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GENERAL ENVIRONMENTAL VERIFICATION STANDARD (GEVS)  
For GSFC Flight Programs and Projects

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Changes to  
 GENERAL ENVIRONMENTAL VERIFICATION STANDARD

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Baseline	April 2005	
A	4/22/2013	Major update - STS references removed; Reference to Office of Mission Success removed; Added definition of anomaly; Significant updates to the following sections: 1.13; 2.2.5; 2.4.1.2; 2.4.1.4.1; 2.4.2.2; 2.4.2.6; 2.4.4.1; 2.4.4.2; 2.6.2.4; Section 1.14 added; Section 2.5 rewritten; Section 2.7 rewritten CCR Number CCR-D-0071

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## TABLE OF CONTENTS

Paragraph		Page
<b>SECTION 1 -- GENERAL INFORMATION</b>		
1.1	PURPOSE .....	1-1
1.2	APPLICABILITY AND LIMITATIONS .....	1-1
1.3	THE GSFC VERIFICATION APPROACH.....	1-1
1.4	OTHER ASSURANCE REQUIREMENTS .....	1-2
1.5	RESPONSIBILITY FOR ADMINISTRATION .....	1-2
1.6	GEVS CONFIGURATION CONTROL AND DISTRIBUTION .....	1-2
1.7	APPLICABLE DOCUMENTS .....	1-2
1.7.1	Spacecraft Tracking and Data Network Simulation .....	1-2
1.7.2	Deep Space Network (DSN) Simulation .....	1-3
1.7.3	NASA Standards .....	1-3
1.7.4	Military Standards for EMC Testing .....	1-3
1.7.5	Military Standards for Non-Destructive Evaluation .....	1-3
1.8	DEFINITIONS.....	1-3
1.9	CRITERIA FOR UNSATISFACTORY PERFORMANCE .....	1-8
1.9.1	Failure Occurrence .....	1-8
1.9.2	Failures with Retroactive Effect.....	1-8
1.9.3	Failure Reporting.....	1-8
1.9.4	Wear Out .....	1-8
1.10	TEST SAFETY RESPONSIBILITIES .....	1-8
1.10.1	Operations Hazard Analysis, Responsibilities For .....	1-9
1.10.2	Treatment of Hazards.....	1-9
1.10.3	Facility Safety .....	1-9
1.10.4	Safety Responsibilities During Test .....	1-9
1.11	TESTING OF SPARE HARDWARE .....	1-9
1.12	TEST FACILITIES, CALIBRATION.....	1-10
1.13	TEST CONDITION TOLERANCES .....	1-10
1.14	TEST MEASUREMENT CONSIDERATIONS .....	1-12
<b>SECTION 2 -- VERIFICATION PROGRAM</b>		
<b>SECTION 2.1 - SYSTEM PERFORMANCE VERIFICATION</b>		
2.1	SYSTEM PERFORMANCE VERIFICATION .....	2.1-1
2.1.1	Documentation Requirements.....	2.1-1
2.1.1.1	System Performance Verification Plan.....	2.1-1
2.1.1.1.1	Environmental Verification Plan .....	2.1-1
2.1.1.2	System Performance Verification Matrix .....	2.1-2
2.1.1.2.1	Environmental Test Matrix .....	2.1-2
2.1.1.3	Environmental Verification Specification .....	2.1-3
2.1.1.4	Performance Verification Procedures .....	2.1-3
2.1.1.5	Verification Reports .....	2.1-3
2.1.1.6	System Performance Verification Report .....	2.1-3
2.1.1.7	Instrument Verification Documentation .....	2.1-4

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**SECTION 2.2 - ENVIRONMENTAL VERIFICATION**

2.2	APPLICABILITY .....	2.2-1
2.2.1	Test Sequence and Level of Assembly .....	2.2-1
2.2.2	Verification Program Tailoring .....	2.2-1
2.2.3	Qualification if Hardware by Similarity .....	2.2-3
2.2.4	Test Factors/Durations .....	2.2-4
2.2.5	Structural Analysis/Design Factors of Safety .....	2.2-4

**SECTION 2.3 - ELECTRICAL FUNCTION & PERFORMANCE**

2.3	ELECTRICAL FUNCTION TEST REQUIREMENTS .....	2.3-1
2.3.1	Electrical Interface Tests .....	2.3-1
2.3.2	Comprehensive Performance Tests .....	2.3-1
2.3.3	Limited Performance Tests .....	2.3-1
2.3.4	Performance Operating Time and Failure-Free Performance Testing .....	2.3-2
2.3.5	Limited-Life Electrical Elements .....	2.3-2

**SECTION 2.4 - STRUCTURAL AND MECHANICAL**

2.4	STRUCTURAL AND MECHANICAL VERIFICATION REQUIREMENTS .....	2.4-1
2.4.1	Structural Loads Qualification .....	2.4-1
2.4.1.1	Coupled Loads Analysis .....	2.4-4
2.4.1.1.1	Analysis-Strength Qualification .....	2.4-4
2.4.1.1.2	Analysis-Clearance Verification .....	2.4-5
2.4.1.2	Modal Survey .....	2.4-6
2.4.1.3	Design Strength Qualification .....	2.4-7
2.4.1.3.1	Strength Qualification - Beryllium .....	2.4-8
2.4.1.4	Structural Reliability (Residual Strength Qualification) .....	2.4-9
2.4.1.4.1	Primary and Secondary Structure .....	2.4-9
2.4.1.5	Acceptance Requirements .....	2.4-11
2.4.2	Vibroacoustic Qualification .....	2.4-12
2.4.2.1	Fatigue Life Considerations .....	2.4-13
2.4.2.2	Payload Acoustic Test .....	2.4-13
2.4.2.3	Payload Random Vibration Tests .....	2.4-14
2.4.2.4	Subsystem/Instrument Vibroacoustic Tests .....	2.4-15
2.4.2.5	Component/Unit Vibroacoustic Tests .....	2.4-15
2.4.2.6	Acceptance Requirements .....	2.4-17
2.4.2.7	Retest of Reflight Hardware .....	2.4-20
2.4.2.8	Retest of Reworked Hardware .....	2.4-20
2.4.3	Sinusoidal Sweep Vibration Qualification .....	2.4-20
2.4.3.1	ELV Payload Sine Sweep Vibration Tests .....	2.4-21
2.4.3.2	ELV Payload Subsystem (including Instruments) and Component Sine Sweep Vibration Tests .....	2.4-22
2.4.3.3	Acceptance Requirements .....	2.4-24
2.4.4	Mechanical Shock Qualification .....	2.4-24
2.4.4.1	Subsystem Mechanical Shock Tests .....	2.4-24

