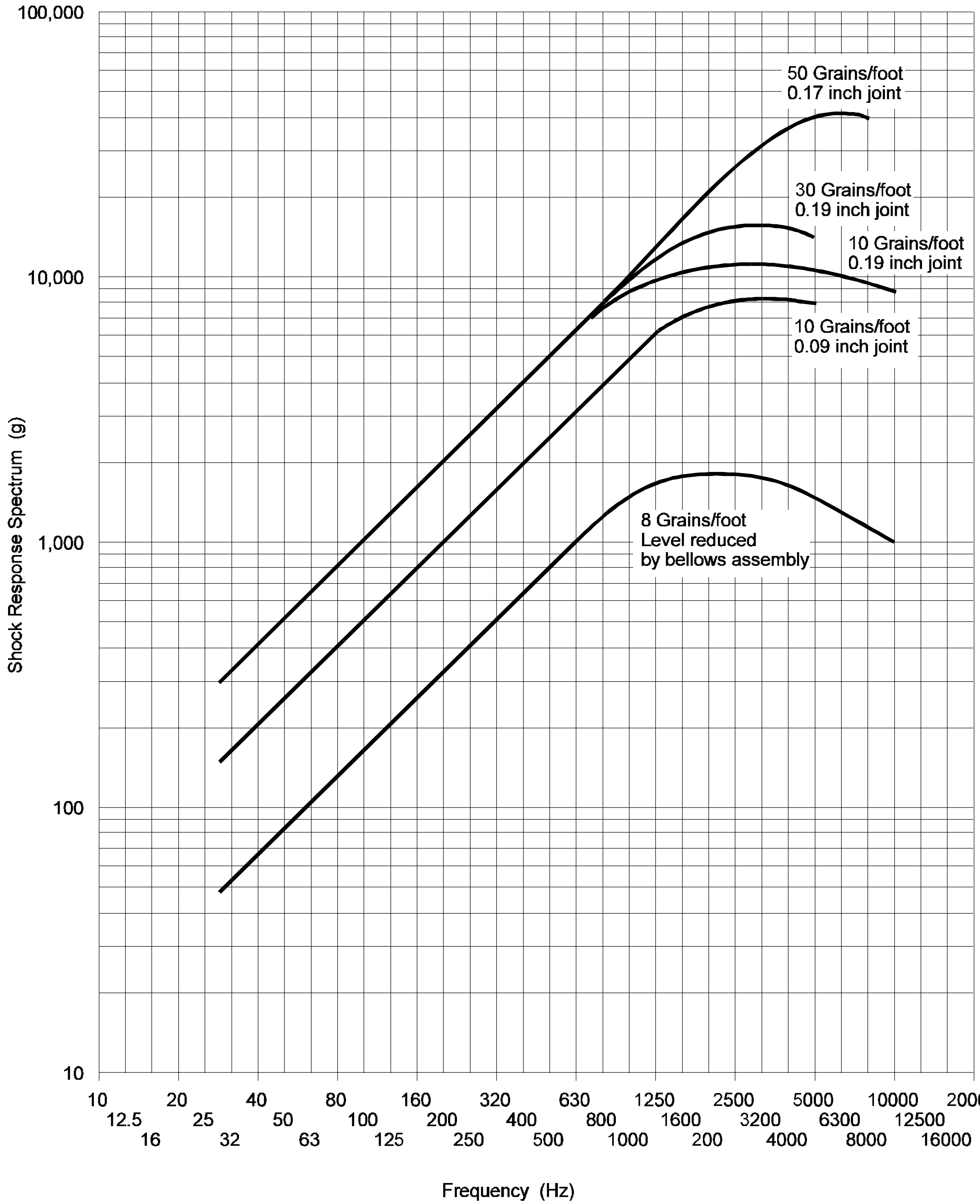


Figure A-4 Shock Environment Produced by Linear Pyrotechnic Devices

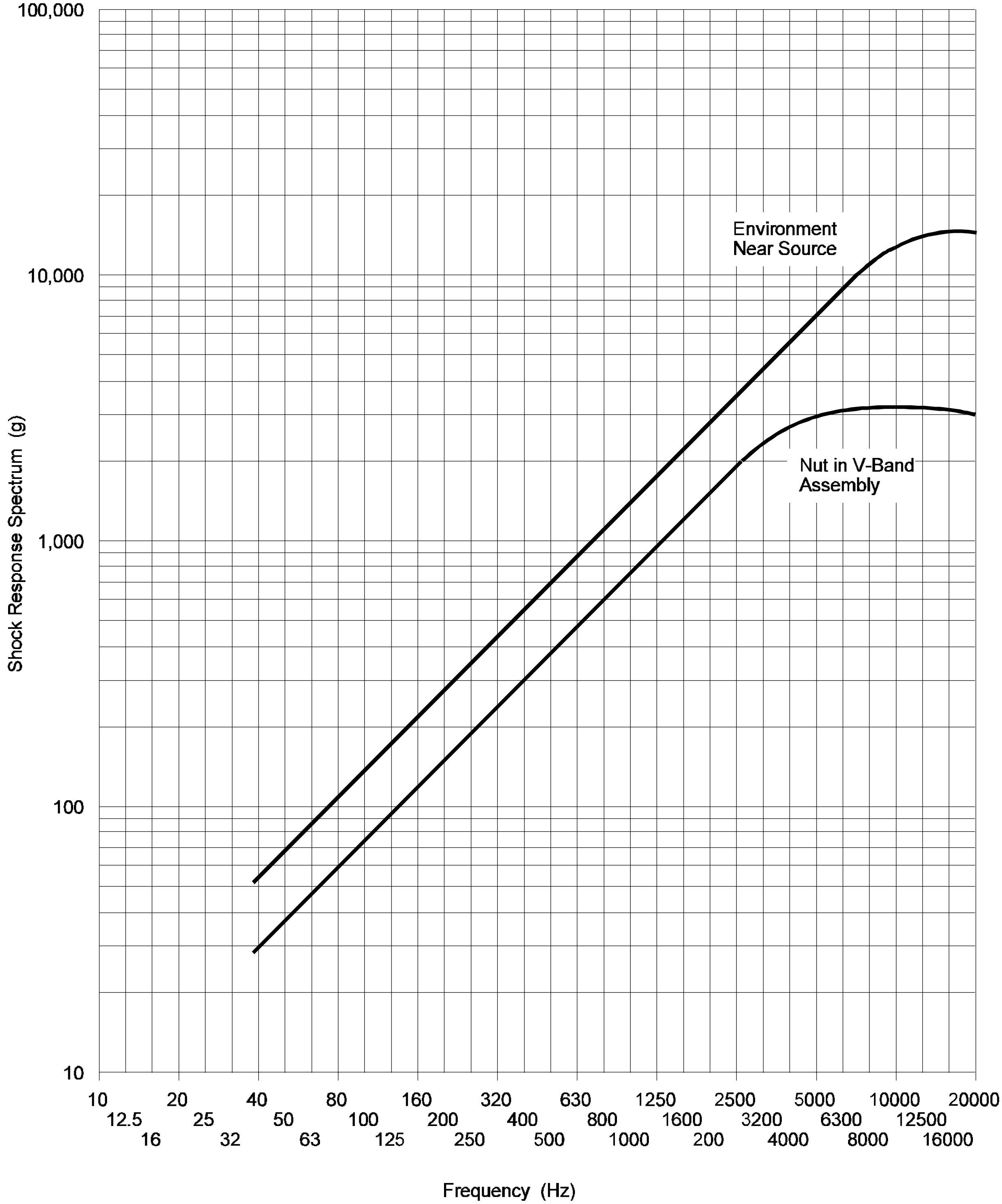


Figure A-5 Shock Environment Produced by Separation Nuts and Explosive Bolts

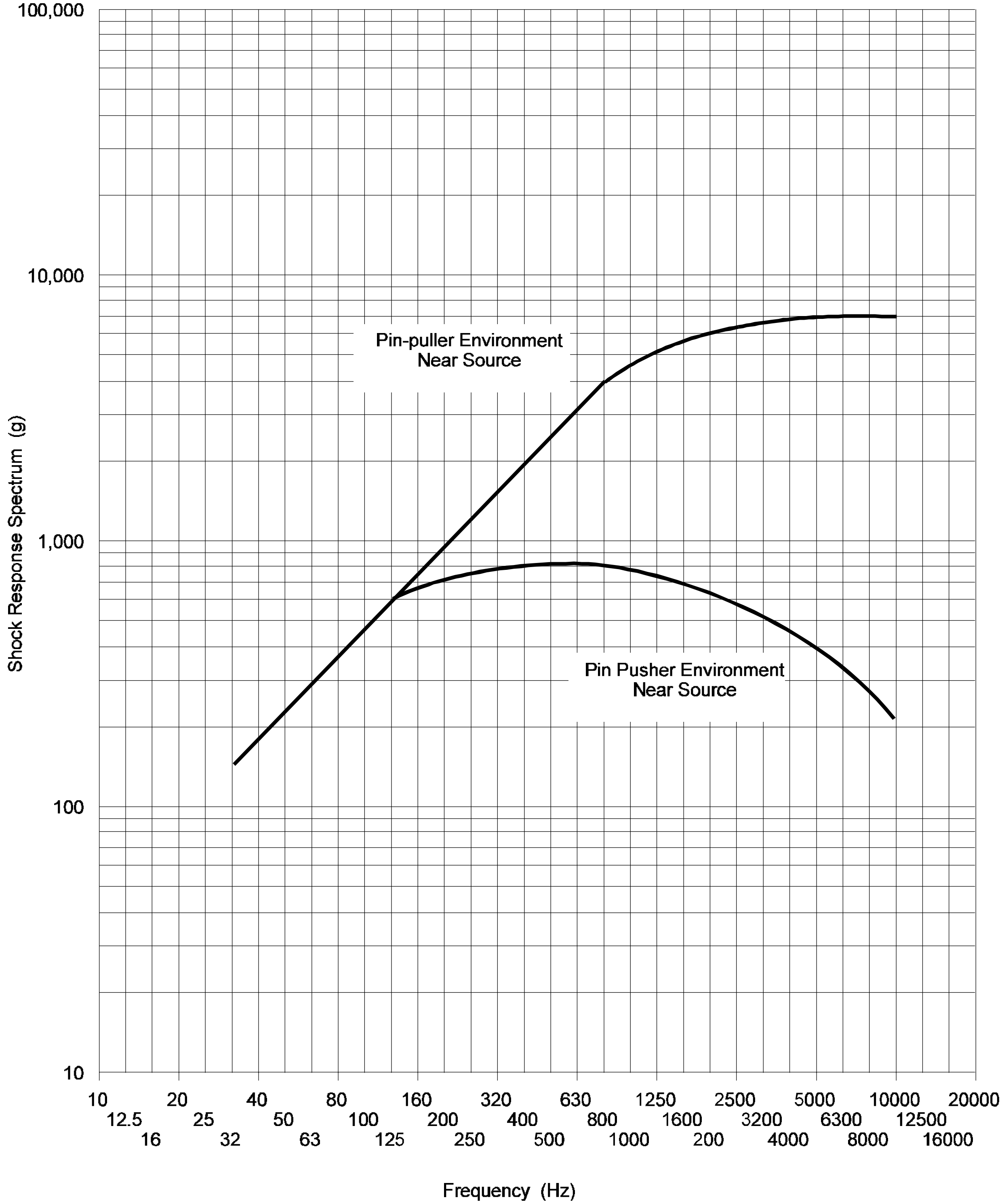


Figure A-6 Shock Environment Produced by Pin-Pullers and Pin-Pushers

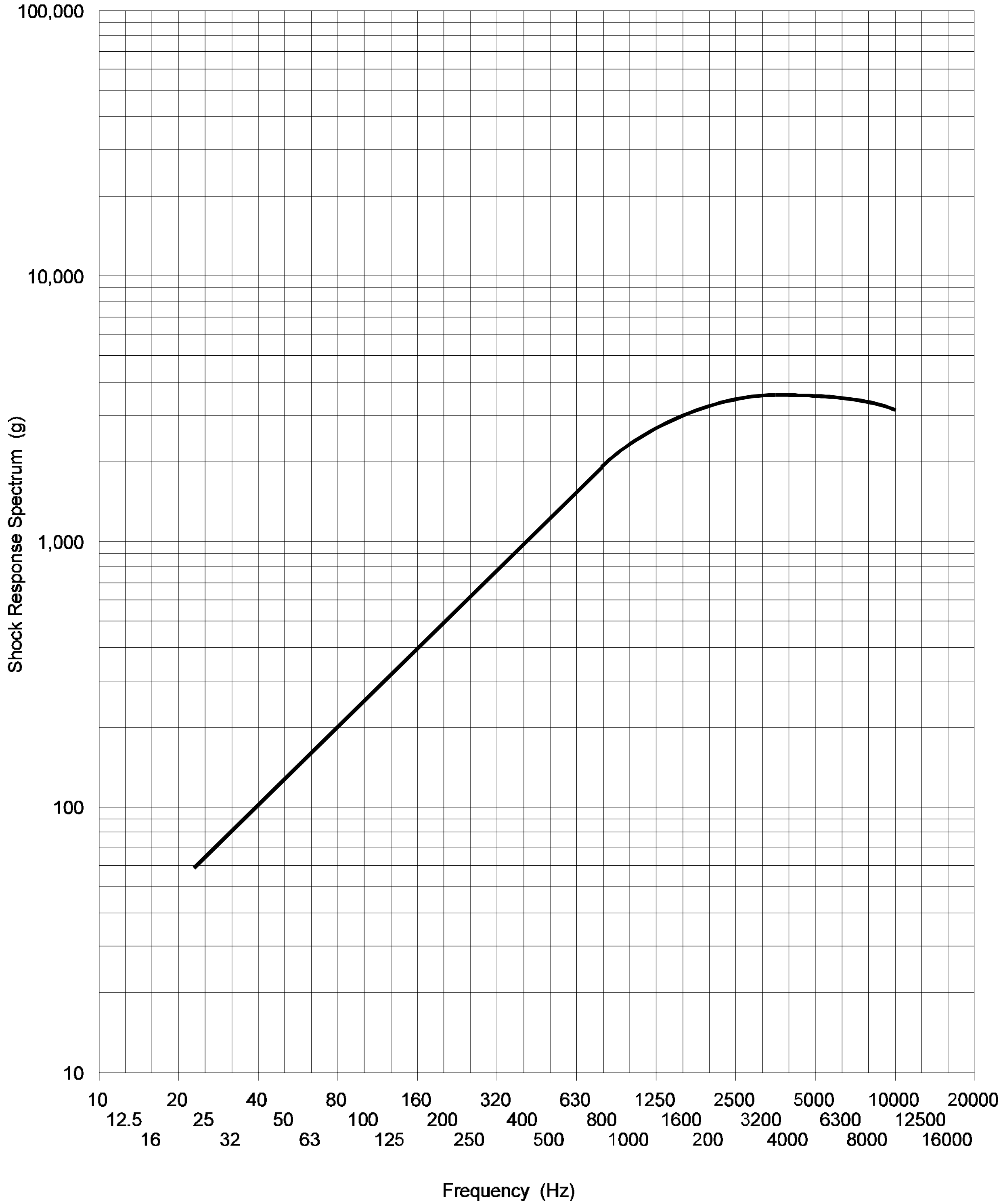


Figure A-7 Shock Environment Produced by Bolt-Cutters, Pin-Cutters, and Cable-Cutters