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Paragraph		Page
2.4.4.2	Payload (Spacecraft) Mechanical Shock Tests .....	2.4-26
2.4.4.3	Acceptance Requirements .....	2.4-27
2.4.5	Mechanical Function Verification .....	2.4-27
2.4.5.1	Life Testing .....	2.4-27
2.4.5.2	Demonstration .....	2.4-30
2.4.5.3	Torque/Force Margin .....	2.4-34
2.4.5.4	Acceptance Requirements .....	2.4-35
2.4.6	Pressure Profile Qualification .....	2.4-36
2.4.6.1	Demonstration .....	2.4-36
2.4.6.2	Acceptance Requirements .....	2.4-36
2.4.7	Mass Properties Verification .....	2.4-36
2.4.7.1	Demonstration .....	2.4-36
2.4.7.2	Acceptance Requirements .....	2.4-38

## SECTION 2.5 - EMC

2.5	ELECTROMAGNETIC COMPATIBILITY (EMC) REQUIREMENTS .....	2.5-1
2.5.1	General Requirements .....	2.5-1
2.5.1.1	Documentation .....	2.5-2
2.5.1.2	Requirements Overview .....	2.5-3
2.5.1.3	Testing at Lower Levels of Assembly.....	2.5-4
2.5.1.4	Safety and Controls .....	2.5-4
2.5.1.5	Test Facility and Procedure Requirements .....	2.5-5
2.5.1.5.1	Ambient Electromagnetic Level.....	2.5-5
2.5.1.5.2	Power Source Impedance .....	2.5-5
2.5.1.5.3	Construction and Arrangement of EUT Cables.....	2.5-6
2.5.1.5.4	Susceptibility Testing.....	2.5-7
2.5.1.5.4.1	Frequency Scanning .....	2.5-7
2.5.1.5.4.2	Thresholds of Susceptibility .....	2.5-8
2.5.1.5.5	General Test Setup .....	2.5-9
2.5.1.6	Documentation Guidelines .....	2.5-10
2.5.1.6.1	EMC Control Plan (EMCCP) .....	2.5-10
2.5.1.6.2	EMI Test Procedure (EMITP) .....	2.5-11
2.5.1.6.3	EMI Test Report (EMITR) .....	2.5-13
2.5.1.6.4	Waiver Requests .....	2.5-15
2.5.2	Detailed Requirements.....	2.5-16
2.5.2.1	Conducted Emissions Requirements .....	2.5-17
2.5.2.1.1	Conducted Emissions, Power Leads, Differential Mode .....	2.5-17
2.5.2.1.2	Conducted Emissions, Common Mode, Power and Signal Lines.....	2.5-19
2.5.2.1.3	Conducted Emissions, Time Domain, Transients .....	2.5-21
2.5.2.1.4	Conducted Emissions, Antenna Terminal .....	2.5-24
2.5.2.2	Conducted Susceptibility Requirements .....	2.5-25
2.5.2.2.1	Conducted Susceptibility, Power Leads.....	2.5-25
2.5.2.2.2	Conducted Susceptibility, Antenna Terminals .....	2.5-27
2.5.2.2.3	Conducted Susceptibility, Transients, Power Leads.....	2.5-27
2.5.2.2.4	Conducted Susceptibility, Bulk Cable Injection .....	2.5-30
2.5.2.2.4.1	Conducted Susceptibility, Power and Signal Cables, Common Mode .....	2.5-30
2.5.2.2.4.2	Conducted Susceptibility, Power Leads.....	2.5-31

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Paragraph		Page
2.5.2.2.5	Conducted Susceptibility, Bulk Cable Injection, Impulse Excitation .....	2.5-33
2.5.2.3	Radiated Emissions Requirements .....	2.5-34
2.5.2.3.1	Radiated Emissions, Magnetic Field .....	2.5-34
2.5.2.3.2	Radiated Emissions, Electric Field .....	2.5-34
2.5.2.4	Radiated Susceptibility Requirements .....	2.5-35
2.5.2.4.1	Radiated Susceptibility, Magnetic Field .....	2.5-35
2.5.2.4.2	Radiated Susceptibility, Electric Field .....	2.5-36
2.5.3	Application Guide .....	2.5-37
2.5.3.1	Engineering Concerns for EMI Testing .....	2.5-37
2.5.3.2	Test Facility and Procedure Requirements .....	2.5-39
2.5.3.2.1	Ambient Electromagnetic Level.....	2.5-39
2.5.3.2.2	Power Source Impedance .....	2.5-42
2.5.3.2.3	Construction and Arrangement of EUT Cables.....	2.5-45
2.5.3.3	Detailed Requirements.....	2.5-46
2.5.3.3.1	Conducted Emissions, Power Leads, Differential Mode .....	2.5-46
2.5.3.3.2	Conducted Emissions, Common Mode, Power and Signal Lines.....	2.5-47
2.5.3.3.3	Conducted Emissions, Time Domain, Transients .....	2.5-51
2.5.3.3.4	Conducted Susceptibility, Power Leads.....	2.5-57
2.5.3.3.5	Conducted Susceptibility, Transients, Power Leads.....	2.5-58
2.5.3.3.6	Conducted Susceptibility, Bulk Cable Injection.....	2.5-59
2.5.3.3.6.1	Conducted Susceptibility, Power and Signal Cables, Common Mode .....	2.5-59
2.5.3.3.6.2	Conducted Susceptibility, Power Leads.....	2.5-60
2.5.3.3.7	Radiated Emissions, Electric Field.....	2.5-61
<b>SECTION 2.6 - THERMAL</b>		
2.6	VACUUM, THERMAL, AND HUMIDITY VERIFICATION REQUIREMENTS .....	2.6-1
2.6.1	Summary of Requirements .....	2.6-1
2.6.2	Thermal-Vacuum Qualification .....	2.6-1
2.6.2.1	Applicability .....	2.6-4
2.6.2.2	Special Considerations.....	2.6-4
2.6.2.3	Level of Testing .....	2.6-6
2.6.2.4	Test Parameters.....	2.6-6
2.6.2.5	Test Setup .....	2.6-10
2.6.2.6	Demonstration .....	2.6-10
2.6.2.7	Special Tests.....	2.6-12
2.6.2.8	Failure-Free-Performance.....	2.6-12
2.6.3	Thermal Balance Qualifications .....	2.6-12
2.6.3.1	Alternative Methods.....	2.6-13
2.6.3.2	Use of a Thermal Analytical Model .....	2.6-13
2.6.3.3	Method of Thermal Simulation .....	2.6-13
2.6.3.4	Internal Power .....	2.6-16
2.6.3.5	Special Considerations.....	2.6-16
2.6.3.6	Demonstration .....	2.6-16
2.6.3.7	Acceptance Requirements .....	2.6-17
2.6.4	Temperature-Humidity Verification.....	2.6-17
2.6.4.1	Temperature-Humidity Verification: Manned Spaces .....	2.6-17
2.6.4.1.1	Applicability .....	2.6-17
2.6.4.1.2	Demonstration .....	2.6-17