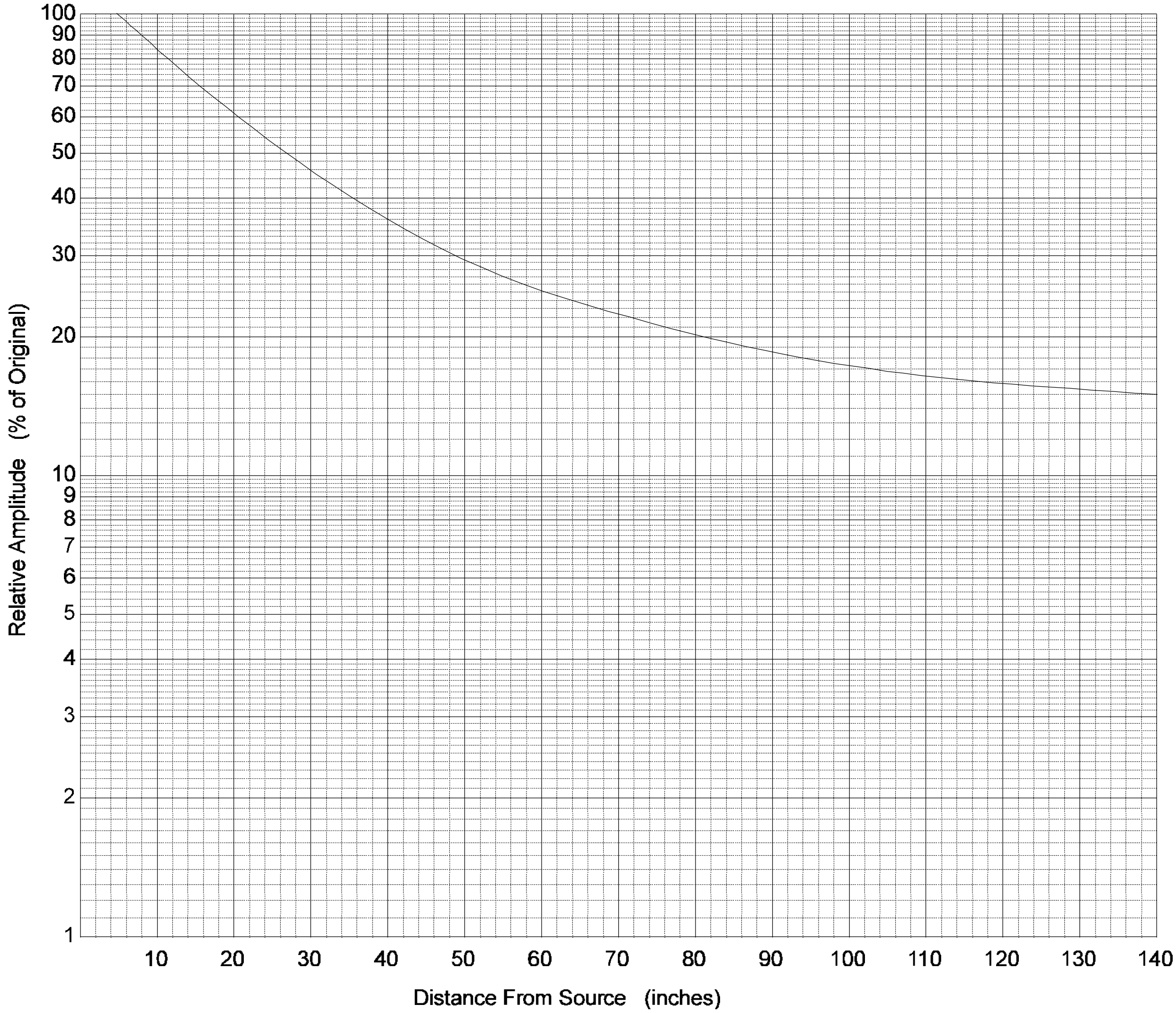
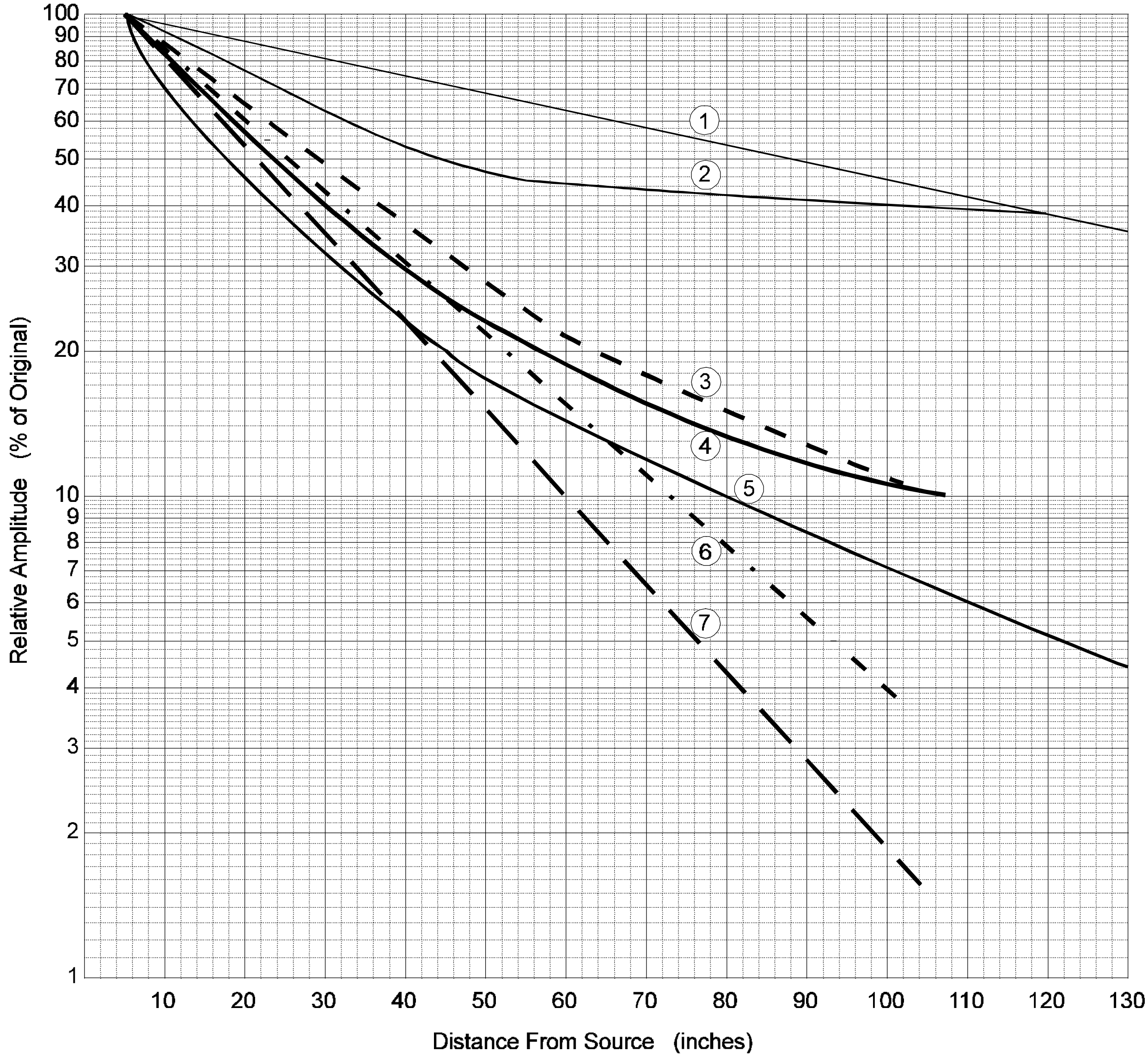


Figure A-8 Attenuation of Constant Velocity Line



① Honeycomb structure
② Longeron or stringer of skin/ring-frame structure
③ Primary truss members
④ Cylindrical shell
⑤ Ring frame of skin/ring-frame structure
⑥ Complex equipment mounting structure
⑦ Complex airframe

Figure A-9 Peak Pyrotechnic Shock Response vs Distance

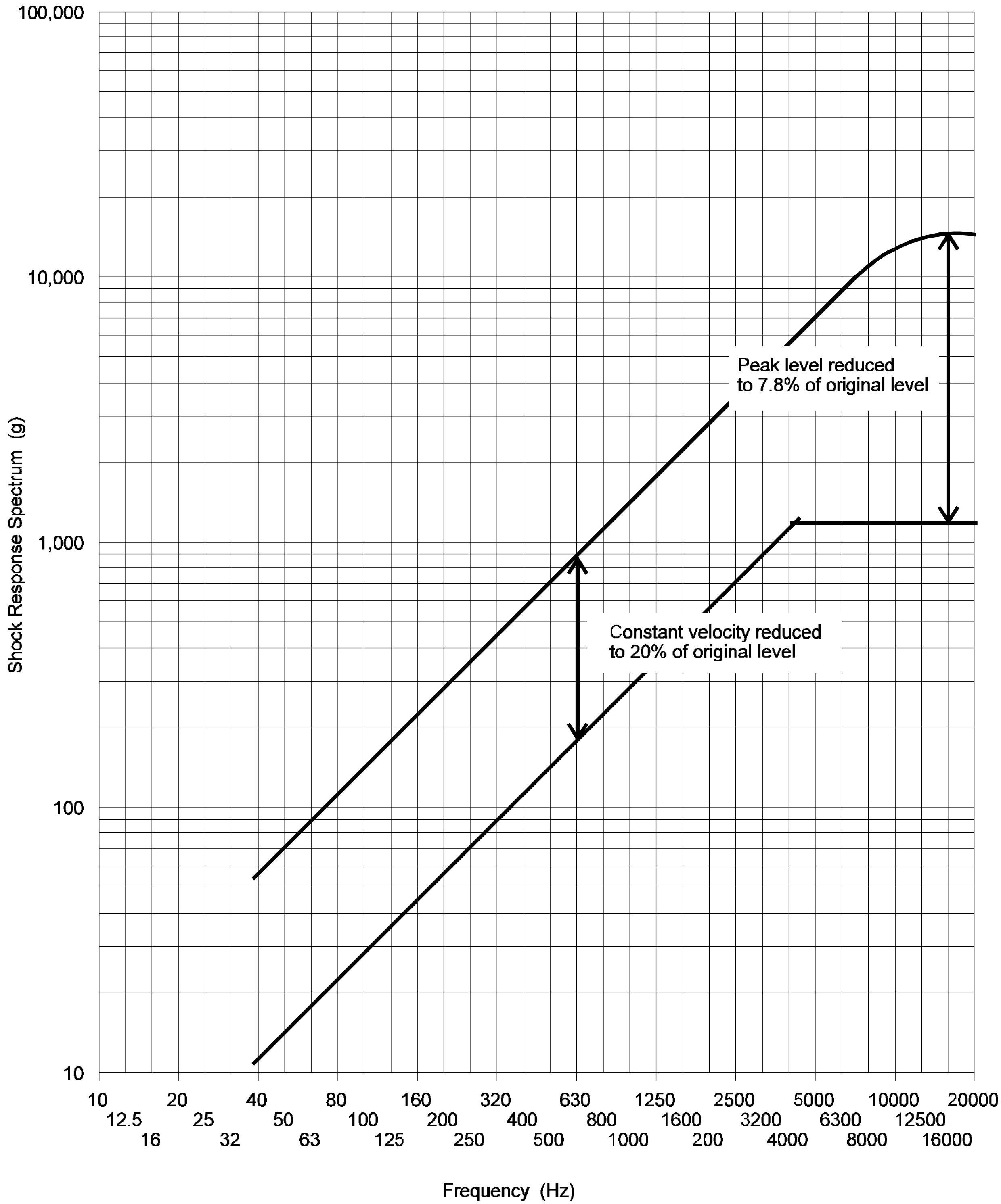
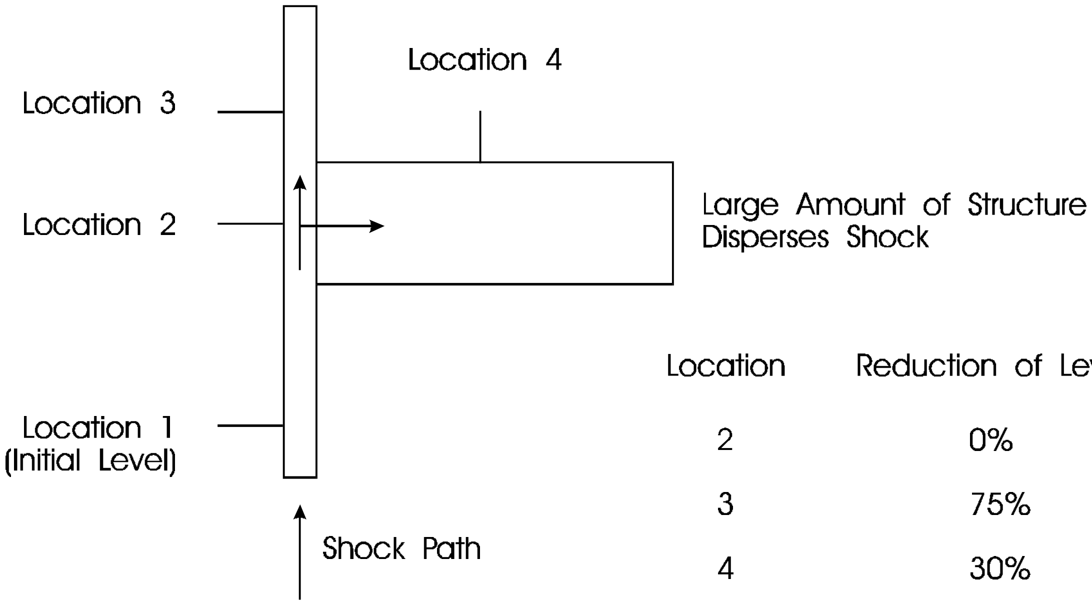
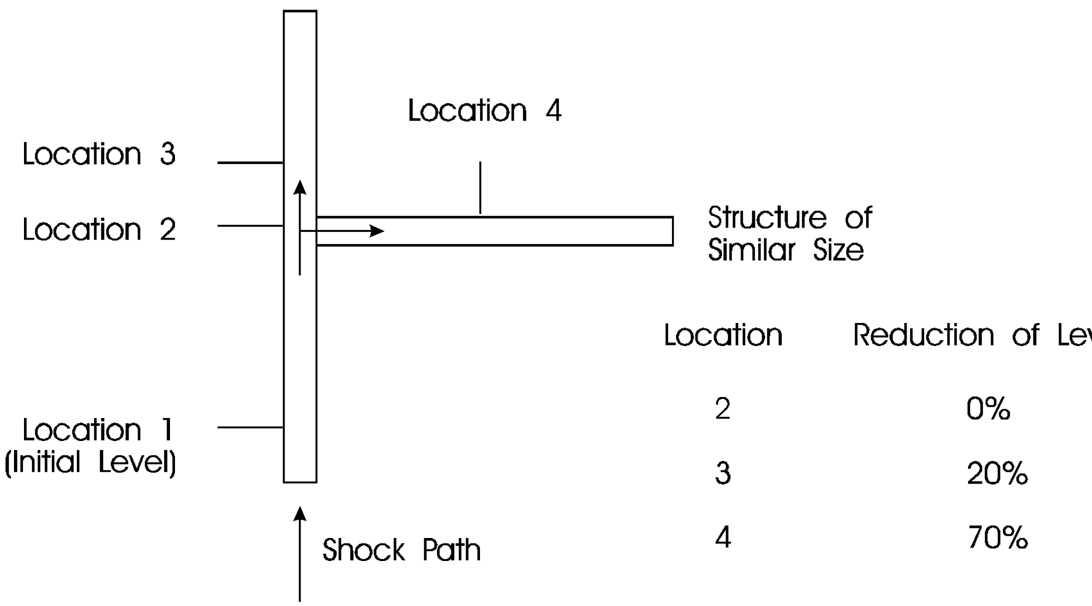


Figure A-10 Shock Attenuation Example



Location	Reduction of Level
2	0%
3	75%
4	30%



Location	Reduction of Level
2	0%
3	20%
4	70%

Figure A-11 Reduction of Pyrotechnic Shock Response due to Intervening Structure

Launch Vehicle Appendices [Appendix B through L]

The following appendices provide maximum expected flight loads and vibroacoustic levels (limit values) for various launch vehicles. The levels are based on data from previous launches, ground tests, and analytical predictions. The levels may be used for initial sizing of spacecraft structure and for test definition; however, the loads and vibroacoustic environments associated with the various phases of a mission (launch, insertion into orbit, orbital operations, landing, etc.) are a function of the launch vehicle configuration, spacecraft design, and mission profile and must be determined on a case-by-case basis and confirmed by the launch vehicle organization.

The data contained in Appendices B through L are based on available documentation, and are subject to change. The data are for information only and are not all inclusive. A verification program must be developed that is consistent with all requirements specified in GEVS, regardless of the launch vehicle that is selected.

APPENDIX B

STS

Structural Loads - Structural loads are determined on a case-by-case basis. Initial design loads shall be provided by the project. Final loads are determined by coupled loads analysis.

Acoustics - The qualification (protoflight) and acceptance test levels for payloads that measure up to 2.75 m (9 feet) in diameter and that will be exposed directly to the cargo bay environment are given in Table B-1. The levels are based on the environment in the empty cargo bay as defined in the STS ICD (1.7.1.2). For larger payloads, a fill-factor (see Appendix A), or computer programs such as PACES or VAPEPS shall be used to estimate the effects of the payload on the acoustic environment and the results shall be used as a basis for modifying the levels of Table B-1. In addition, payload elements in proximity to the cargo bay vent doors may experience a higher noise level in the one-third octave band centered at 315 Hz because of an acoustic tone generated by the vents during transonic flight. When applicable, consideration shall be given to modifying the acoustic specification to account for this effect. Finally, if protective acoustic devices are employed in the payload design, their attenuation characteristics shall be used to adjust the levels of Table B-1.

Component Random Vibration - Generalized component qualification, and acceptance, random vibration test levels are given in Table B-2.

Mechanical Shock - The mechanical shock environment produced by the orbiter is considered negligible. The hardware must be qualified for self-induced shocks or for shocks produced by propulsion assist upper stages and separation.