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Paragraph		Page
2.6.4.2	Temperature-Humidity Verification: Descent and Landing .....	2.6-17
2.6.4.2.1	Special Considerations.....	2.6-18
2.6.4.2.2	Demonstration .....	2.6-18
2.6.4.2.3	Acceptance Requirements .....	2.6-18
2.6.4.3	Temperature-Humidity: Transportation and Storage .....	2.6-18
2.6.4.3.1	Applicability .....	2.6-18
2.6.4.3.2	Demonstration .....	2.6-18
2.6.4.3.3	Acceptance Requirements .....	2.6-19
2.6.5	Leakage (Integrity Verification) .....	2.6-19
2.6.5.1	Levels of Assembly .....	2.6-19
2.6.5.2	Demonstration .....	2.6-19
2.6.5.3	Acceptance Requirements .....	2.6-19

## SECTION 2.7 - CONTAMINATION CONTROL

2.7	CONTAMINATION AND COATING ENGINEERING, AND PLANETARY PROTECTION	2.7-1
2.7.1	Contamination .....	2.7-1
2.7.1.1	Summary of Contamination Verification Process.....	2.7-1
2.7.1.2	Contamination Engineering Approach .....	2.7-2
2.7.1.3	Contamination Sensitivities .....	2.7-4
2.7.1.4	Contamination Requirements.....	2.7-4
2.7.1.5	Contamination Budget.....	2.7-4
2.7.1.6	Contamination Control Plan (CCP) .....	2.7-5
2.7.1.6.1	Supporting Contamination Documentation .....	2.7-5
2.7.1.7	Contamination Engineering During Design .....	2.7-6
2.7.1.8	Contamination Engineering Analyses .....	2.7-7
2.7.1.8.1	Molecular Contamination Analyses.....	2.7-7
2.7.1.8.2	Particulate Contamination Analyses .....	2.7-7
2.7.1.8.3	Other Analyses.....	2.7-8
2.7.1.9	Contamination Engineering During Integration and Test.....	2.7-9
2.7.1.10	Thermal Vacuum Bakeouts.....	2.7-9
2.7.1.11	Spacecraft Transportation and Storage Phases .....	2.7-10
2.7.1.12	Launch Site Contamination Control .....	2.7-10
2.7.1.13	Launch Vehicle and Companion Payload Considerations .....	2.7-10
2.7.1.14	Launch and Orbit Insertion Mission Phases .....	2.7-11
2.7.1.15	On-Orbit Through End-of-Life Mission Phases .....	2.7-11
2.7.1.16	Spacecraft Post-Mission and Follow-on Program Phases.....	2.7-12
2.7.2	Coatings Engineering.....	2.7-13
2.7.2.1	Key Verification Milestones for coatings Selections and Testing.....	2.7-14
2.7.3	Planetary Protection .....	2.7-14
2.7.3.1	Classification of Missions .....	2.7-15
2.7.3.2	Planetary Protection Categorization of Missions .....	2.7-15
2.7.3.3	Planetary Protection Plan Development (Categories II-V).....	2.7-16
2.7.3.4	Measurements and Verification.....	2.7-17
2.7.3.5	Documentation Requirements for Planetary Protection Requirements and Verification .....	2.7-17

## SECTION 2.8 - END-TO-END TESTING

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Paragraph		Page
2.8	END-TO-END COMPATIBILITY TESTS AND SIMULATIONS .....	2.8-1
2.8.1	Compatibility Tests .....	2.8-1
2.8.2	Mission Simulations.....	2.8-1

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Appendix A

General Information.....A-1 - A-14

Tables

Table

2.2-1	Flight System Hardware Levels of Assembly.....	2.2-3
2.2-2	Test Factors/Durations.....	2.2-5
2.2-3	Flight Hardware Design/Analysis Factors of Safety Applied to Limit Loads.....	2.2-6
2.4-1	Structural and Mechanical Verification Test Requirements.....	2.4-2
2.4-2	Minimum Probability-Level Requirements for Flight Limit (maximum expected) Level.....	2.4-3
2.4-3	Generalized Random Vibration Test Levels, Components.....	2.4-18
2.4-4	Component Minimum Workmanship Random Vibration Test Levels.....	2.4-19
2.5-1	Susceptibility Scanning (Replacement for MIL-STD-461F Table III).....	2.5-7
2.5-2	EMI Requirements Summary.....	2.5-16
2.6-1	Vacuum, Thermal, and Humidity Requirements.....	2.6-2
A-1	Acoustic Fill-Factor, 3 meter Payload Fairing.....	A-2

FIGURES

Figure

2.1-1	Environmental Test Matrix.....	2.1-5
2.1-2a	Verification Test Report.....	2.1-6
2.1-2b	Verification Test Report (continued).....	2.1-7
2.4-1	Shock Response Level (SRS) – for Assessing Component Test Requirements.....	2.4-26
2.5-1	Capacitor Network Simulating Battery Dominated Bus Impedance.....	2.5-6
2.5-2	General Test Setup Diagram.....	2.5-9
2.5-3	Conducted Emissions Test Setup.....	2.5-17
2.5-4	Conducted Emissions Differential Mode Limits.....	2.5-18

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2.5-5	Mode Conducted Emissions Default Limit .....	2.5-19
2.5-6	Common Mode Conducted Emissions Test Setup .....	2.5-20
2.5-7	Inrush Current Transient Default Limit .....	2.5-22
2.5-8	Inrush Current Emissions Test Setup (Current Probe) .....	2.5-23
2.5-9	Inrush Current Emissions Alternate Test Setup (Series Resistor) .....	2.5-24
2.5-10	Conducted Susceptibility Voltage Limit, (30 Hz to 150 kHz) .....	2.5-26
2.5-11	Conducted Susceptibility Power Limit (30 Hz to 150 kHz) .....	2.5-26
2.5-12	Conducted Susceptibility Transient Waveform .....	2.5-28
2.5-13	Conducted Transient Susceptibility Test Setup – Series Injection .....	2.5-28
2.5-14	Conducted Transient Susceptibility Test Setup – Parallel Injection .....	2.5-29
2.5-15	Common Mode Conducted Susceptibility Calibration Limit .....	2.5-30
2.5-16	Conducted Susceptibility, Power Leads, 150 kHz to 50 MHz Limit .....	2.5-31
2.5-17	Conducted Susceptibility, Power Leads Test Setup .....	2.5-32
2.5-18	CS115 Default Limit .....	2.5-33
2.5-19	Radiated Emissions Electric Field Limit with Suggested Notches .....	2.5-35
2.5-20	Impacts of Uncontrolled EGSE Common Mode Currents .....	2.5-39
2.5-21	Proper EGSE Shielding and Grounding Configuration .....	2.5-40
2.5-22	EGSE Cable Shield Termination – Connector Break (Preferred Method) .....	2.5-41
2.5-23	EGSE Cable Shield Termination –Stuffing Tube (Alternate Method) .....	2.5-41
2.5-24	Common Source Impedance and Harness Impedance (General) .....	2.5-43
2.5-25	Power Distribution Wiring, Large Military Platforms (Worst Case) .....	2.5-43
2.5-26	Power Distribution Wiring, GSFC Platforms (Typical) .....	2.5-44
2.5-27	Capacitor Network Simulating Battery Dominated Bus Impedance .....	2.5-44
2.5-28	Conducted Emissions Test Setup – Single Ended (Traditional) .....	2.5-47
2.5-29	Basic Common Mode Conducted Emissions Calibration Setup .....	2.5-49
2.5-30	Nominal Transfer Impedance of CISPR 16 Absorbing Clamp .....	2.5-50

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2.5-31	Solid State Switch Example Design .....	2.5-52
2.5-32	Measurement System Check (Current Probe) .....	2.5-53
2.5-33	Measurement System Check (Series Resistor) .....	2.5-54
2.5-34	Inrush Current Calibration Waveform (20% Tolerance).....	2.5-54
2.5-35	Input Power Voltage Sag Measurement Setup .....	2.5-55
2.5-36	Conducted Susceptibility Alternate Test Setup (Differential Mode, 30 Hz to 150 kHz) ...	2.5-58
2.6-1	Section 2.6 Thermal Requirements.....	2.6-3
2.6-2	Qualification (Protoflight or Prototype) and Flight Acceptance Thermal-Vacuum Temperatures .....	2.6-5
2.6-3	Temperature-Humidity Profile for Descent and Landing Demonstration .....	2.6-17
2.7-1	Contamination Engineering Approach .....	2.7-3
A-1	Cylindrical Payload in Fairing Acoustic Fill-Factor .....	A-1
A-2	Acoustic Fill-Factor for Various Size Payloads in a 3 Meter Diameter Payload Fairing .....	A-3
A-3	Determination of Qualification and Acceptance Random Vibration Test Levels .....	A-4
A-4	Shock Environment Produced by Linear Pyrotechnic Devices .....	A-7
A-5	Shock Environment Produced by Separation Nuts and Explosive Bolts .....	A-8
A-6	Shock Environment Produced by Pin-Pullers and Pin-Pushers .....	A-9
A-7	Shock Environment Produced by Bolt-Cutters, Pin-Cutters, and Cable-Cutters .....	A-10
A-8	Attenuation of Constant Velocity Line .....	A-11
A-9	Peak Pyrotechnic Shock Response vs. Distance .....	A-12
A-10	Shock Attenuation Example .....	A-13
A-11	Reduction of Pyrotechnic Shock Response due to Intervening Structure .....	A-14

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## SECTION I

### GENERAL INFORMATION

## 1.1 PURPOSE

This standard provides requirements and guidelines for environmental verification programs 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 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.

This standard shall be used by GSFC projects and contractors. This standard shall be tailored to create a project specific verification plan and verification specification as discussed in section 2.1. GSFC projects must select from the options to fulfill the specific payload (spacecraft) requirements in accordance with the launch vehicle to be used, Atlas, Delta, Pegasus, etc., or to cover other mission-specific considerations.

## 1.2 APPLICABILITY AND LIMITATIONS

This standard applies to GSFC hardware and associated software that is to be launched on an ELV. Hardware launched by balloons and sounding rockets is not included. 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 provisions herein are generally limited to the verification of 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.

The standard 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 standard 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 includes 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 standard rest with the project manager. The general/environmental requirements are intended for use by the flight project managers, assisted by the systems assurance managers, and systems engineering in developing project-unique performance verification requirements, plans, and specifications that are consistent with current NASA program/project planning.

#### 1.6 GEVS CONFIGURATION CONTROL AND DISTRIBUTION

This document is controlled and maintained by the GSFC Institutional Support Office and is available through the Goddard Document Management System (GDMS).

#### 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 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.

1.7.2 Deep Space Network (DSN) Simulation - The Deep Space Network/Flight Project Interface Design Handbook, 810-5, 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.

1.7.3 NASA Standards – The following standards provide supporting information:

- a. NASA-STD 7002, Payload Test Requirements
- b. NASA-STD-7001, Payload Vibroacoustic Test Criteria
- c. NASA-STD-7003, Pyroshock Test Criteria
- d. NASA-HDBK-7004, Force Limited Vibration Testing
- e. NASA-HDBK-7005, Dynamic Environmental Criteria
- f. NASA-STD-5001, Structural Design and Test Factors of Safety for Space Flight Hardware
- g. NASA-STD-5002, Load Analyses of Spacecraft and Payloads
- h. NASA-STD-5009, Nondestructive Evaluation Requirements for Fracture Critical Metallic Components
- i. NASA-STD-5019, Fracture Control Requirements for Spaceflight Hardware

1.7.4 Military Standards for EMI Testing - Pertinent sections of the following standards are needed to conduct the EMI tests:

- a. MIL-STD-461F, Requirements for the Control of Electromagnetic Interference Characteristics of Subsystems and Equipment, 10 December 2007

Additional documentation is specified in Section 2.5.

1.7.5 Military Standards for Non-Destructive Evaluation

- a. MIL-HDBK-6870, Inspection Program Requirements, Non-Destructive Testing for Aircraft and Missile Materials and Parts.
- b. NAS-410, Certification and Qualification of Nondestructive Test Personnel.
- c. MSFC-STD-1249, Standard NDE Guidelines and Requirements for Fracture Control Programs.
- d. MIL-HDBK-728, Nondestructive Testing.