Table B-1
Acoustic Test Levels
STS Cargo Bay
Payloads up to 2.75 meters (9 feet) in diameter

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	122.0	119.0
32	125.0	122.0
40	128.0	125.0
50	130.5	127.5
63	131.5	128.5
80	132.0	129.0
100	132.0	129.0
125	132.0	129.0
160	131.5	128.5
200	130.5	127.5
250	130.0	127.0
315	129.0	126.0
400	128.0	125.0
500	127.0	124.0
630	126.0	123.0
800	124.5	121.5
1000	123.0	120.0
1250	121.5	118.5
1600	119.5	116.5
2000	118.5	115.5
2500	116.0	113.0
3150	114.5	111.5
4000	112.5	109.5
5000	111.0	108.0
6300	109.0	106.0
8000	107.5	104.5
10000	106.0	103.0
Overall	142	139

Table B-2
Generalized Random Vibration Test Levels
STS Components
22.7-kg (50 lb) or less

Frequency (Hz)	ASD Level (G^2/Hz)	
	Qualification	Acceptance
20	.025	.0125
20-50	+6 dB/oct	+6 dB/oct
50-600	.15	.075
600-2000	-4.5 dB/oct	-4.5 dB/oct
2000	.025	.0125
Overall	12.9 G_{rms}	9.1 G_{rms}

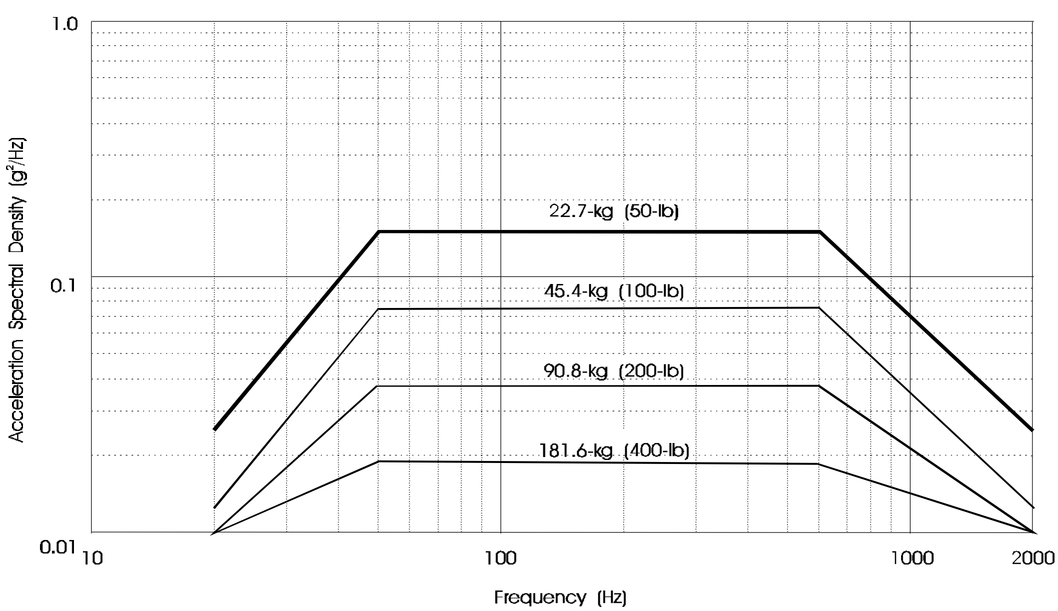
The acceleration spectral density level may be reduced for components weighing more than 22.7-kg (50 lb) according to:

	Weight in kg	Weight in lb	
dB reduction	= 10 LOG(W/22.7)	10 LOG(W/50)	
$ASD_{(50\text{-}800\ Hz)}$	= .15·(22.7/W)	.15·(50/W)	for protoflight
$ASD_{(50\text{-}800\ Hz)}$	= .075·(22.7/W)	.075·(50/W)	for acceptance

Where W = component weight.

The slopes shall be maintained at +6 and -4.5 for components weighing up to 57-kg (125-lb). Above that weight, the slopes shall be adjusted to maintain an ASD level of 0.01 G^2/Hz at 20 and 2000 Hz.

For components weighing over 182-kg (400-lb), the test specification will be maintained at the level for 182-kg (400 pounds).



APPENDIX C

ATLAS

Structural Loads

Limit load factors for various mission events are given in Table C-1 for the commercial Atlas I, II, IIA, and IIAS launch vehicles. The load factors given are intended to provide a conservative design envelope for a typical spacecraft in the 1800 kg (4000 lb) to 3600 kg (8000 lb) weight class with first lateral modes above 10 Hz and first axial mode above 15 Hz. In addition, the center of gravity offset from the payload adapter interface is in the range of 89-152 cm (35-60 inches).

Gust/flight wind is a low frequency event (<12 Hz) that produces maximum loss of clearance between the spacecraft and payload fairing, and high loads near the base of the spacecraft primary structure. BECO/BPJ excites all frequencies (3 to 40 Hz) and produces the majority of the maximum loads throughout the spacecraft. MECO excites all frequencies and produces the highest tension (negative axial) loads and sometimes the maximum loads on secondary structure.

Table C-1
ATLAS I, II, IIA & IIAS
Limit Load Factor (G)
at spacecraft C.G.

Event	Axial	Lateral
Launch		
ATLAS I, II, IIA	1.2 ± 1.2	± 1.0
ATLAS IIAS	1.3 ± 1.8	± 1.3
Flight Winds	2.2 ± 0.3	0.4 ± 1.2
BECO/BPJ		
(max axial)	5.2 ± 0.5	± 0.5
(max lateral)	2.5-1.0 ± 1.0	± 2.0
SECO*	2.0-0.0 ± 0.4	± 0.3
MECO*	4.0-0.0 ± 0.5	± 0.2
	0.0 ± 2.0	± 0.6

Note: Dynamic Uncertainty Factors (DUF's) are not accounted for in the above load factors.

* Decaying to zero.

+ Is compression.

BECO = Booster Engine Cut-off
BPJ = Booster Package Jettison
SECO = Sustainer Engine Cut-off
MECO = Main Engine Cut-off

Acoustics

Qualification and acceptance acoustic test levels are given in Tables C-2 and C-3 for the ATLAS I, II, or IIA with 3.4 m (11-ft) and 4.3 m (14-ft) payload fairings respectively. The acoustic levels for the ATLAS IIAS are given in Table C-4.

For the 4.3 m (14-ft) payload fairing with acoustic blanket, special consideration should be given to components located within 76 cm (30-in.) of the payload fairing vents; the expected sound pressure level can be greater than the levels given in Tables C-3 and C-4 at higher frequencies. Table C-5 gives expected Sound pressure levels for components located 0.3 m (1 ft) from the vents. The 3.4 m (11-ft) payload fairing vents are fewer in number and located farther from the spacecraft envelope.

Spacecraft Random Vibration

The maximum expected random vibration flight levels (limit levels) at the spacecraft interface are given in Table C-6.

Sine Vibration

The maximum expected sine vibration levels given in the ATLAS user's guide are given in Table C-7.

Mechanical Shock

Test levels representing typical spacecraft separation and payload nose fairing and insulation panel jettison are given in Tables C-8.

The maximum acceptable shock level at the equipment module interface for a customer-provided separation system is given in Figure C-1.

Table C-2
ATLAS I, II, & IIA
Acoustic Test Levels
Inside 3.4 m (11 ft) Payload Fairing
Assumes 50-60% Fill by Cross Section Area)

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa			
	Without Acoustic Blankets		With Acoustic Blankets	
	Qualification	Acceptance	Qualification	Acceptance
25	121	118	121	118
32	123	120	123	120
40	124.5	121.5	124.5	121.5
50	126	123	126	123
63	128	125	128	125
80	129	126	129	126
100	130.5	127.5	130	127
125	132	129	131	128
160	132.5	129.5	131	128
200	133.5	130.5	131.5	128.5
250	134	131	131	128
315	133	130	129	126
400	132	129	127	124
500	131	128	125	122
630	129.5	126.5	123.5	120.5
800	127	124	121	118
1000	125	122	119	116
1250	122	119	116	113
1600	120	117	114	111
2000	119	116	113	110
2500	118.5	115.5	112.5	109.5
3150	118	115	112	109
4000	117.5	114.5	111.5	108.5
5000	117	114	111	108
6300	116.5	113.5	110.5	107.5
8000	116	113	110	107
10000	115.5	112.5	109.5	106.5
Overall	143	140	140	137

Table C-3
ATLAS I, II, & IIA
Acoustic Test Levels
Inside 4.3 m (14 ft) Payload Fairing
Assumes 50-60% Fill by Cross Section Area)

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa			
	Without Acoustic Blankets		With Acoustic Blankets	
	Qualification	Acceptance	Qualification	Acceptance
25	116	113	116	113
32	120	117	120	117
40	123.5	120.5	124.5	121.5
50	124.5	121.5	125.5	122.5
63	125.5	122.5	127	124
80	126.5	123.5	128	125
100	127	124	129.5	126.5
125	128	125	131	128
160	128	125	132	129
200	128	125	133	130
250	127.5	124.5	132.5	129.5
315	127	124	131.5	128.5
400	126	123	131	128
500	123.5	120.5	130.5	127.5
630	121.5	118.5	129	126
800	119.5	116.5	126.5	123.5
1000	116.5	113.5	123.5	120.5
1250	114.5	111.5	121	118
1600	113	110	121	118
2000	113	110	122	119
2500	111	108	119.5	116.5
3150	111	108	118	115
4000	110.5	107.5	117	114
5000	110	107	116.5	113.5
6300	110.5	107.5	116	113
8000	112.5	109.5	116.5	113.5
10000	113.5	110.5	117.5	114.5
Overall	138	135	142	139

Table C-4
ATLAS IIAS
Acoustic Test Levels
Inside 4.3 m (14-ft) Payload Fairing
Assumes 50-60% Fill by Cross Section Area)

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa			
	Without Acoustic Blankets		With Acoustic Blankets	
	Qualification	Acceptance	Qualification	Acceptance
25	117	114	117	114
32	121	118	121	118
40	124.5	121.5	125	122
50	125.5	122.5	126	123
63	127	124	127.5	124.5
80	127.5	124.5	129	126
100	128.5	125.5	130.5	127.5
125	129	126	132	129
160	129.5	126.5	133	130
200	130	127	134	131
250	129.5	126.5	133.5	130.5
315	129.5	126	133	130
400	128	125	133	130
500	126.5	123.5	133	130
630	125	122	131.5	128.5
800	122.5	119.5	130	127
1000	119.5	116.5	127	124
1250	117	114	125	122
1600	115	112	123.5	120.5
2000	115	112	122	119
2500	114	111	121	118
3150	113	110	120	117
4000	112	109	118.5	115.5
5000	111.5	108.5	117.5	114.5
6300	111	108	116.5	113.5
8000	112.5	109.5	117	114
10000	113.5	110.5	118	115
Overall	139	136	143	140

Table C-5
Acoustic Levels 0.3 m (1 ft) from the Vents
for the 4.3 m (14 ft) Payload Fairing

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
1600	113.5	110.5
2000	115.5	112.5
2500	115.5	112.5
3150	114	111
4000	115	112
5000	116.5	113.5
6300	116.5	113.5
8000	117	114
10000	117.5	114.5

Table C-6
ATLAS I, II, IIA, IIAS
Spacecraft Random Vibration
Limit Levels

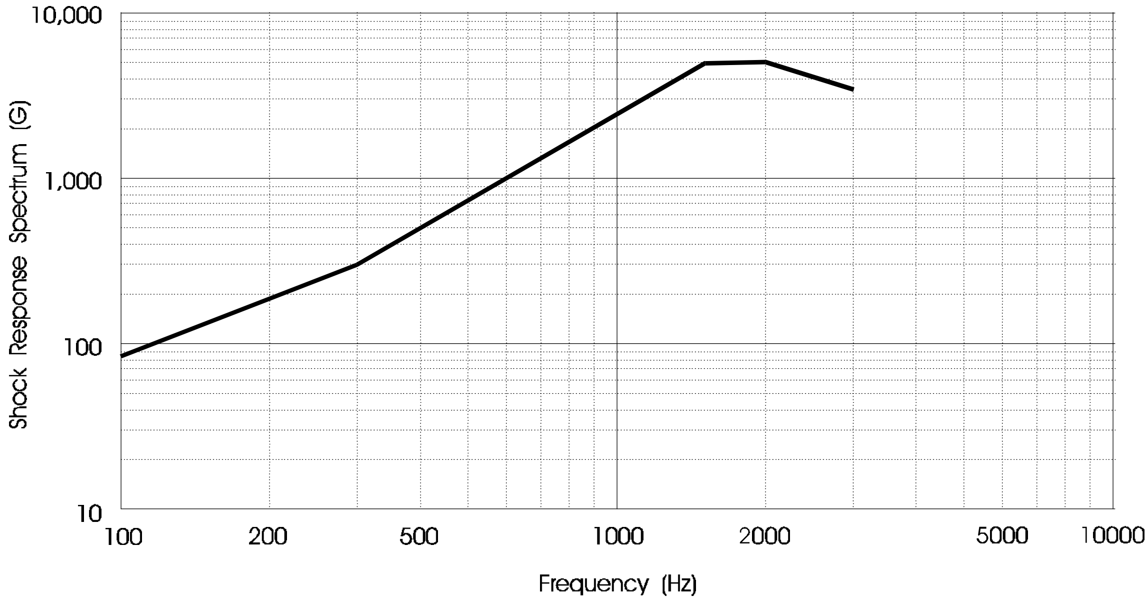
Frequency (Hz)	ASD Level (G^2/Hz)
20	.00048
20-80	+9 dB/oct
80-200	.03
200-2000	-9 dB/oct
2000	.00003
Overall	2.7 G_{rms}

Table C-7
ATLAS I, II, IIA, & IIAS
Maximum Expected Spacecraft Interface
Sinusoidal Vibration Environment

Frequency (Hz)		Sine Vibration Level ($G_{0\text{-}p}$)
Thrust Axis	5-6.2 6.2-100	12.5-mm (0.5-in) DA 1.0
Lateral Axes	5-100	0.7

Table C-8
ATLAS I, II, IIA, & IIAS
Spacecraft Separation
Shock Response Spectrum
Q=10

Event	Frequency (Hz)	Shock Response Spectrum (G)	
		Qualification	Acceptance
Spacecraft Separation			
Type D Payload Adapter [1.65 m (66 in)]	100 100-800 800-3000	210 +7.1 dB/oct 4200	150 +7.1 dB/oct 3000
Type B & B1 Payload Adapter [1.18 m (47 in)]	100 100-1500 1500-3000	140 8.5 dB/oct 6300	100 8.5 dB/oct 4500
Type A & A1 Payload Adapter [0.92 m (37 in)]	100 100-1500 1500-3000	70 10 dB/oct 6300	50 10 dB/oct 4500
Payload Fairing and Insulation Panel Jettison	100 100-500 1500-2000	20 5.4 dB/oct 84	14 5.4 dB/oct 60



.

Figure C-1 Maximum Allowable Spacecraft-Produced Shock at Equipment Module Interface

APPENDIX D

DELTA

The Delta launch vehicle configuration is specified by a four digit number designating the configuration of the launch vehicle stages. For other than the standard payload fairing, the designation of the fairing is added to the end of the number. The current designators are as follows:

1st digit - Type of augmentation/ first stage:
- 2 - Castor II augmentation, extended long tank, RS-27 engine.
- 3 - Castor IV augmentation, extended long tank, RS-27 engine.
- 4 - Castor IVA augmentation, extended long tank, MB-3 engine.
- 5 - Castor IVA augmentation, extended long tank, RS-27 engine.
- 6 - Castor IVA augmentation, extra extended long tank, RS-27 engine.
- 7 - GEM solid motors augmentation, extra extended long tank, modified RS-27 engine.

2nd digit - Quantity of augmentation motors:
- 3 - Three motors.
- 9 - Nine motors.

3rd digit - Type of second stage:
- 1 - Standard second stage [4536 Kg (10,000 lb) propellant, TRW TR-201 engine].
- 2 - Uprated second stage [5987 Kg (13,200 lb) propellant, AJC ITIP engine].

4th digit - Type of third stage:
- 0 - No third stage.
- 3 - TE-364-3 third stage [653 Kg (1,440 lb) propellant].
- 4 - TE-364-4 third stage [1043 Kg (2,300 lb) propellant].
- 5 - PAM-D third stage [2009 Kg (4,430 lb) propellant max.].

Payload fairing size is designated as:

- None - Standard fairing [2.9 m (9.5 ft)].
- -8 - 2.4 m (8 ft) diameter fairing.
- -10 - 3 m (10 ft) diameter fairing, 7.9 m (26 ft) fairing length.

Structural Loads

Preliminary limit load factors for the Delta II launch vehicle are given in Table D-1.

Table D-1
DELTA II
Limit Load Factor (G)
at Spacecraft C.G.

Axis	Liftoff		MECO
	Thrust	Lateral	
Lateral	± 2.0 ± 2.5(1)	± 2.0 ± 3.0(1)	-
Thrust	+2.4/-0.2(2)	+2.4/-0.2(2)	+6.0 ± 0.6(3)

(1) To provide correct bending moment at spacecraft separation plane, use lateral load factors of ± 2.5 for two stage and ± 3.0 for three stage Delta vehicles.

(2) Plus indicates compression load.

(3) This value, based on a three-stage mission with 1900-Kg (4200 lb) spacecraft, consists of a static component which is a function of spacecraft weight (see Figure D-1) and a dynamic component.

Examples of steady-state axial acceleration at MECO and third-stage burnout are given in Figures D-1 and D-2.

To avoid dynamic coupling between low-frequency vehicle and spacecraft modes, the stiffness of the spacecraft structure should be designed to produce fundamental frequencies above 35 Hz in the thrust axis and 15 Hz in the lateral axis for a spacecraft hard-mounted at the spacecraft separation plane (i.e., without payload attach fitting and separation clamp). The lateral fundamental frequency should be greater than 12 Hz if a two stage DELTA vehicle is used.