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.

**Anomaly:** An unexpected event that is outside of design/performance specification limits.  
**NOTE:** Design limits are those identified in approved design-level documents.

**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 as well as 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 record keeping, 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, Satellite and System Segment.

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. The science instruments or experiments are also included as subsystems of the payload.

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 specified 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 stresses 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 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.

### 1.9.1 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.

### 1.9.2 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.

### 1.9.3 Failure Reporting

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

### 1.9.4 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.

## 1.10 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.

1.10.1 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. (sec. 2.1.1)

1.10.2 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.

1.10.3 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.

1.10.4 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.

1.11 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.

1.12 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.

1.13 TEST CONDITION TOLERANCES

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

<u>Acoustics</u>	Overall Level:	$\leq 1$ dB	
	1/3 Octave Band Tolerance:	<u>Frequency (Hz)</u>	<u>Tolerance (dB)</u>
		$f \leq 40$	+3, -6
		$40 < F < 3150$	$\pm 3$
	$f \geq 3150$	+3, -6	
	Duration	+10%, -0%	
<u>Antenna Pattern Determination</u>		$\pm 2$ dB	

<u>Electromagnetic Compatibility</u>			
	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	
<u>Humidity</u>		± 5% RH	
<u>Loads</u>			
	Steady-State (Acceleration):	± 5%	
	Sine Burst Amplitude:	± 5%	
	Static:	± 5%	
<u>Magnetic Properties</u>			
	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	
<u>Mass Properties</u>			
	Weight:	± 0.2%	
	Center of Gravity:	± 0.15 cm (± 0.06 in.)	
	Moments of Inertia:	± 1.5%	
<u>Mechanical Shock</u>			
	Response Spectrum:	Frequency (Hz)	Tolerance (dB)
	Simulated	$F_n \leq 3 \text{ kHz}$	± 6
		$F_n \geq 3\text{kHz}$	+9/-6

	Shaker	$F_n \leq 3$ kHz	$\pm 3$
	Overall Spectrum	> 50% of SRS magnitude above nominal test level	
	Time History:		$\pm 10\%$
<u>Pressure</u>	Greater than $1.3 \times 10^4$ Pa (Greater than 100 mm Hg):		$\pm 5\%$
	$1.3 \times 10^4$ to $1.3 \times 10^2$ Pa (100 mm Hg to 1 mm Hg):		$\pm 10\%$
	$1.3 \times 10^2$ to $1.3 \times 10^1$ Pa (1 mm Hg to 1 micron):		$\pm 25\%$
	Less than $1.3 \times 10^1$ Pa (less than 1 micron):		$\pm 80\%$
<u>Temperature</u>			$\pm 2^\circ\text{C}$
<u>Vibration</u>	Sinusoidal:	Amplitude Frequency	$\pm 10\%$ $\pm 2\%$
	Random:	RMS level Accel. Spectral Density Duration	$\pm 10\%$ $\pm 3$ dB $+10\%/-0\%$

1.14 TEST MEASUREMENT CONSIDERATIONS

From the moment a test article is excited or illuminated by an environmental source until the article is returned to an ambient condition, measurements should be collected over time at sufficient temporal resolution to capture any relevant frequency- or time-dependent effects and to eliminate, in the cases of an anomaly or test failure, the possibility that the incident was caused by overtest due to an error in process or in the test equipment itself.



## SECTION 2

### VERIFICATION PROGRAM

## SECTION 2.1

### SYSTEM PERFORMANCE VERIFICATION

## 2.1 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 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

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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 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 subsystem/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**

PAGE _____ OF _____	REVISION NO. _____ REVISION DATE _____		REMARKS
	HARDWARE DESCRIPTION	SUPPLIER	
ITEM	SIN	LEVEL OF ASSEMBLY	
STRUCTURAL MECHANICAL	QUALIFICATION STATUS		
	MODAL SURVEY		
	STATIC LOADS		
	ACCELERATION		
	SINE BURST		
	SINE VIBRATION		
	RANDOM VIBRATION		
	ACOUSTICS		
	MECHANICAL SHOCK		
	PRESSURE PROFILE		
	MECHANICAL FUNCTION		
	LIFE TESTS		
EMC & MAGNETICS	EMISSIONS		
	CONDUCTED		
	RADIATED		
	SUSCEPTIBILITY		
	CONDUCTED		
	RADIATED		
	LEAK		
	MAGNETIC SUSCEPTIBILITY		
	MAGNETIC FIELD SUSCEPTIBILITY		
	NUMBER OF THERMAL-VACUUM CYCLES		
	THERMAL BALANCE		
	TEMPERATURE HUMIDITY		
BAKEOUT			
FLIGHT TEMPERATURE PREDICTIONS			
TEST TEMPERATURE LIMITS			

NOTES:

LEGEND  
 LEVEL OF ASSEMBLY

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

VERIFICATION TEST REPORT		Page _____ of _____
PROJECT		
TEST ITEM		
MANUFACTURER		
SERIAL NUMBER		
<div style="border: 1px solid black; padding: 5px; margin-bottom: 5px;"> <p style="text-align: center; margin: 0;">LEVEL OF ASSEMBLY</p> <input type="checkbox"/> SUBASSEMBLY or ASSEMBLY  <input type="checkbox"/> UNIT/COMPONENT  <input type="checkbox"/> SECTION  <input type="checkbox"/> SUBSYSTEM/INSTRUMENT  <input type="checkbox"/> MODULE  <input type="checkbox"/> SPACECRAFT/PAYLOAD                 </div>	<div style="border: 1px solid black; padding: 5px; margin-bottom: 5px;"> <p style="text-align: center; margin: 0;">HARDWARE</p> <input type="checkbox"/> ENGINEERING MODEL  <input type="checkbox"/> PROTOTYPE  <input type="checkbox"/> PROTOFLIGHT  <input type="checkbox"/> FLIGHT  <input type="checkbox"/> SPARE                 </div>	<div style="border: 1px solid black; padding: 5px; margin-bottom: 5px;"> <p style="text-align: center; margin: 0;">TEST</p> <input type="checkbox"/> INITIAL TEST                      STARTING DATE OF INITIAL TEST                      _____  <input type="checkbox"/> RETEST  <input type="checkbox"/> PARTIAL  <input type="checkbox"/> FULL                 </div>
<div style="border: 1px solid black; padding: 5px; margin-bottom: 5px;"> <p style="text-align: center; margin: 0;">STRUCTURAL - MECHANICAL</p> <input type="checkbox"/> STRUCTURAL LOADS  <input type="checkbox"/> STATIC  <input type="checkbox"/> ACCEL.  <input type="checkbox"/> SINE BURST  <input type="checkbox"/> VIBRATION  <input type="checkbox"/> RANDOM  <input type="checkbox"/> SINE  <input type="checkbox"/> ACOUSTICS  <input type="checkbox"/> MECHANICAL SHOCK  <input type="checkbox"/> ACTUATION  <input type="checkbox"/> SIMULATED  <input type="checkbox"/> MECHANICAL FUNCTION  <input type="checkbox"/> MODAL SURVEY  <input type="checkbox"/> PRESSURE PROFILE  <input type="checkbox"/> MASS PROPERTIES  <input type="checkbox"/> OTHER (explain)                 </div>	<div style="border: 1px solid black; padding: 5px; margin-bottom: 5px;"> <p style="text-align: center; margin: 0;">ELECTROMAGNETIC COMPATIBILITY</p> <input type="checkbox"/> CONDUCTED EMISSIONS  <input type="checkbox"/> RADIATED EMISSION  <input type="checkbox"/> CONDUCTED SUSCEPTIBILITY  <input type="checkbox"/> RADIATED SUSCEPTIBILITY  <input type="checkbox"/> MAGNETIC PROPERTIES                 </div>	<div style="border: 1px solid black; padding: 5px; margin-bottom: 5px;"> <p style="text-align: center; margin: 0;">THERMAL</p> <input type="checkbox"/> THERMAL-VACUUM                      (no. of cycles _____ )  <input type="checkbox"/> THERMAL CYCLING                      (no. of cycles _____ )  <input type="checkbox"/> THERMAL BALANCE  <input type="checkbox"/> TEMPERATURE-HUMIDITY  <input type="checkbox"/> LEAKAGE  <input type="checkbox"/> OTHER (explain)                 </div>
	<div style="border: 1px solid black; padding: 5px; margin-bottom: 5px;"> <p style="text-align: center; margin: 0;">ELECTRICAL PERFORMANCE</p> <input type="checkbox"/> LPT  <input type="checkbox"/> CPT  <input type="checkbox"/> END-TO-END  <input type="checkbox"/> COMPATIBILITY TEST  <input type="checkbox"/> MISSION SIMULATIONS                 </div>	<div style="border: 1px solid black; padding: 5px; margin-bottom: 5px;"> <p style="text-align: center; margin: 0;">OPTICAL</p> <input type="checkbox"/> EXPLAIN                 </div>
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

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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.)

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## 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 and spacecraft that will be launched by expendable launch vehicles (ELVs). 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 "failure-free-performance" requirement may be varied, with GSFC approval, 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 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 Palette,      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 Qualification of Hardware by Similarity

There are cases in which hardware qualified for one flight program is to be built and used on another program. Hardware that has been previously qualified may be considered qualified for use on a new program by showing that the hardware is sufficiently similar to the original hardware and that the previous qualification program has adequately enveloped the new mission environments. The details for performing this comparison should be defined by the project but as a minimum the following areas should be reviewed and documented:

- (1) Design and test requirements must be shown to envelope the original requirements. This should include a review of the test configuration and of all waivers and deviations that may have occurred during testing of the original hardware.
- (2) Manufacturing information shall be reviewed to determine if changes have been made that would invalidate the previous hardware qualification. This review should cover

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parts, materials, packaging techniques as well as changes to the assembly process or procedures.

- (3) Test experience with the previous flight build shall be reviewed to verify that no significant modifications were made to the hardware during testing to successfully complete the test program. Any significant change shall be identified and shown to be implemented on the current flight hardware.

If the review of the above criteria shows that the hardware is of sufficiently similar design as the first build and that the previous test requirements envelope any new environmental requirements, then the hardware can be treated as qualified and need only to be subjected to acceptance level test requirements. The review of the hardware for similarity must be documented and included as part of the verification package.

#### 2.2.4 Test Factors/Durations

Test factors 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.

#### 2.2.5 Structural Analysis/Design Factors of Safety

Structural and mechanical verification testing shall be supported by structural analysis to provide confidence that the hardware will not experience failure or detrimental permanent deformation under test or launch conditions. The factors of safety that shall be applied to limit loads in order to calculate structural margins are shown in Table 2.2-3. These factors of safety have been selected to be consistent with the test factors shown in Table 2.2-2. The yield factor of safety ensures that a prototype or protoflight test can be conducted with low risk of the hardware experiencing detrimental yielding. The ultimate factor of safety provides adequate separation between yield and ultimate failure modes and ensures that the hardware will not experience an ultimate failure under expected loading conditions.

In the case of thermally induced loads or stresses, the factors of safety shown in Table 2.2-3 for the static loading condition are to be used for calculating strength margins. If the absolute value of the temperature differential between the stressed and un-stressed condition for the hardware using flight acceptance temperatures is  $\geq 20^{\circ}\text{C}$ , then the static factors of safety from Table 2.2-3 shall be applied to the loads/stresses induced by acceptance temperatures. Flight acceptance temperatures are defined as the maximum predicted flight temperature plus acceptance margin which is typically  $5^{\circ}$  for thermal vacuum or  $20^{\circ}$  for thermal cycling at ambient pressure as defined in Table 2.2-2. If the absolute value of the temperature differential between the stressed and un-stressed condition using flight acceptance temperatures is  $< 20^{\circ}\text{C}$ , then the static factors of safety from Table 2.2-3 shall be applied to the loads/stresses induced by qualification temperatures for the hardware. Both hot and cold conditions shall be evaluated and the factors of safety should be applied to the appropriate test temperature (acceptance or qualification). Thermally induced loads shall be combined with mechanical loading due to launch loads, gravity, or external loads due to enforced deflection if these mechanical loads occur at the same time.