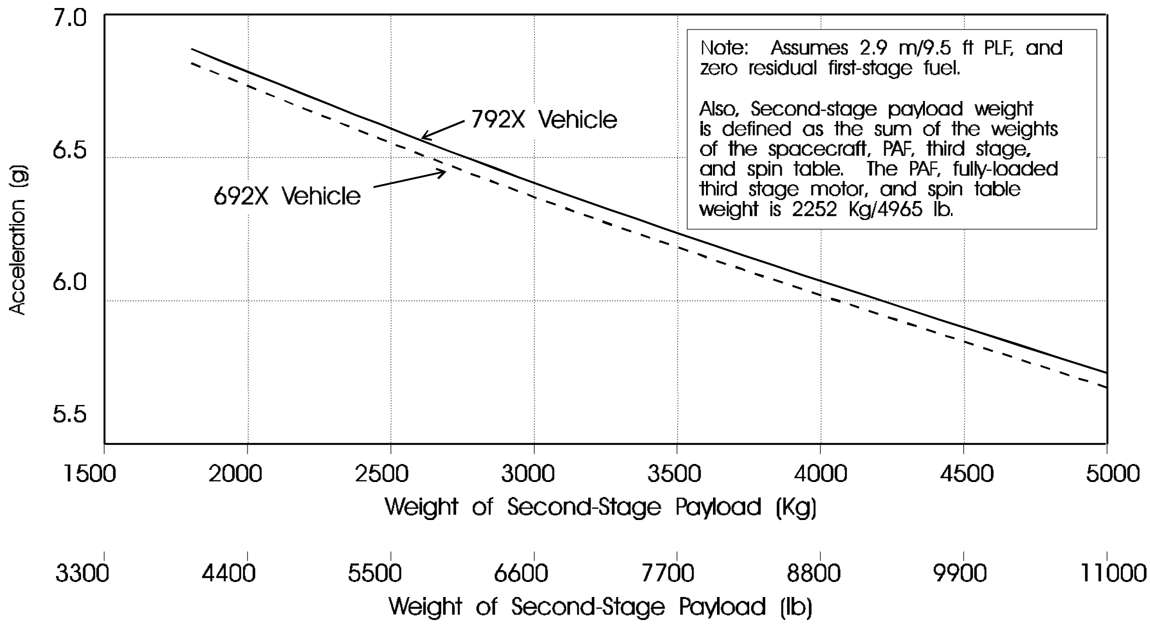


Figure D-1 DELTA II Axial Steady-State Acceleration at MECO Versus Second-Stage Payload Weight

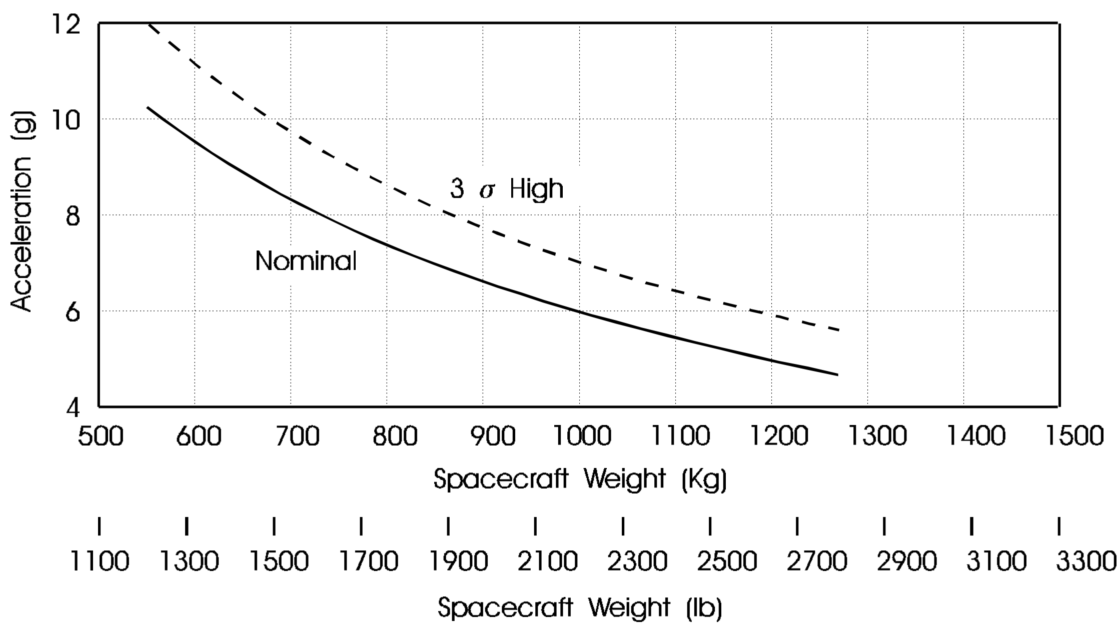


Figure D-2 DELTA II Axial Steady-State Acceleration at Third Stage Burnout

Acoustics

The qualification and acceptance acoustic test levels for a three stage Delta-II utilizing the 2.9 m (9.5-ft) and 3 m (10-ft) payload fairings are given in Tables D-2 and D-3 respectively. For a two stage Delta-II using a 3-m/10-ft payload fairing, the levels are the same as for the three stage version. For a three stage DELTA II utilizing the 2.4 m (8-ft) payload fairing, the acoustic test levels are given in Table D-4.

Spacecraft Random Vibration

The maximum expected random vibration levels (limit levels) at the spacecraft interface during Delta II launch are given in Table D-6.

Sine Vibration

The Delta user's guide provides the maximum expected levels given in Table D-7.

Mechanical Shock

The maximum expected separation shocks for the two and three stage Delta II vehicles are given in Tables D-7 and D-8.

Spacecraft Spin Balance

Refer to the Delta II Commercial Payload Planner's Guide, McDonnell Douglas Corporation, document MDC H3224B for the spacecraft spin balance requirements.

Table D-2
DELTA II (7925 Vehicle)*
Acoustic Test Levels
Inside 2.9 m (9.5-ft) Payload Fairing
with 7.6 cm (3 in.) Acoustic Blankets Installed

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	-	-
32	124.5	121.5
40	127	124
50	129	126
63	130	127
80	131.5	128.5
100	132	129
125	132.5	129.5
160	132.5	129.5
200	133	130
250	133	130
315	133	130
400	132	129
500	129	126
630	126.5	123.5
800	123	120
1000	119.5	116.5
1250	117.5	114.5
1600	115	112
2000	112.5	109.5
2500	110	107
3150	108.5	105.5
4000	106.5	103.5
5000	106	103
6300	105	102
8000	104.5	101.5
10000	104.5	101.5
Overall	142.6	139.6

* For the 6925 vehicle, decrease all levels by 0.5 dB.
For 6920 or 7920 vehicles, contact the System Reliability & Safety Office (Code 302).

Table D-3
DELTA II (7920 and 7925 Vehicles)*
Acoustic Test Levels
Inside 3 m (10-ft) Payload Fairing
with 7.6 cm (3 in.) Acoustic Blankets Installed

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	-	-
32	122.5	119.5
40	125.5	122.5
50	128	125
63	130	127
80	131.5	128.5
100	132.5	129.5
125	133	130
160	133	130
200	133	130
250	133	130
315	133	130
400	132.5	129.5
500	131	128
630	128	125
800	125	122
1000	123	120
1250	121	118
1600	120	117
2000	119.5	116.5
2500	119	116
3150	118	115
4000	116.5	113.5
5000	114	111
6300	110	107
8000	106	103
10000	106	100
Overall	143	140

* For 6920 and 6925 vehicles, decrease all levels by 0.5 dB.

Table D-4
DELTA II (7920 and 7925 Vehicles)*
Acoustic Test Levels
Inside 3 m (10-ft) Payload Fairing
with 3.8 cm (1.5 in.) Acoustic Blankets Installed

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	-	-
32	122.5	119.5
40	125.5	122.5
50	128.5	125.5
63	131	128
80	133	130
100	134	131
125	134.5	131.5
160	135	132
200	135	132
250	135	132
315	135	132
400	135	132
500	133.5	130.5
630	130.5	127.5
800	127.5	124.5
1000	125	122
1250	122.5	119.5
1600	121	118
2000	120	117
2500	119.5	116.5
3150	118.5	115.5
4000	117.5	114.5
5000	115.5	112.5
6300	111.5	108.5
8000	107.5	104.5
10000	104	101
Overall	145	142

* For 6920 and 6925 vehicles, decrease all levels by 0.5 dB.

Table D-5
DELTA II 7920 and 7925 Vehicles*
Acoustic Test Levels
Inside 2.4 m (8 ft) Payload Fairing,
with 3.8 cm (1.5 in) Blanket

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	122	119
32	123.5	120.5
40	125	122
50	126.5	123.5
63	128	125
80	129.5	126.5
100	131	128
125	132.5	129.5
160	134	131
200	134.5	131.5
250	135.5	132.5
315	137	134
400	139	136
500	140.5	137.5
630	139	136
800	135	132
1000	132	129
1250	131	128
1600	130.5	127.5
2000	129.5	126.5
2500	128.5	125.5
3150	127	124
4000	125.5	122.5
5000	124.5	121.5
6300	123.5	120.5
8000	122.5	119.5
10000	121.5	118.5
Overall	148	145

* For 6920 and 6925 vehicles, decrease all levels by 0.5 dB.

Table D-6
DELTA II
Spacecraft Random Vibration
Limit Levels

Frequency (Hz)	ASD Level (G^2/Hz)
20	.0016
20-300	+4 dB/oct
300-700	.06
700-2000	-3 dB/oct
2000	.021
Overall	8.7 G_{rms}

Table D-7
DELTA II
Maximum Expected Spacecraft Interface
Sinusoidal Vibration Environment

Frequency (Hz)		Sine Vibration Level ($G_{0\text{-}p}$)
Thrust Axis	5 - 6.2 6.2 - 100	12.5-mm (0.5-in) DA 1.0
Lateral Axes	5 - 100	0.7

Table D-8
DELTA II (6920 & 7920 Vehicles)
Maximum Flight Shock Levels
(Explosive Nut Separation System)
Q=10

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
350	140	100
350-1700	+12.3 dB/oct	+12.3 dB/oct
1700	3500	2500
1700-4000	+5.5 dB/oct	+5.5 dB/oct
4000-5000	7700	5500
5000-10000	-9 dB/oct	-9 dB/oct
10000	2730	1950

Table D-9
DELTA II (6920 & 7920 Vehicles)
Maximum Flight Shock Levels
(Clampband Separation System)
Q=10

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100	140	100
100-800	+9.9 dB/oct	+9.9 dB/oct
800-3000	4200	3000

Table D-10
DELTA II (6925 & 7925 Vehicles)
Maximum Flight Shock Levels
(Clampband Separation System)
Q=10

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100	56	40
100-1500	+10.3 dB/oct	+10.3 dB/oct
1500-3000	5740	4100

APPENDIX E

TITAN

Structural Loads

Limit load factors are given in Table E-1 for the Titan II launch vehicle. These load factors are for preliminary use and will be updated as the program develops.

Table E-1
TITAN II
Limit Load Factor (G)
at Spacecraft C.G.
1350-2270 kg (3000-5000 lb) Payload

Event	Vehicle			
	Titan II G	Titan II G/ Third Stage	Titan II S/ Enhanced ACS	Titan II S/ Third Stage
Liftoff				
- Axial	3.0 to -0.5	2.5 to -1.0	2.1 to 0	2.1 to 0
- Lateral	± 3.0	± 3.5	± 1.0	± 1.0
Maximum Airloads				
- Axial	3.0 to 1.0	**	**	**
- Lateral	± 2.5			
Stage I Burnout				
- Axial	8.5 to 3.0	7.5 to 3.0	8.5 to 3.0	7.5 to 3.0
- Lateral	± 2.5	± 3.5	± 3.5	± 3.5
Stage II Shutdown				
- Axial*	10.0 to 1.0	6.5 to 2.5	8.5 to 1.0	6.5 to 2.5
- Lateral	± 1.0	± 3.0	± 1.0	± 3.0

* Function of spacecraft weight, assume 2270 kg (5000-lb) spacecraft and may be reduced by Stage-II reduced thrust modification.

** Enveloped by other load events

Acoustics

The qualification and acceptance test levels are given in Tables E-2 and E-3 for various payload fairings (PLF).

Mechanical Shock

The qualification and acceptance shock test levels representing the maximum expected Titan II induced shocks at the payload interface are given in Table E-4.

Table E-2
TITAN IIG
Acoustic Test Levels
with Acoustic Blankets
(Inside Payload Fairing)

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	110	107
32	112.5	109.5
40	118	115
50	117	114
63	119.5	116.5
80	124	121
100	125	122
125	126.5	123.5
160	126	123
200	126	123
250	127	124
315	128.5	125.5
400	128.5	125.5
500	128.5	125.5
630	129	126
800	128.5	125.5
1000	126.5	123.5
1250	127.5	124.5
1600	126	123
2000	123.5	120.5
2500	122	119
3150	120	117
4000	115	112
5000	112.5	109.5
6300	109.5	106.5
8000	105	102
10000	102.5	99.5
Overall	139	136

Table E-3
TITAN IIS
Acoustic Test Levels
with Acoustic Blankets
(Inside Payload Fairing)

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	116	113
32	119	116
40	120.5	117.5
50	123	120
63	125.5	122.5
80	127.5	124.5
100	128	125
125	128.5	125.5
160	129	126
200	129	126
250	129.5	126.5
315	129.5	126.5
400	129.5	126.5
500	129.5	126.5
630	129.5	126.5
800	128	125
1000	125.5	122.5
1250	124.5	121.5
1600	121.5	118.5
2000	121	118
2500	120	117
3150	120	117
4000	116.5	113.5
5000	115.5	112.5
6300	113.5	110.5
8000	110	107
10000	107	104
Overall	140	137

Table E-4
TITAN II
Shock at Payload Interface
Q=10

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100	100	70
100-500	+3.9 dB/oct	+3.9 dB/oct
500-1250	280	200
1250-5000	-4.9 dB/oct	-4.9 dB/oct
5000	90	65

Structural Loads

Limit load factors are given in Table E-5 for the Titan III commercial launch vehicle. These load factors are for preliminary use and will be updated as the program develops.

Table E-5
TITAN III
Limit Load Factors
at Spacecraft C.G.

Loading Condition	Limit Load Factor (G)	
	Axial	Lateral
Maximum Lateral	2.75	± 1.8
Maximum Axial	+5.4 / -1.3	± 0.5

Note: Assumes use of water suppression system at lift-off. Non-steady-state part of loads contains a dynamic uncertainty factor of 1.5.

In general, it is recommended that the first mode frequencies for the fixed base payload, including adapter, be greater than 15 Hz lateral and 26 Hz axial.

Acoustics

Typical acoustic levels are given in Table E-6 for the Titan III vehicle. The levels should be verified with the Titan III program office.

Spacecraft Random Vibration

The maximum expected random vibration flight levels (limit levels) at the spacecraft interface are given in Table E-7

Table E-6
TITAN III
Acoustic Test Levels
Inside Payload Fairing
(Assumes 50% Payload Fill)

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	130	127
32	131	128
40	132.5	129.5
50	133	130
63	134	131
80	134.5	131.5
100	135	132
125	135	132
160	135	132
200	134.5	131.5
250	133.5	130.5
315	132.5	129.5
400	131.5	128.5
500	129.5	126.5
630	128	125
800	126	123
1000	124	121
1250	122	119
1600	120	117
2000	118	115
2500	116	113
3150	114	111
4000	112	109
5000	110	107
6300	108	105
8000	106	103
10000	104	101
Overall	145	142

Table E-7
TITAN III
Spacecraft Random Vibration
Limit Levels

Frequency (Hz)	ASD Level (G^2/Hz)
20	.002
20-160	+5.0 dB/oct
160-800	.018
800-2000	-6.0 dB/oct
2000	.00065
Overall	4.2 G_{rms}

Mechanical Shock

The maximum expected launch vehicle induced mechanical shock at the spacecraft interface is given in Table E-8 for a V-band separation system, and in Figure E-1 for the Expanding Tube Separation System (ETSS).

Table E-8
TITAN III
Maximum Expected Shock Spectrum
(V-Band Separation System)
Q=10

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
160	140	100
160-1250	+10.9 dB/oct	+10.9 dB/oct
1250-10000	5740	4100

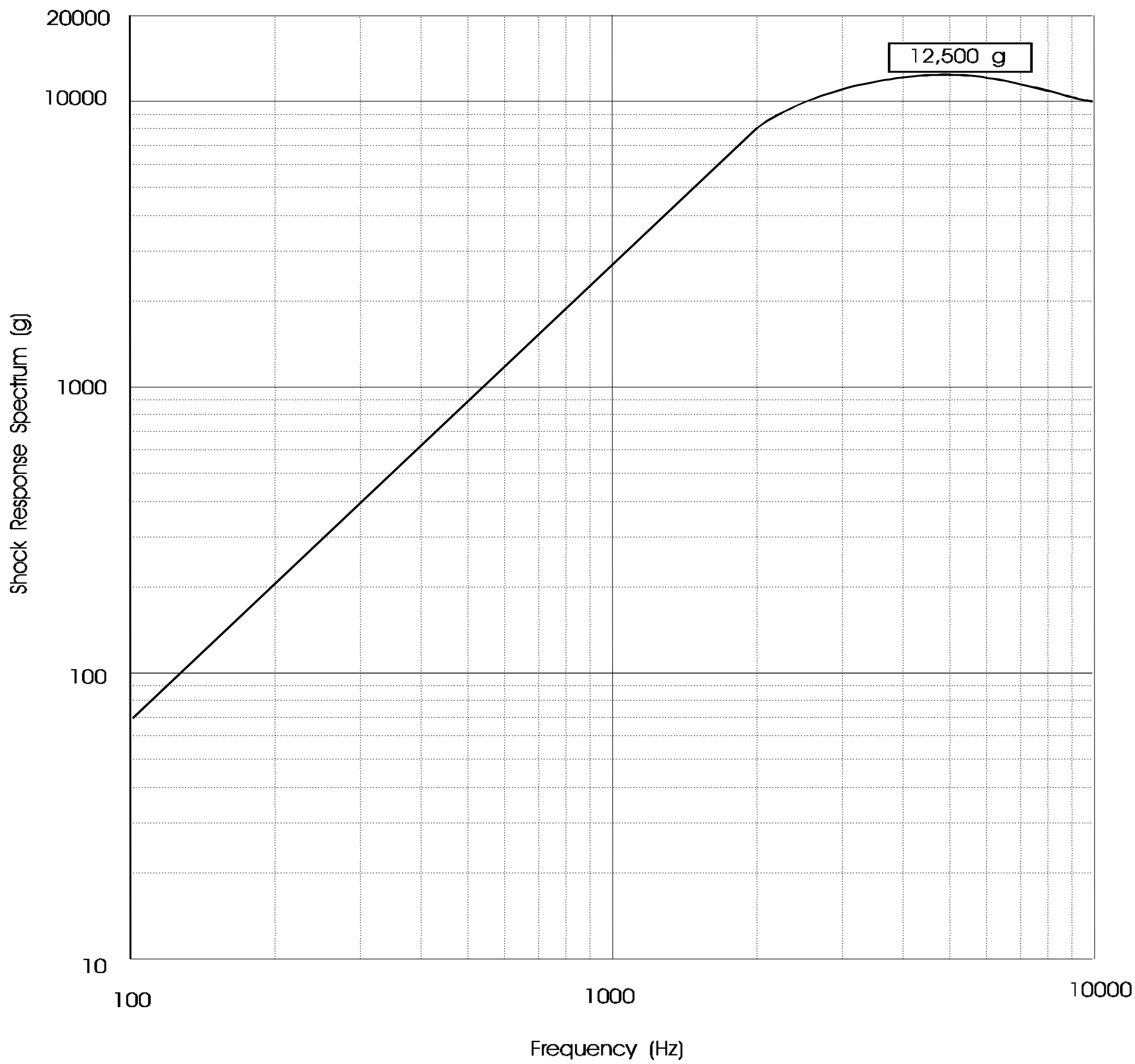


Figure E-1 Titan III Maximum Expected Shock Response at Payload Interface for the Expanding Tube Separation System (ETSS)

Structural Loads

The spacecraft response is a function of its weight, stiffness, and lateral/axial coupling as well as the launch vehicle/booster configuration. Limit load factors are provided in Tables E-9 through E-12 for preliminary evaluation of spacecraft primary structure utilizing various TITAN IV configurations and spacecraft weights. For these limit loads, the spacecraft center of gravity is assumed to be located approximately 30-40% of the spacecraft length from the spacecraft/launch vehicle interface. Transient loads analyses must be performed during the development of the spacecraft to provide detailed member loads and responses required for complete design and evaluation of the structure including interface loads and loss-of-clearance.

Significant spacecraft lateral excitation can occur in the 4-10 Hz range due to liftoff and maximum airloads, and significant axial excitation can occur in the 17-24 Hz range due to Stage I Shutdown thrust oscillations.

To minimize the interaction between low frequency lateral modes and the launch vehicle control system performance, the first lateral mode should be above 2.5 Hz.

Acoustics

The qualification and acceptance acoustic levels are given in Table E-10 for the Titan IV vehicle. These represent the levels for an empty payload fairing and should be adjusted by a fill-factor. The expected levels should be confirmed with the Titan IV program office since they are dependent on the payload fairing length, the location and number of vents and access doors, the amount and location of acoustic blankets, and spacecraft volume and acoustic properties.

Mechanical Shock

The primary sources of pyroshock during a launch occur at SRM separation, Stage I/II separation, PLF separation, Upper Stage separation, if applicable, and payload separation. The last three are the only ones of significance to the spacecraft. The maximum expected launch vehicle induced shocks at the spacecraft interface for the Titan IV/ Centaur and Titan IV/ NUS (no upper stage), configurations are given in Figures E-2 and E-3 respectively.

The maximum allowable spacecraft produced shock response levels for Centaur 8- and 22-hardpoint configurations are given in Figures E-4 and E-5 respectively

Figure E-6 gives the maximum IUS induced and maximum spacecraft induced shock levels at the IUS-spacecraft interface.

Table E-9
TITAN IV
Limit Load Factors
at Spacecraft C.G.
Centaur Configuration
4,500-6,800 kg (10,000-15,000 lb) Spacecraft

Event	Load Factor			
	Direction	Static	Dynamic	Total
Lateral (liftoff and maximum airloads)	Axial Lateral Torsion Rotation	1.0/2.0 0.0 0.0 0.0	± 1.0 ± 2.5 ± 0.02 ± 0.03	0.0/3.0 ± 2.5 ± 0.02 ± 0.03
Axial (stage I and II shutdown)	Axial Lateral Torsion	0.0/4.0 0.0 0.0	± 2.0 ± 1.5 ± 0.02	-2.0/6.0 ± 1.5 ± 0.02
Torsion (FBR release)	Axial Torsion	2.0 0.0	0.0 ± 0.04	2.0 ± 0.04

Notes:
1. Load factors are limit values at the spacecraft center of gravity in g for the axial and lateral values and g/in. for torsion and rotation.
2. Lateral and rotational load factors are RSS values which may be applied at any azimuth.
3. Load factors are for major structural members and do not include margin for component design.
4. Load factors for lateral envelopes OSS and no-OSS liftoff cases.

Table E-10
TITAN IV
Limit Load Factors
at Spacecraft C.G.
IUS Configuration
2,270-3,630 kg (5,000-8,000 lb) Spacecraft

Event	Load Factor			
	Direction	Static	Dynamic	Total
Lateral (liftoff and maximum airloads)	Axial	1.0/2.0	± 1.0	0.0/3.0
	Lateral	0.0	± 3.0	± 3.0
	Torsion	0.0	± 0.03	± 0.03
	Rotation	0.0	± 0.02	± 0.02
Axial (stage I and II shutdown)	Axial	0.0/4.0	± 2.0	-2.0/6.0
	Lateral	0.0	± 1.5	± 1.5
	Torsion	0.0	± 0.02	± 0.02

Notes:

1. Load factors are limit values at the spacecraft center of gravity in g for the axial and lateral values and g/in. for torsion and rotation.
2. Lateral and rotational load factors are RSS values which may be applied at any azimuth.
3. Load factors are for major structural members and do not include margin for component design.
4. Load factors for lateral envelopes OSS and no-OSS liftoff cases.

Table E-11
TITAN IV
Limit Load Factors
at Spacecraft C.G.
NUS Configuration
9,080-13,620 kg (20,000-30,000 lb Spacecraft)

Event	Load Factor			
	Direction	Static	Dynamic	Total
Lateral (liftoff and maximum airloads)	Axial	1.0/2.0	± 1.0	0.0/3.0
	Lateral	0.0	±2.5	± 2.5
	Torsion	0.0	± 0.03	± 0.03
	Rotation	0.0	± 0.03	± 0.03
Axial (stage I and II shutdown)	Axial	0.0/4.0	± 2.0	-2.0/6.0
	Lateral	0.0	± 1.5	± 1.5
	Torsion	0.0	± 0.02	± 0.02
Torsion (FBR release)	Axial	2.0	0.0	2.0
	Torsion	0.0	± 0.01	± 0.01

Notes:
1. Load factors are limit values at the spacecraft center of gravity in g for the axial and lateral values and g/in. for torsion and rotation.
2. Lateral and rotational load factors are RSS values which may be applied at any azimuth.
3. Load factors are for major structural members and do not include margin for component design.
4. Load factors for lateral envelopes OSS and no-OSS liftoff cases.

Table E-12
TITAN IV
Limit Load Factors
at Spacecraft C.G.
NUS Configuration
13,020-18,160 kg (30,000-40,000 lb) Spacecraft

Event	Load Factor			
	Direction	Static	Dynamic	Total
Lateral (liftoff and maximum airloads)	Axial	1.0/2.0	± 1.0	0.0/3.0
	Lateral	0.0	±2.0	± 2.0
	Torsion	0.0	± 0.02	± 0.02
	Rotation	0.0	± 0.02	± 0.02
Axial (stage I and II shutdown)	Axial	0.0/4.0	± 2.0	-2.0/6.0
	Lateral	0.0	± 1.0	± 1.0
	Torsion	0.0	± 0.01	± 0.01

Notes:
1. Load factors are limit values at the spacecraft center of gravity in g for the axial and lateral values and g/in. for torsion and rotation.
2. Lateral and rotational load factors are RSS values which may be applied at any azimuth.
3. Load factors are for major structural members and do not include margin for component design.
4. Load factors for lateral envelopes OSS and no-OSS liftoff cases.

Table E-13
Titan IV
Acoustic Test Levels
(Inside Empty Payload Fairing,
with Acoustic Blankets)

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	126.0	123.0
32	128.0	125.0
40	129.5	126.5
50	130.0	127.0
63	131.0	128.0
80	131.5	128.5
100	132.0	129.0
125	132.0	129.0
160	132.0	129.0
200	131.5	128.5
250	131.0	128.0
315	130/5	127.5
400	129.5	126.5
500	128.5	125.5
630	127.5	124.5
800	126.0	123.0
1000	124.5	121.5
1250	123.0	120.0
1600	121.0	118.0
2000	119.5	116.5
2500	118.0	115.0
3150	116.0	113.0
4000	114.5	111.5
5000	112.5	109.5
6300	110.5	107.5
8000	109.0	106.0
10000	107.0	104.0
Overall	142	139

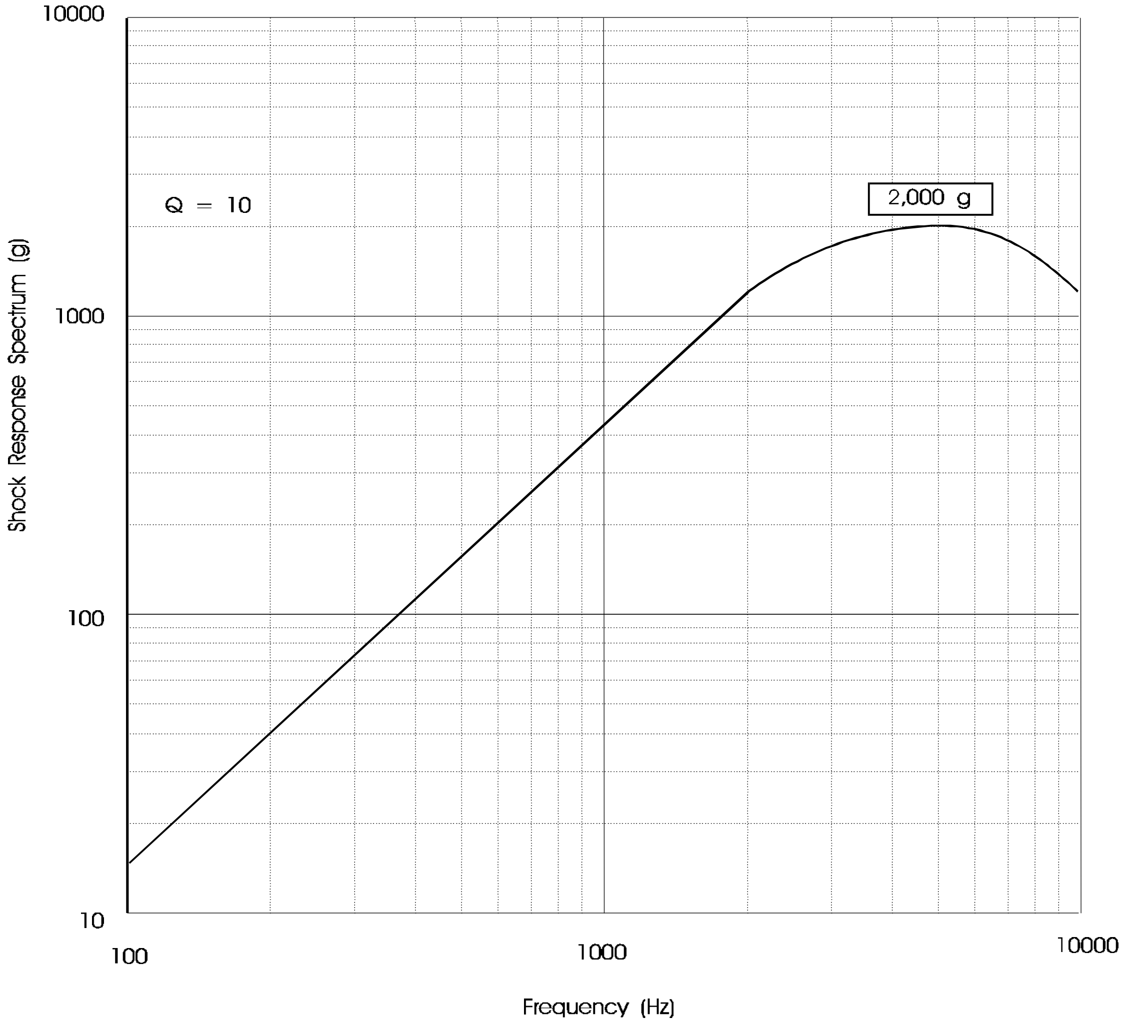


Figure E-2 Maximum Shock Response at the Centaur/Spacecraft Interface Produced by either the TITAN IV or the Centaur (8 or 22 Hardpoint Interface)

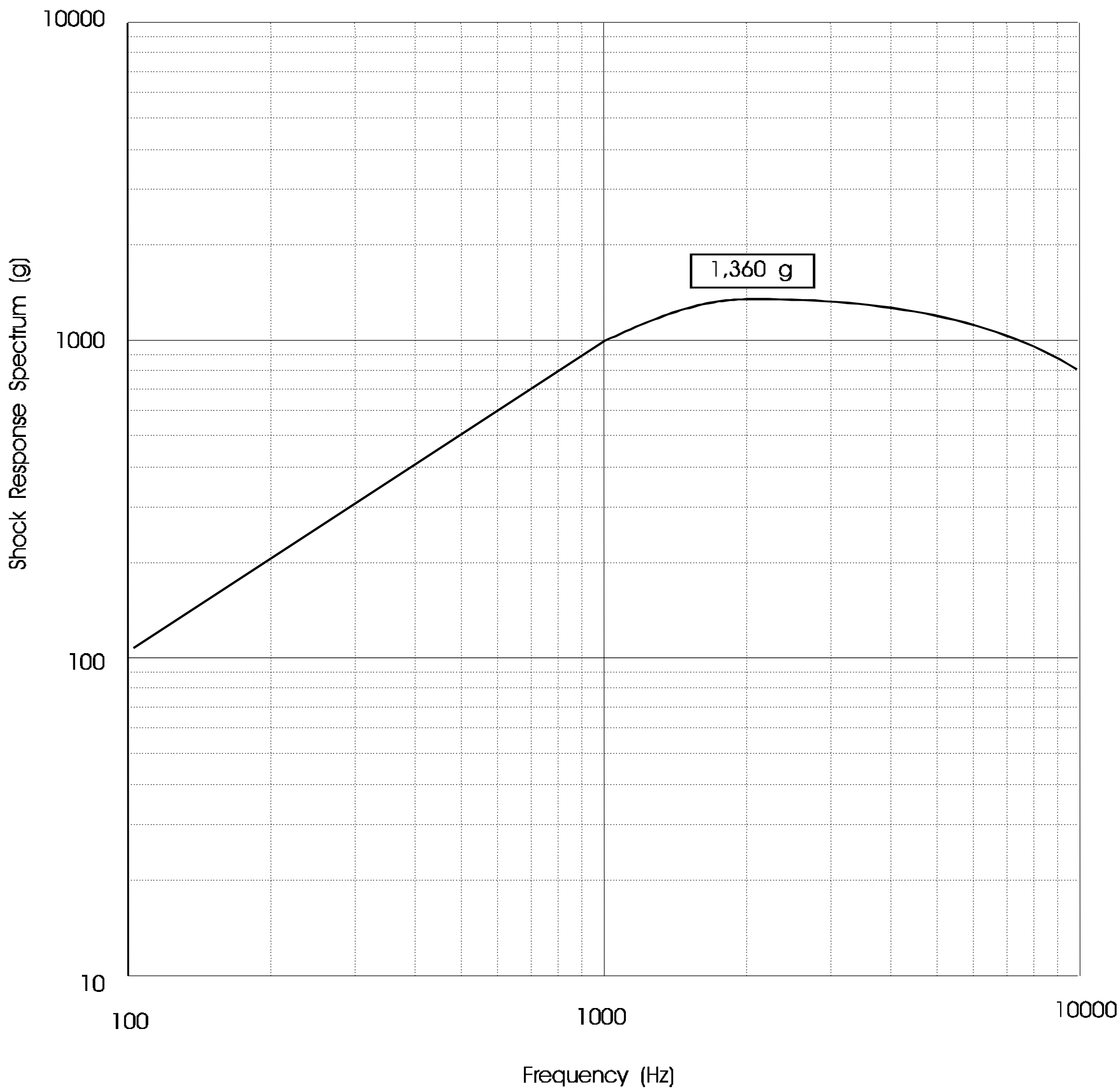


Figure E-3 Maximum Allowable Shock Response at Launch Vehicle/ Spacecraft Interface (VS 163) Produced by the TITAN IV or the Spacecraft for the NUS Configuration