Table 2.2-2  
 Test Factors/Durations

Test	Prototype Qualification	Protoflight Qualification	Acceptance
Structural Loads <sup>1</sup> Level  Duration Centrifuge/Static Load <sup>6</sup> Sine Burst	1.25 x Limit Load  1 minute 5 cycles @ full level per axis	1.25 x Limit Load  30 seconds 5 cycles @ full level per axis	1.0 x Limit Load  30 seconds 5 cycles @ full level per axis
Acoustics Level <sup>2</sup> Duration	Limit Level + 3dB 2 minutes	Limit Level + 3dB 1 minute	Limit Level 1 minute
Random Vibration Level <sup>2</sup> Duration	Limit Level + 3dB 2 minutes/axis	Limit Level + 3dB 1 minute/axis	Limit Level 1 minute/axis
Sine Vibration <sup>3</sup> Level Sweep Rate	1.25 x Limit Level 2 oct/min	1.25 x Limit Level 4 oct/min	Limit Level 4 oct/min
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. ± 5°C
Thermal Cycling <sup>4,5</sup>	Max./min. predict. ± 25°C	Max./min. predict. ± 25°C	Max./min. predict. ± 20°C
EMC & Magnetics	As Specified for Mission	Same	Same

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

Note: Test levels for beryllium and composite structure, including metal matrix, are 1.25 x Limit Level for both qualification and acceptance testing.

- 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 and dwell times be increased by 50% for thermal cycle (ambient pressure) testing.
- 5 - Thermal cycling testing performed as a screen for mechanical hardware with no heat generating devices may be tested to Thermal-Vacuum Test factors (See Section 2.6.2.4)

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- 6 - Shorter durations may be used in static testing if necessary to protect the hardware from damage due to facility limitations. If a shorter duration is used then the dwell time at load shall be sufficient to demonstrate that the target loading condition has been achieved within the specified tolerances, all test measurements have been recorded, and the structure is stable under the applied loading condition.

Table 2.2-3  
 Flight Hardware Design/Analysis Factors of Safety Applied to Limit Loads <sup>1,2</sup>

Type	Static	Sine	Random/Acoustic <sup>4,5</sup>
Metallic Yield	1.25 <sup>3</sup>	1.25	1.6
Metallic Ultimate	1.4 <sup>3</sup>	1.4	1.8
Stability Ultimate	1.4	1.4	1.8
Beryllium Yield	1.4	1.4	1.8
Beryllium Ultimate	1.6	1.6	2.0
Composite Ultimate	1.5	1.5	1.9
Bonded Inserts/Joints Ultimate	1.5	1.5	1.9

- 1 – Factors of safety for pressurized systems to be compliant with AFSPCMAN 91-710 (Range Safety).
- 2 – Factors of safety for glass and structural glass bonds specified in NASA-STD-5001
- 3 – If qualified by analysis only, positive margin must be shown for factors of safety of 2.0 on yield and 2.6 on ultimate. See section 2.4.1.1.1
- 4 – Factors shown should be applied to statistically derived peak response based on RMS level. As a minimum, the peak response shall be calculated as a 3-sigma value.
- 5 – Factors shown assume that qualification/protoflight testing is performed at acceptance level plus 3dB. If difference between acceptance and qualification levels is less than 3dB, then above factors may be applied to qualification level minus 3dB.



## 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. For electronics consisting of a prime and redundant, i.e. A and B sides, 1000 hours should be accumulated on each side.

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. The 350-hour demonstration should include at least 200 hours in vacuum. 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.

These requirements also apply to instruments and other spacecraft subsystem hardware prior to delivery for integration into the spacecraft. The Failure-free durations should be set dependent on the mission risk level, hardware complexity, and hardware criticality to the mission.

#### 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 entry, 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 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 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.

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	*	$T^2$	*
Design Qualification	*	A,T/A <sup>1</sup>	*
Structural Reliability Primary & Secondary Structure	*	(A,T) <sup>1</sup>	*
Vibroacoustics			
Acoustics	T	$T^2$	$T^2$
Random Vibration	$T^2$	$T^2$	T
Sine Vibration	$T^3, T^4$	$T^3$	$T^3$
Mechanical Shock	T	$T^5$	$T^5$
Mechanical Function	A,T	A,T	-
Pressure Profile	-	A,T <sup>2</sup>	A
Mass Properties	A/T	A,T <sup>2</sup>	*

\* = 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<sup>1</sup> = Analysis and Test or analysis only if no-test factors of safety given in 2.4.1.1.1 are used.

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

T<sup>2</sup> = Test must be performed unless assessment justifies deletion.

T<sup>3</sup> = Test performed to simulate low frequency transient vibration and any sustained periodic mission environment, or to satisfy other requirement such as strength qualification.

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

T<sup>5</sup> = 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
	ELV Payloads
Structural Loads	97.72/50 (1),(2)
Vibroacoustics Acoustics Random Vibration	95/50
Sine Vibration	97.72/50 (1)
Mechanical Shock	95/50
<p>Notes:</p> <p>(1) 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.</p> <p>(2) 97.72% probability of not exceeding level, estimated with 50% confidence. Equal to the mean plus two-sigma level for normal distributions.</p>	

A modal survey 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 if appropriate.

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

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.

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 using the appropriate factors of safety defined in Table 2.2-3.. 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 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 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 ELV payloads, the coupled loads analysis shall consider all flight events required by the ELV provider. None of the flight events 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 small and to demonstrate compliance with the launch vehicle interface verification and safety requirements. The analysis shall show positive margins using the appropriate factors of safety defined in Table 2.2-3.

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 additional qualification by analysis requirements, see 2.4.1.3. When qualification by analysis is used, the upper frequency of the modal survey may have to be increased.

2.4.1.1.2 Analysis - Clearance Verification - Analysis shall be conducted for all 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 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 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 have modes with significant modal mass within the frequency range of the launch vehicle coupled loads analysis. The frequency range covered by coupled loads analysis is dependent on the specific launch vehicle. The determination that a modal test is required will be made on a case-by-case basis and will be specified in the design and test requirements. Modal analysis of the hardware with appropriate boundary conditions may be used to determine the need for performing a modal survey. If the determination is made that a modal survey test is not required because the hardware does not have significant modes in the coupled loads range, then the fundamental frequency of the hardware shall be verified during vibration testing. A low-level sine survey is generally an appropriate method for determining fundamental frequency.

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. A mode is considered significant if

it has modal effect mass that is equal to or greater than 5% of the total mass of the hardware at the level of assembly for which the modal survey is being considered. Modes that drive high responses of critical components from a coupled loads analysis should also be considered as target modes for a modal survey. 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. Frequencies between the corresponding test and analytical modes shall match within 5%. Any test method that is capable of meeting the test objectives with the necessary accuracy may be used to perform the modal survey.