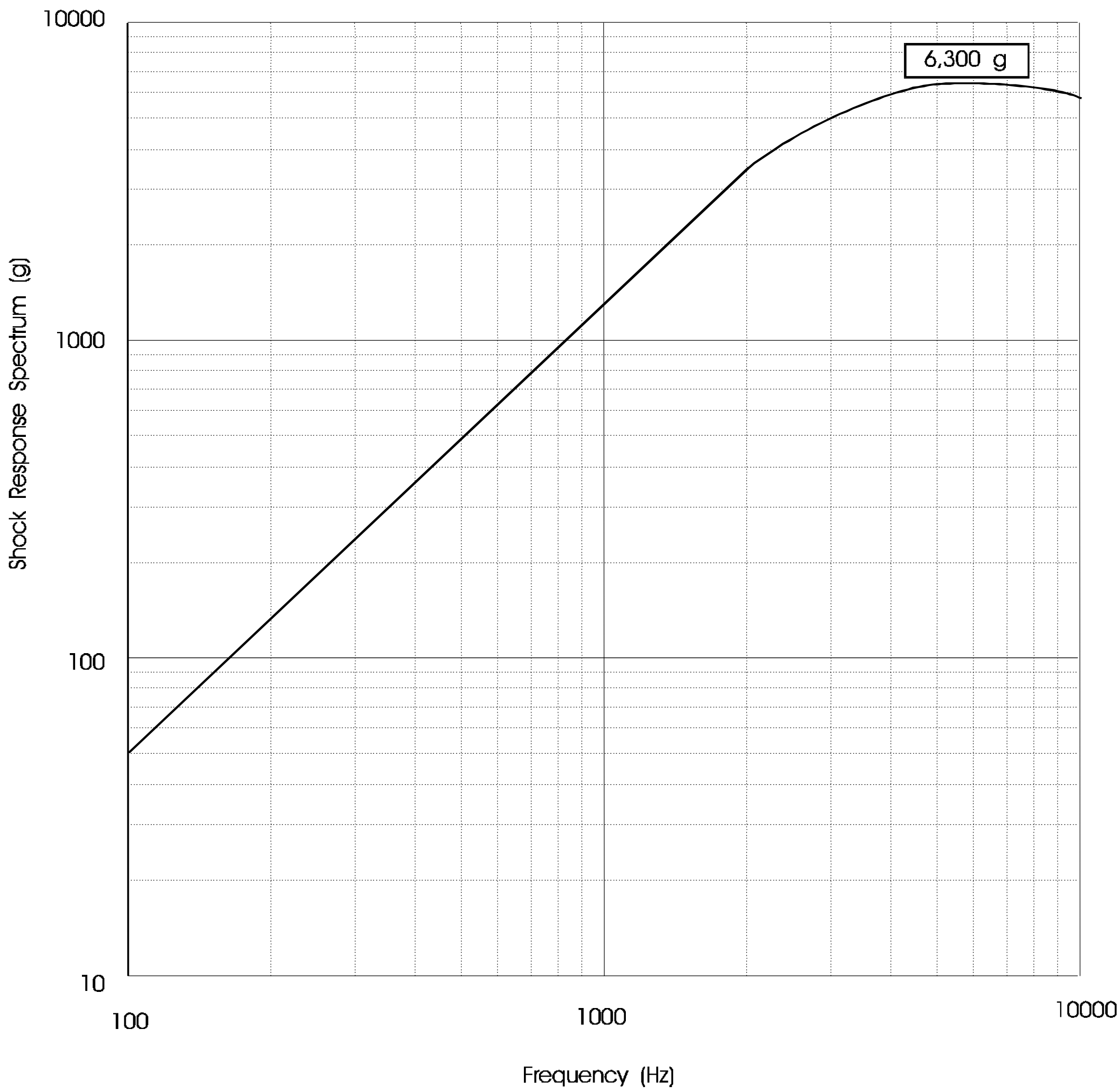


Figure E-4 Maximum Allowable Spacecraft Produced Interface Shock for Centaur Configuration (8- Hardpoint)

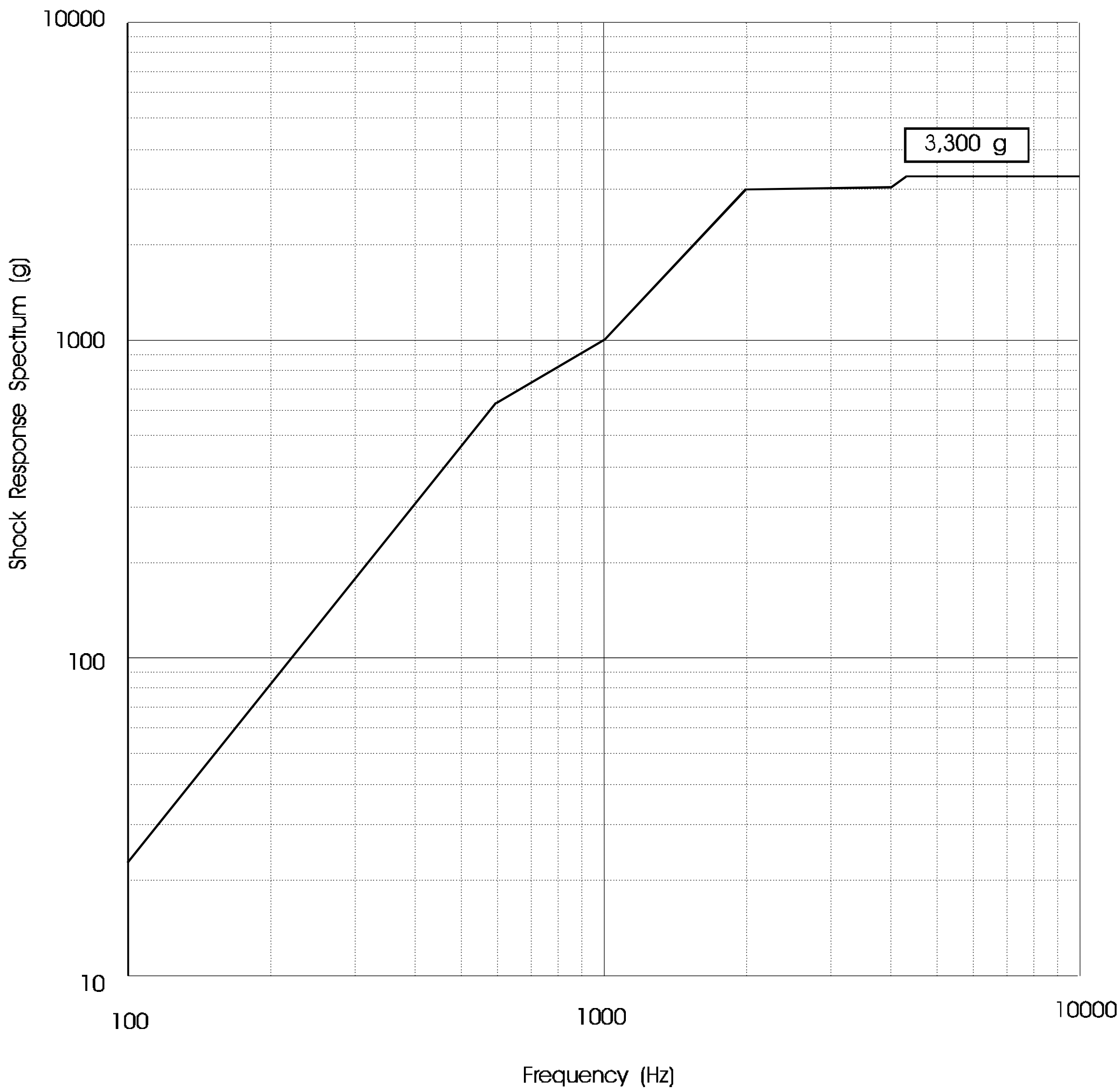


Figure E-5 Maximum Allowable Spacecraft Produced Interface Shock for Centaur Configuration (22- Hardpoint)

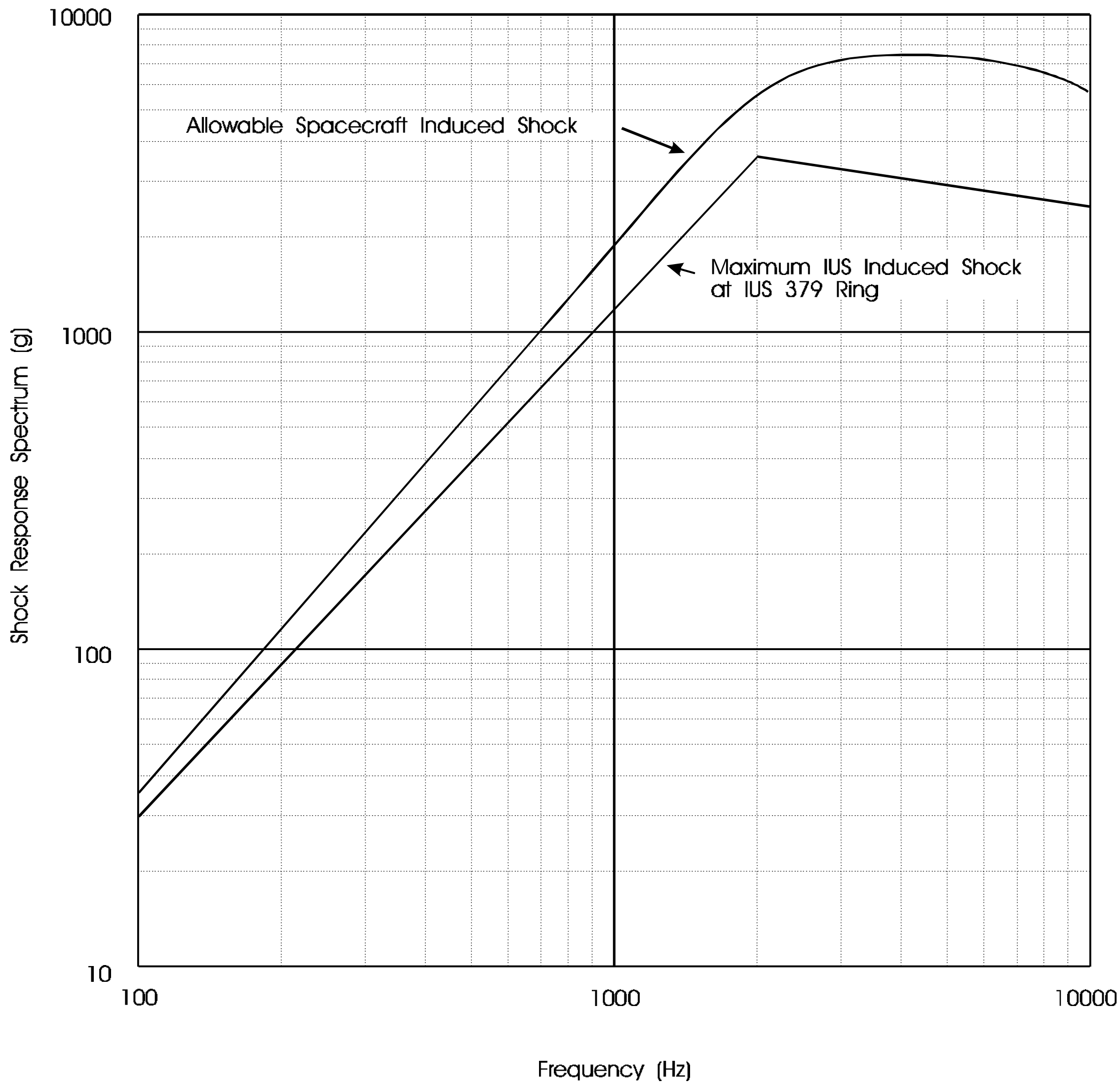


Figure E-6 Maximum IUS Induced and Maximum Allowable Spacecraft Induced Shock Levels at IUS-Spacecraft Interface

APPENDIX F

SCOUT

Structural Loads

For Scout the worst case loads must include a combination of steady-state acceleration with low frequency transient and vibroacoustic levels; spin forces are a major contributor to the limit loads to be applied and must be determined on a case-by-case basis. The maximum expected axial acceleration levels experienced during third- and fourth-stage thrusting are given in Table F-1. The maximum expected lateral acceleration is approximately 4g. The lateral acceleration levels determined by coupled loads are very dependent on the spacecraft configuration. The project should obtain a coupled loads analysis as early as possible to determine these levels.

Table F-1
SCOUT
Axial Acceleration

Payload Weight kg (lb)	3rd Stage (G)	4th Stage (G)
45.4 (100)	11.0	18.1
90.8 (200)	10.4	13.0
136.2 (300)	9.7	10.4
181.6 (400)	9.2	8.8
227 (500)	8.7	7.7

Acoustics

The qualification and acceptance acoustic noise levels are given in Table F-2.

Spacecraft Random Vibration

The spacecraft random vibration test levels are given in Table F-3.

Table F-2
SCOUT
Acoustic Test Levels
(Inside Payload Fairing)

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
50	124	121
63	124	121
80	124	121
100	124	121
125	124	121
160	124.5	121.5
200	125.5	122.5
250	126.5	123.5
315	127.5	124.5
400	129	126
500	129	126
630	129	126
800	128.5	125.5
1000	128	125
1250	127.5	124.5
1600	127	124
2000	126	123
2500	125	122
3150	124.5	121.5
4000	123	120
5000	122	119
6300	121.5	118.5
8000	121	118
10000	120.5	117.5
Overall	140	137

Table F-3
SCOUT
Spacecraft Random Vibration

Frequency (Hz)	ASD Level (G^2/Hz)	
	Qualification	Acceptance
0-100	.0028	.0014
100-500	+5 dB/oct	+5 dB/oct
500-2000	.04	.02
Overall	8.2 G_{rms}	5.8 G_{rms}

Mechanical Shock

There are two shock spectrum requirements for Scout payloads. Table F-4 give the spacecraft shock spectrum levels associated with higher frequency separation events, and Figures F-1 and F-2 give the qualification and acceptance levels associated with low-frequency transients. For the latter test, the spacecraft and adapter, attached to a vibrator, shall be excited by a complex acceleration transient; the positive and negative shock spectra of the applied transient shall match the reference levels within the allowed tolerance band for the three values of Q (damping) given at each one-third octave center frequency. The transient shall be applied one time in the thrust axis.

As an alternative to the low-frequency transient test, a three axis swept sine vibration test may be performed. The specifications for the sine test are given in Table F-5.

Table F-4
SCOUT
Shock Spectra
Q=10

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100	14	10
100-600	+10 dB/oct	+10 dB/0ct
600-2000	280	200

Table F-5
SCOUT
Sine Vibration
(Alternative for Transient)

Axis	Frequency (Hz)	Level (G_{peak})	
		Qualification	Acceptance
Thrust	5-43	24 cm/s (9.4 ips)	17.8 cm/s (7.0 ips)
	43-100	6.25	5.0
	100-200	5.0	4.0
Lateral	5-25	7.9 cm/s (3.1 ips)	6.4 cm/s (2.5 ips)
	25-80	1.25	1.0
	80-200	1.5	1.2
Sweep Rate		6 oct/min	

Balance - The acceptable launch mass unbalance of the payload with respect to the axis passing through the center of the support ring normal to the plane of the support ring is:

Dynamic unbalance: 36,500 gm-cm^2 (200 oz-in^2)

Static Unbalance: 800 gm-cm (12 oz-in).

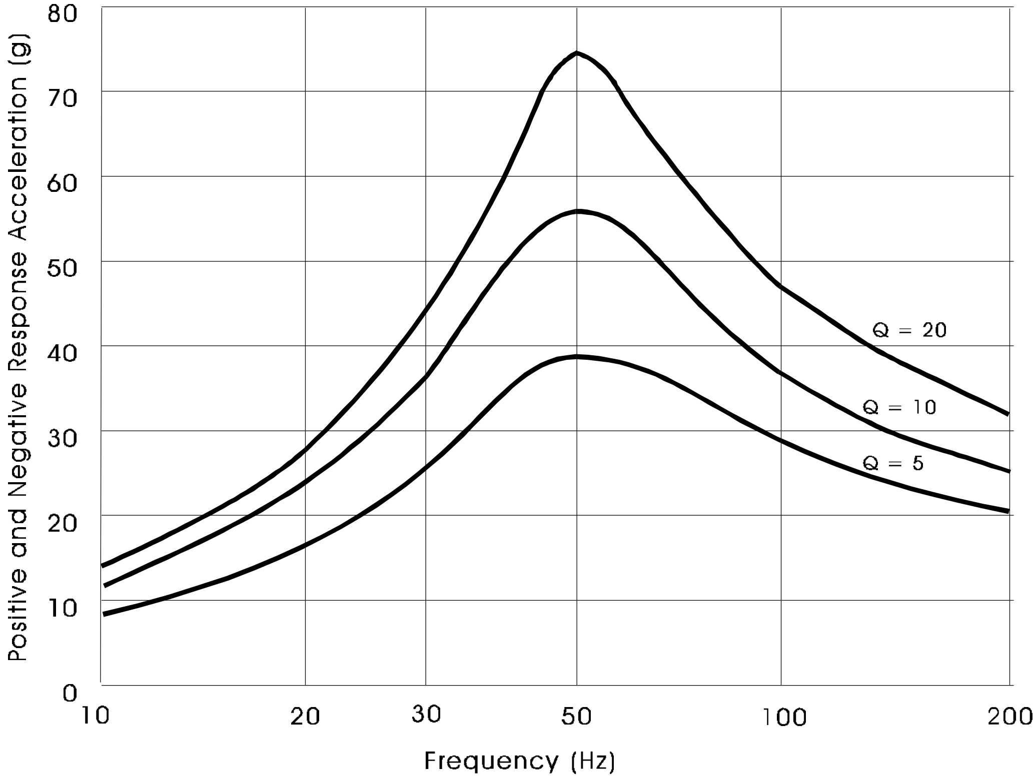


Figure F-1 Spacecraft Qualification Thrust Axis, Transient Shock Spectra

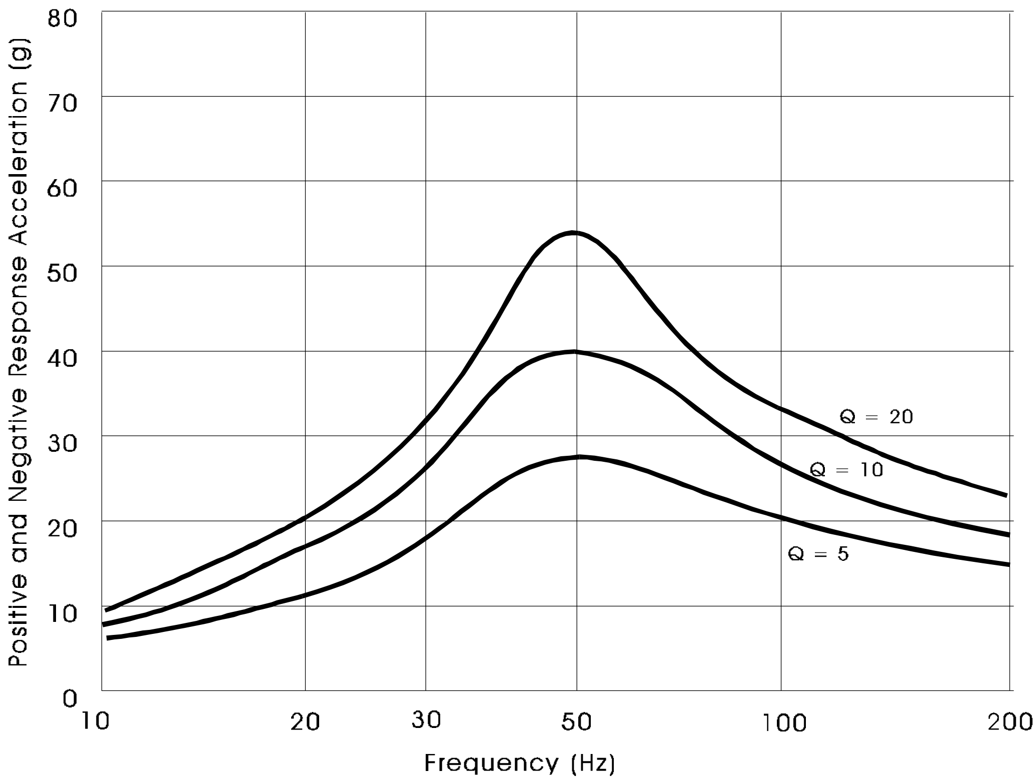


Figure F-2 Spacecraft Flight Acceptance Thrust Axis Transient Shock Spectra

APPENDIX G

PEGASUS

Structural Loads

Preliminary acceleration load levels are given in Table G-1.

Table G-1
PEGASUS
Acceleration Levels

Flight Mode	Translational (G)			Angular $(rad/sec)^2$		
	X	Y	Z	X	Y	Z
Captive Carry	.9 -.68	.82 -.92	3.5 -1.4	.74 -.74	2.1 -1.5	.57 -.57
Powered Flight	0 -8.5	.9 -.9	2.8 -3.3	.2 -.2	.2 -.2	.2 -.2

Acoustics

The maximum acoustic environment occurs during the take-off of the carrier aircraft. The qualification and acceptance test levels based on this environment are given in Table G-2. However, to comply with the NASA minimum vibroacoustic test level, the test level should be raised to 138 dB.

Spacecraft Random Vibration

The spacecraft random vibration levels are given in Table G-3.

Mechanical Shock - The pyro-shock response at the payload interface is defined by Table G-4.

Table G-2
Pegasus
Acoustic Test Levels
(Inside Payload Fairing)

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	112	109
32	112	109
40	112	109
50		
63	112	109
80	112	109
100	112	109
125	112	109
160	117	114
200	122	119
250	127	124
315	127	124
400	127	124
500	127	124
630	127	124
800	127	124
1000	127	124
1250	127	124
1600	124	121
2000	121	118
2500	118	115
3150	115	112
4000	112	109
5000	109	106
6300	106	103
8000	103	100
10000	100	97
Overall	137*	134*

* The minimum test level should be 138 dB to comply with NASA vibroacoustic test recommendations.

Table G-3
PEGASUS
Spacecraft Random Vibration

Frequency (Hz)	ASD Level (G2/Hz)	
	Qualification	Acceptance
20-165	.008	.004
165-200	+5 dB/oct	+5 dB/oct
200-800	.011	.0055
800-1000	+3 dB/oct	+3 dB/oct
1000-1300	.014	.007
1300-2000	-9 dB/oct	-9 dB/oct
Overall Level	4.5 G_{rms}	3.2 G_{rms}

Table G-4
PEGASUS
Separation Shock Response Spectrum
(Near Source)
Q=10

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100	70	50
100-1300	+10 dB/oct	+10 dB/oct
1300-10000	4900	3500
Overall Level	4.5 G_{rms}	3.2 G_{rms}

APPENDIX H

ARIANE

Structural Loads

For structural loads testing, limit loads shall be determined by combining the acceleration with low-frequency dynamic levels. Preliminary design limit loads are given in Table H-1.

To avoid dynamic coupling between the low-frequency vehicle and spacecraft modes, the spacecraft should be designed such that the fundamental lateral and axial frequencies are above 10 and 31 Hz respectively assuming that the spacecraft is hardmounted at the separation plane.

Table H-1
ARIANE 4
Limit Load Factor (G)
at Spacecraft C.G.

Flight Event	Quasi-Static Load (G)	
	Axial	Lateral
Maximum Dynamic Pressure	3.0	± 1.5
Before Thrust Termination	5.5	± 1.0
During Thrust Tail-off	-2.5 -4.5(1)	± 1.0

(1) For spacecraft having a mass < 1200 kg (2640-lb) and a longitudinal frequency > 40 Hz.

Note: Plus represents a compression load.

Acoustics

The acoustic specification levels are given in Table H-2.

Spacecraft Random Vibration

The maximum expected random vibration (limit levels) at the spacecraft interface is given in Table H-3.

Sine Vibration

A spacecraft sine vibration test is generally required by ARIANE. The test levels are given in Table H-4.

Mechanical Shock

The primary sources of shock are payload fairing separation and spacecraft separation. Table H-5 provides specification levels for the maximum expected levels due to launch vehicle induced separation events.

If the spacecraft is providing the separation system, the maximum allowable shock levels at the spacecraft interface plane are given in Table H-6.

Table H-2
ARIANE-4
Acoustic Test Levels*
Inside Payload Fairing

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	115	112
32	118	115
40	120	117
50	122	119
63	125	122
80	128	125
100	130	127
125	133	130
160	135	132
200	136	133
250	138	135
315	138	135
400	135	132
500	132	129
630	130	127
800	129	126
1000	126	123
1250	122	119
1600	122	119
2000	122	119
2500	122	119
3150	120	117
4000	118	115
5000	117	114
6300	116	113
8000	114	111
10000	113	110
Overall	145	142

* Test levels represent 1/3 octave band levels derived from Ariane 1/1 octave band levels

Table H-3
ARIANE-4
Spacecraft Random Vibration

Frequency (Hz)	ASD Level (G^2/Hz)
20	.0007
20-150	+6 dB/oct
150-700	.04
700-2000	-3 dB/oct
2000	.014
Overall Level	7.3 G_{rms}

Table H-4
ARIANE-4
Sine Vibration Requirements

Axis	Frequency (Hz)	Sine Vibration ($G_{0\text{-}p}$)	
		Qualification	Acceptance
Axial	5-6	1	1.0
	6-100	1.25	1.0
Lateral	5-18	1.0	0.8
	18-100	0.8	0.6

Table H-5
ARIANE 4
Maximum Launch Vehicle Generated Shock Levels
at Spacecraft Separation Plane
Shock Response Spectrum
Q=10

Payload Adapter	Frequency	Shock Response Spectrum (G)	
	(Hz)	Qualification	Acceptance
1194A, and 1194B	100 100-600 600 600-2150 2150-10000	28 15 dB/oct 2520 +4.8 dB/oct 7000	20 15 dB/oct 1800 +4.8 dB/oct 5000
937, 937A, 937C, and additional Cylindrical Adapter	100 100-1600 1600-10000	28 10 dB/oct 2800	20 10 dB/oct 2000
937B	100 100-2000 2000 2000-10000 10000	25 10.5 dB/oct 4620 1.6 dB/oct 7000	18 10.5 dB/oct 3300 1.6 dB/oct 5000
1666A, and 1666B	100 100-800 800 800-10000 10000	28 13.7 dB/oct 3220 1.3 dB/oct 5600	20 13.7 dB/oct 2300 1.3 dB/oct 4000

Table H-6
ARIANE 4
Maximum Allowable Spacecraft Generated Shock Levels
at Designated Launch Vehicle Bolted Interface Plane
Shock Response Spectrum
Q=10

Interface Plane	Frequency	Shock Response Spectrum (G)	
	(Hz)	Qualification	Acceptance
2624	100 100-1600 1600-10000	28 10 dB/oct 2800	20 10 dB/oct 2000
1920	100 100-3700 3700-10000	28 10 dB/oct 14000	20 10 dB/oct 10000

Structural Loads

For structural loads testing, limit loads shall be determined by combining the acceleration with low-frequency dynamic levels. Preliminary design limit loads are given in Table H-7.

To avoid dynamic coupling between the low-frequency vehicle and spacecraft modes, the spacecraft must be designed such that its lateral and axial fundamental frequencies are above minimum levels. These are dependent on the spacecraft, the adapter system used, and the spacecraft interface plane. These minimum frequencies, along with allowable C.G offsets and balance requirements must be negotiated on a case-by-case basis.

Table H-7
ARIANE 5
Limit Load Factor (G)
at Spacecraft C.G.

Flight Event	Quasi-Static Load (G)			
	Axial		Lateral	
	Static	Dynamic	Static	Dynamic
Lift-off	1.7	± 1.5	0	± 1.5
Maximum Dynamic Pressure	2.7	± 0.5	0	± 2
P 230 Burn-out	4.25	± 0.25	± 0.25	± 0.25
H 155 Burn-out	3.6	± 1.0	± 0.1	0
H 155 Thrust Tail-off	0.7	± 1.4	0	0

For a payload with a mass > 5,000 kg (11,000 lb.), the user should contact ARIANSPACE to obtain the appropriate load factors.

Acoustics

The acoustic specification levels are given in Table H-8.

Spacecraft Random Vibration

The maximum expected random vibration (limit levels) at the spacecraft interface is given in Table H-2..

Sine Vibration

A spacecraft sine vibration test is generally required by ARIANE. The test levels are given in Table H-9.

Mechanical Shock

The primary sources of shock are payload fairing separation and spacecraft separation. Table H-10 provides specification levels for the maximum expected levels due to launch vehicle induced separation events.

If the spacecraft is providing the separation system, the maximum allowable shock levels at the spacecraft interface plane are given in Table H-11.

Table H-8
ARIANE-5
Acoustic Test Levels
Inside Payload Fairing

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	119	116
32	122	119
40	124	121
50	125	122
63	128	125
80	131	128
100	132	129
125	133	130
160	135	132
200	136	133
250	138	135
315	138	135
400	134	131
500	132	129
630	130	127
800	129	126
1000	126	123
1250	122	119
1600	122	119
2000	122	119
2500	122	119
3150	120	117
4000	118	115
5000	117	114
6300	116	113
8000	114	111
10000	113	110
Overall	145	142

* Test levels represent 1/3 octave band levels derived from Ariane 1/1 octave band levels

Table H-9
ARIANE-5
Sine Vibration Requirements

Axis	Frequency (Hz)	Sine Vibration (G_{0-p})	
		Qualification	Acceptance
Axial	4-100	1.25	1.0
Lateral	2-18 18-100	1.0 0.8	0.8 0.6

Table H-10
ARIANE 5
Maximum Launch Vehicle Generated Shock Levels
at Spacecraft Separation Plane
Shock Response Spectrum
Q=10

Payload Adapter	Frequency (Hz)	Shock Response Spectrum (G) Qualification	Shock Response Spectrum (G) Acceptance
1194A	100	28	20
	100-600	15 dB/oct	15 dB/oct
	600	2520	1800
	600-2150	4.8 dB/oct	4.8 dB/oct
	2150-10000	7000	5000
937	100	28	20
	100-1600	10 dB/oct	10 dB/oct
	1600-10000	2800	2000
937B	100	25	18
	100-2000	10.5 dB/oct	10.5 dB/oct
	2000	4620	3300
	2000-10000	1.6 dB/oct	1.6 dB/oct
	10000	7000	5000
1666A	100	28	20
	100-800	13.7 dB/oct	13.7 dB/oct
	800	3220	2300
	800-10000	1.3 dB/oct	1.3 dB/oct
	10000	5600	4000
Separable 1920	100	28	20
	100-700	+13.7 dB/oct	+13.7 dB/oct
	700	2380	1700
	700-3350	+6.8 dB/oct	+6.8 dB/oct
	3350-10000	14000	10000

Table H-11
ARIANE 5
Maximum Allowable Spacecraft Generated Shock Levels
at Designated Launch Vehicle Bolted Interface Plane
Shock Response Spectrum
Q=10

Launch Vehicle Bolted Interface	Frequency (Hz)	Shock Response Spectrum (G)	
		Qualification	Acceptance
1920, 2624, and 3936	100 100-700 700 700-3350 3350-10000	28 13.7 dB/oct 2380 6.8 dB/oct 14000	20 13.7 dB/oct 1800 6.8 dB/oct 10000

APPENDIX I

TAURUS

Structural Loads

Table I-1 specifies the maximum quasi-static limit load factors and angular accelerations expected at the payload interface during various mission phases for both three and four stage Taurus vehicles. These lateral load factors and angular accelerations are generated by motor firing and thrust vectoring. Longitudinal load factors are governed by payload mass and upper stage configuration as shown in Figure I-1 for the Taurus vehicle, and Figure I-2 for the Taurus XL and XLS vehicles

Table I-1
Taurus
Quasi-Static Limit Load Factors and Angular Accelerations
(At Spacecraft Interface Plane)

Mission Segment	Thrust X (g)	Lateral Y (g)	Vertical Z (g)	Roll (rad/s^2)	Pitch (rad/s^2)	Yaw (rad/s^2)
Ground Operations	± 1.5	± 1.7	± 1.7	-	-	-
Flight Operations	Figure I-1,2	± 0.5	± 0.5	± 0.035	± 0.15	± 0.15
On-Orbit Operations	± 0.02	± 0.02	± 0.02	± 0.7	± 0.2	± 0.2

Acoustics

The qualification and acceptance acoustic test levels are given in Table I-2 for the Taurus and Taurus XL vehicles. These levels assume use of the standard Taurus fairing acoustic blanket and a full fairing (i.e., maximum size payload. For the Taurus XLS vehicle, add 2 dB to all levels.

Spacecraft Random Vibration

The random vibration input to the spacecraft and its components is highly sensitive to spacecraft mass and structural configuration. Table I-3 provides the maximum expected random vibration levels at the payload interface ring for a typical 1135 kg (2500 lb) payload on a four stage Taurus or Taurus XL vehicle. Table I-4 provides the corresponding levels for a Taurus XL/S vehicle.

Shock

The maximum expected, launch vehicle produced, shock spectrum at the payload interface is provided in Table I-5.

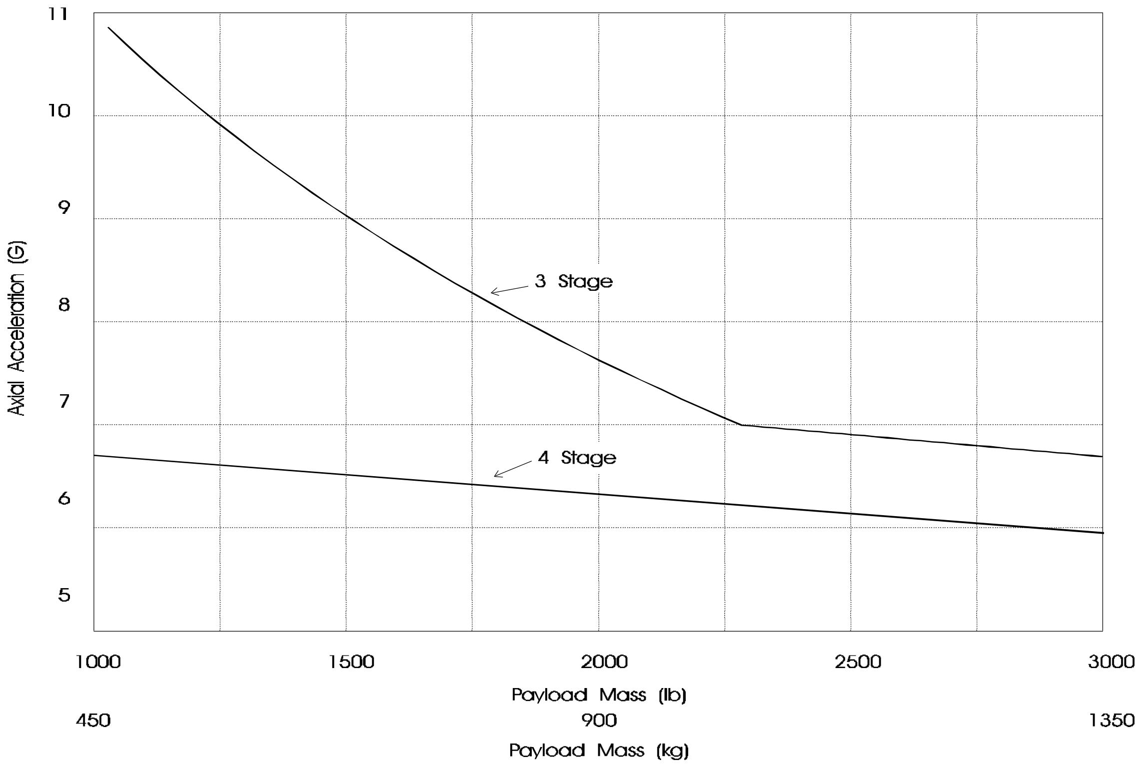


Figure I-1 Taurus Axial Acceleration Loads vs. Payload Mass

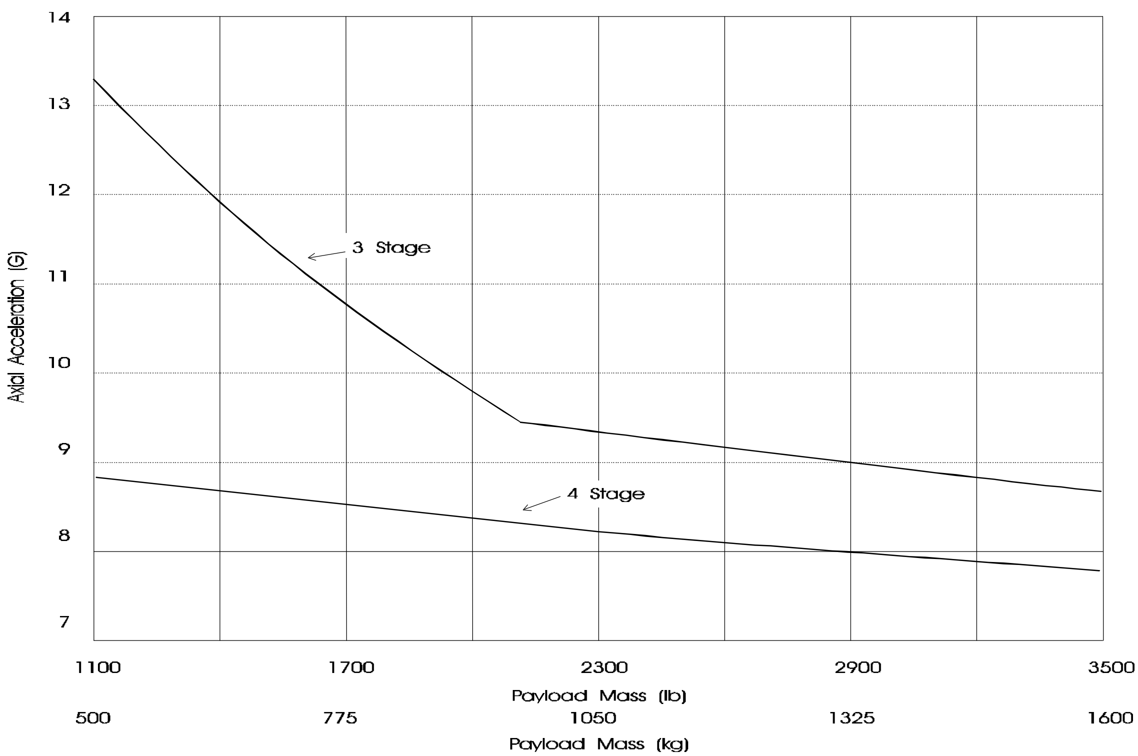


Figure I-2 Taurus XL and XL/S Axial Acceleration Loads vs. Payload Mass

Table I-2
Taurus
Acoustic Test Levels
(Inside Payload Fairing)

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	120.7	117.7
32	123.1	120.1
40	125.3	122.3
50	127.2	124.2
63	128.8	125.8
80	130.9	127.9
100	130.6	127.6
125	130.9	127.9
160	131.4	128.4
200	133.7	130.7
250	135.2	132.2
315	134.0	131.0
400	135.6	132.6
500	136.1	133.1
630	132.9	129.9
800	129.8	126.8
1000	127.7	124.7
1250	125.1	122.1
1600	120.7	117.7
2000	122.4	119.4
2500	116.9	113.9
3150	109.8	106.8
4000	106.8	103.8
5000	105.1	102.1
6300	103.0	100.0
8000	100.7	97.7
10000	99.4	96.4
Overall	144.3	141.3

Table I-3
TAURUS and TAURUS XL
Maximum Expected Random Vibration Level at
Spacecraft Interface

Frequency (Hz)	ASD Level (G^2/Hz)
20	.005
20-100	+5 dB/oct
100-125	.07
125-200	-3.5 dB/oct
200-500	.04
500-1250	-10 dB/oct
1250-2000	.002
Overall Level	5.4 G_{rms}

Table I-4
TAURUS XL/S
Maximum Expected Random Vibration Level at
Spacecraft Interface

Frequency (Hz)	ASD Level (G^2/Hz)
20	.007
20-100	+5 dB/oct
100-125	.1
125-200	-3.5 dB/oct
200-500	.057
500-1250	-10 dB/oct
1250-2000	.0028
Overall Level	6.5 G_{rms}

Table I-5
TAURUS
Launch Vehicle Induced
Pyro Shock Response Spectrum
at Payload Interface
Q=10

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100	140	100
100-2000	+7.4 dB/oct	+7.4 dB/oct
2000-10000	5600	4000

APPENDIX J

CONESTOGA

Structural Loads

Table J-1 provides worst case limit load factors for various launch vehicle configurations. These values are usable for preliminary design.

Table J-1
Conestoga
Quasi-Static Limit Load Factors

Model	1229	1379	1620	1669	1679	3632
No. of Motors	3	4	7	7	7	7
Type of Motor	Castor IV's	Castor IV's	Castor IV's	Castor IV's	Castor IV's	Castor IV -XL's
Load Factor (g)						
Axial Compression	9.0	11.0	11.0	10.5	9.0	10.2
Lateral (any direction)	1.5	2.0	2.0	1.8	1.5	1.5

Acoustics

The qualification and acceptance acoustic test levels are given in Table J-2 for the Conestoga vehicles. These levels assume use of a 10 cm (4 in.) acoustic blanket and no payload fill effects.

Spacecraft Random Vibration

The maximum expected random vibration input at the spacecraft interface is given in Table J-3.

Mechanical Shock

Test levels representing launch vehicle induced shock levels at the payload separation plane are given in Table J-4.

Table J-2
Conestoga
Acoustic Test Levels
Inside Empty Payload Fairing
with 10 cm (4 in.) Acoustic Blanket

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	103	100
32	106	103
40	108	105
50	111.5	108.5
63	116	113
80	119.5	116.5
100	122.5	119.5
125	125	122
160	127.5	124.5
200	129	126
250	130.5	127.5
315	131	128
400	131.5	128.5
500	131.5	128.5
630	131	128
800	130.5	127.5
1000	129	126
1250	127.5	124.5
1600	126	123
2000	124	121
2500	121.5	118.5
3150	119.5	116.5
4000	116.5	113.5
5000	114.5	111.5
6300	112	109
8000	109	106
10000	107	104
Overall	141	138

Table J-3
CONESTOGA
Spacecraft Random Vibration

Frequency (Hz)	ASD Level (G^2/Hz)	
	Qualification	Acceptance
20 20-90 90-500 500-2000 2000	.0038 +4 dB/oct .028 -6 dB/oct .0018	.0019 +4 dB/oct .014 -6 dB/oct .00088
Overall Level	4.8 G_{rms}	3.4 G_{rms}

Table J-4
CONESTOGA
Launch Vehicle Induced
Shock Response Spectrum
Q=10

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100-350	56	40
350-500	+8 dB/oct	+8 dB/oct
1000-4000	2800	2000

APPENDIX K

H II

Structural Loads

Table K-1 provides quasi-static limit load factors usable for preliminary design.

To avoid dynamic coupling between the spacecraft and launch vehicle, the first axial and lateral spacecraft fixed base modes should be greater than 30 Hz and 10 Hz respectively.

Table K-1
H II
Quasi-Static Limit Load Factors
Spacecraft 3500 kg (7700-lb)

Event	Axial (G)	Lateral (G)
Lift-Off	3.2 max 1.7 static + 1.5 dynamic 0.3 min 1.5 static - 1.2 dynamic	± 2.0
MECO	4.0 ± 1.0	± 0.8

Acoustics

The qualification and acceptance acoustic test levels are given in Table K-2 for the H II vehicle.

Spacecraft Sine Vibration

The maximum expected, 3 sigma, sinusoidal vibration level at the spacecraft interface is given in Table K-3.

Mechanical Shock

Test levels representing spacecraft separation are given in Tables K-4.

Table K-2
H II
Acoustic Test Levels
(Inside Empty Payload Fairing)

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	121	118
32	123	120
40	125	122
50	127	124
63	128.5	125.5
80	130	127
100	131.5	128.5
125	133	130
160	134	131
200	135	132
250	135	132
315	134	131
400	133	130
500	132	129
630	131	128
800	130	127
1000	129	126
1250	127.5	124.5
1600	126	123
2000	124	121
2500	122	119
3150	120	117
4000	118	115
5000	116	113
6300	114	111
8000	112	109
10000	110	107
Overall	144	141

Table K-3
H II
Spacecraft Sine Vibration

Frequency (Hz)	Level ($G_{0\text{-}p}$)	
	Thrust	Lateral
5-100	1.0	0.8
Sweep Rate*	4 oct/min	4 oct/min

* Sweep rates recommended by NASDA.

Table K-4
H II
Spacecraft Separation
Shock Response Spectrum
Q=10

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100	56	40
100-1500	56	56
1500-3000	5800	4100

APPENDIX L

LOCKHEED MARTIN LAUNCH VEHICLE (LMLV)

Structural Loads

Table L-1 provides limit load factors for the LMLV1. LMLV2, and LMLV3 configurations. These values are usable for preliminary design.

Table L-1
LMLV
Quasi-Static Limit Load Factors[1]

Flight Event	LMLV1		LMLV2		LMLV3	
	Axial[2]	Lateral	Axial[2]	Lateral	Axial[2]	Lateral
Launch/First Stage Ignition	-4.0/+6.0	± 2.0	-1.0/+3.0	± 1.5	-1.0/+3.0	± 1.5
First Stage Motor Resonance	3.5 ± 2.0	±2.0	2.0 ± 1.0	± 1.5	2.0 ± 1.0	± 1.5
Wind Gust	3.5	±2.5	2.0	±2.5	2.0	±2.5
First Stage Maximum Acceleration	8.0	±2.0	2.0	±2.0	4.5	±2.0
Second Stage Ignition	~	~	-1.0/+6.0	± 1.5	-1.0/+6.0	± 1.5
Second Stage Motor Resonance	~	~	4.0 ± 3.0	± 2.5	4.0 ± 3.0	± 2.5
Second Stage Maximum Acceleration	~	~	8.0	± 2.0	6.0	± 2.0
ORBUS	-2.0/+5.0	±2.0	-2.0/+5.0	± 2.0	-2.0/+5.0	± 2.0
ORBUS	7.0	±1.0	7.0	± 1.0	7.0	± 2.0
Envelope	-4.0/+8.0	±2.5	-2.0/+8.0	± 2.5	-2.0/+7.0	± 2.5

1 - 99th percentile.

2 - Positive axial load factor acts aft at spacecraft c.g.

Acoustics

The qualification and acceptance acoustic test levels are given in Tables L-2 through L-4 for the LMLV1, LMLV2, and LMLV3 vehicles. These levels are for empty payload fairings without acoustic blankets. The payload fairings are as noted in the Tables.

Spacecraft Random Vibration

The maximum expected random vibration input at the spacecraft interface for the LMLV1, LMLV2, and LMLV3 are given in Tables L-5, L-6, and L-7 respectively.

Mechanical Shock

Test levels representing launch vehicle induced shock levels at the payload separation plane are given in Table L-8.

Table L-2
LMLV1
Acoustic Test Levels
Inside Empty Model 92 Aluminum Payload Fairing

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	120	117
32	121	118
40	122	119
50	122	119
63	122	119
80	122	119
100	122	119
125	122	119
160	122	119
200	123	120
250	123	120
315	123	120
400	124	121
500	128	125
630	126	123
800	120	117
1000	119	116
1250	120	117
1600	121	118
2000	126	123
2500	120	117
3150	117	114
4000	115	112
5000	112	109
6300	109	106
8000	105	102
10000	103	100
Overall	136	133

* The minimum test level should be 138 dB to comply with NASA vibroacoustic test recommendations.

Table L-3
LMLV2 Acoustic Test Levels
Inside Empty Model 120 Aluminum Payload Fairing

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	119	116
32	119	116
40	120	117
50	120	117
63	120	117
80	120	117
100	121	118
125	120	117
160	119	116
200	119	116
250	119	116
315	120	117
400	122	119
500	127	124
630	119	116
800	118	115
1000	115	112
1250	114	111
1600	114	111
2000	119	116
2500	115	112
3150	116	113
4000	111	108
5000	107	104
6300	105	102
8000	102	99
10000	98	95
Overall	133	130

* The minimum test level should be 138 dB to comply with NASA vibroacoustic test recommendations.

Table L-4
LMLV3 Acoustic Test Levels
Inside Empty Model 141 Aluminum Payload Fairing

One-Third Octave Center Frequency (Hz)	Noise Level (dB) re: .00002 Pa	
	Qualification	Acceptance
25	121	118
32	121	118
40	121	118
50	122	119
63	123	120
80	124	121
100	125	122
125	126	123
160	127	124
200	128	125
250	128	125
315	129	126
400	129	126
500	129	126
630	130	127
800	130	127
1000	129	126
1250	127	124
1600	124	121
2000	135	132
2500	128	125
3150	124	121
4000	121	118
5000	118	115
6300	115	112
8000	114	111
10000	112	109
Overall	141	138

Table L-5
LMLV1
Spacecraft Random Vibration

Frequency (Hz)	ASD Level (G^2/Hz)	
	Qualification	Acceptance
20 20-500 500 500-2000 2000	.0038 +2.5 dB/oct .015 0 dB/oct .0015	.0001 +2.5 dB/oct .015 0 dB/oct .0015
Overall Level	7.2 G_{rms}	5.1 G_{rms}

Table L-6
LMLV2
Spacecraft Random Vibration

Frequency (Hz)	ASD Level (G^2/Hz)	
	Qualification	Acceptance
20 20-600 600 600-1000 1000 1000-2000 2000	.0028 +0.8 dB/oct .07 0 dB/oct ..002 -10 dB/oct .007	.002 +0.8 dB/oct .05 0 dB/oct .0015 -10 dB/oct .005
Overall Level	10.9 G_{rms}	7.8 G_{rms}

Table L-7
LMLV3
Spacecraft Random Vibration

Frequency (Hz)	ASD Level (G^2/Hz)	
	Qualification	Acceptance
20	.07	.05
20-600	+0.9 dB/oct	+0.9 dB/oct
600	.2	.14
600-1000	0 dB/oct	0 dB/oct
1000	.2	.14
1000-2000	-11.5 dB/oct	-11.5 dB/oct
2000	.014	.01
Overall Level	17.8 G_{rms}	12.7 G_{rms}

Table L-8
LMLV
Launch Vehicle Induced
Shock Response Spectrum
at Spacecraft Interface
Q=10

Frequency (Hz)	Shock Response Spectrum (G)	
	Qualification	Acceptance
100	35	25
100-600	10.7 dB/oct	10.7 dB/oct
600	840	600
600-1000	4.8 dB/oct	4.8 dB/oct
1000	1260	900
1000-1500	8.5 dB/oct	8.5 dB/oct
1500	2240	1600
1500-2000	0 dB/oct	0 dB/oct
2000	2260	1600