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 that will generate forces in the hardware that are equal to 1.25 times limit loads. The strength qualification test must be shown to produce forces equal to 1.25 times limit at structural interfaces as well as in structural elements which have been shown to have the lowest margins for all identified failure modes of the hardware. As many test conditions as necessary shall be applied to achieve the appropriate loads for qualification. Structural qualification testing should be performed at the lowest level of assembly as possible to reduce overtest and to limit the risk of damage to other components/subsystems should structural failure occur. After structural testing, the hardware must be capable of meeting its performance criteria (see 2.4.1.3.1 for special requirements for beryllium structure). 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 positive margins using the appropriate factors of safety defined in Table 2.2-3. 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, 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, beryllium or structures that have bonded or welded joints 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 1.25 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 1.4 on yield and 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 NASA-STD-5009.

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 applications that are not listed in Table 1 of MSFC-SPEC-3029, 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:

ELV Payloads - The following requirements regarding beryllium, nonmetallic-composite, and metallic-honeycomb structural elements (both primary and secondary), and bonded structural joints apply to 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): It is preferred that all flight structural elements shall be proof tested to 1.25 times limit load (even if previously qualified on valid prototype hardware). However, if this is not feasible then it is acceptable to proof test a representative set of structural elements to 1.25 times the highest limit load for that type of structure. The remainder of the structural elements may then be considered qualified by similarity. In order to use this approach, the allowables used to assess structural margins must be developed based on coupon testing and standard statistical techniques. As a minimum, B-basis allowables shall be used. 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) It is preferred that 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. However, in cases where this approach is not feasible, it is acceptable to test a representative sample of the bonded structural joints in the flight article. As a minimum, at least one of each type of bonded joint in the flight article shall be tested to 1.25 times the maximum predicted limit load for that joint type. The remainder of the bonded joints may then be considered to be qualified by similarity. The use of this approach requires that bonded joint allowables be developed based on coupon testing or testing of sample joints and standard statistical techniques. As a minimum, B-basis allowables shall be used.
  - (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.
  - (3) Composite or metallic honeycomb panels with bonded structural joints (fittings, inserts, doublers, and splices) in composite and metallic honeycomb shall be thermally cycled to the worst case temperature extremes prior to being subjected to structural proof testing.
- e. Weldments:
- (1) All flight structure with critical welds shall be proof tested to 1.25 times limit loads. A critical weld is defined as a weld in which a single failure will result in loss of load carrying capability under the applied loading condition.
  - (2) Appropriate NDE inspection shall be performed on critical welds before and after proof testing.

- f. Fracture Control Requirements: If the payload is to be placed in orbit by an ELV, fracture control requirements (per NASA-STD-5019) shall apply to the following elements only:
- (1) Pressure vessels, dewars, lines, and fittings (per NHB-8071.1),
  - (2) Castings (unless hot isostatically pressed and the flight article is proof tested to 1.25 times limit load),
  - (3) Weldments,
  - (4) Parts made of materials on Tables II or III of MSFC-SPEC-3029 if under sustained tensile stress. (Note: All structural applications of these materials require that a Materials Usage Agreement (MUA) must be negotiated with the project office; refer to project Materials and Processes Control Requirements,
  - (5) Parts made of materials susceptible to cracking during quenching,
  - (6) 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 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 ELV Payloads

- (1) Beryllium structure (primary and secondary) shall be proof tested to 1.25 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.
- (4) Critical welds in flight structure shall be proof tested to 1.25 times limit load (See Section 2.4.2.4.1e for the definition of critical welds).

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. More information on implementing the force-limiting control strategy can be found in Force Limited Vibration Testing NASA Technical Handbook, NASA-HDBK-7004.



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.

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 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. 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 an area 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. One of the microphones should be located above the test item for a free-field test. A distance from any surface of at least 1/4 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. If no such item is readily available, it is possible to preshape the spectrum in an empty test area. In that case, however, a low-level test should be performed after the test item has been placed in the test area to permit final adjustments to the shape of the acoustic spectrum.

Acoustic testing may be performed in a reverberant chamber or may be performed as a direct-acoustic field (DAF) test in which the acoustic pressure field is generated by banks of speakers. The preferred method for performing acoustic testing on flight hardware is with a reverberant chamber test. Comparison of data from test articles subjected to both reverberant and current state-of-the art DAF testing showed that the pressure field and measured responses from DAF testing can differ significantly from a reverberant field test even if the control microphones are kept within the test tolerances specified in Section 1.13. Because of the non-uniformity that may exist in the acoustic field generated by DAF testing, care must be taken when performing this type of test to have sufficient instrumentation on the test article to prevent exceeding hardware capability as the test level is increased and have an adequate number of microphones in place during the test to monitor the pressure field generated near critical items. It should also be noted that variability in the acoustic field generated by a DAF test may result in under-testing as well as over-testing in specific frequency bands and all efforts should be made to map the acoustic field relative to acoustically sensitive hardware to ensure that an adequate test can be achieved.

- 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, 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 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 2.4-3 may be used.

As a minimum, all components shall be subjected to the levels of Table 2.4-4, which represent a workmanship screening test. The minimum workmanship test levels are primarily intended for use on electrical, electronic, and electromechanical hardware.

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

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.

For components mounted on isolators, flexures, or other highly compliant mounting structure, adequate workmanship testing may not be achieved in the flight configuration. In this case, it may be necessary to test the component hard-mounted to the shaker to achieve sufficient input levels to verify workmanship. The hard-mounted test would be run in addition to testing the component with flight-like mounting hardware. The component must be assessed for the hard-mounted test configuration to ensure that the hardware can survive the test without damage.

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 2.4-4 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.

If the test levels create conditions that exceed appropriate design safety margins or cause unrealistic modes of failure, the input spectrum can be notched below the minimum workmanship level. This can be accomplished when flight or test responses at the higher level of assembly are known or when appropriate force limits have been calculated.

- 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.

Hardware that has beryllium, composite (including metal matrix), ceramic, or bonded joints in the structural load path and whose strength margins are driven by vibro-acoustic loading shall be tested to protoflight levels for random and/or acoustic testing even if the design has been previously qualified on a valid prototype or protoflight unit. Protoflight vibro-acoustic testing ensures that structure whose strength is workmanship or fabrication dependent is adequately screened to preclude failure at higher levels of assembly. Protoflight testing should be performed at the lowest level of assembly practical for the hardware.

During the test, performance shall be monitored in accordance with the verification specification.

Table 2.4-3  
 Generalized Random Vibration Test Levels  
 Components (ELV)  
 22.7-kg (50-lb) or less

Frequency (Hz)	ASD Level ( $g^2/Hz$ )	
	Qualification	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 $G_{rms}$

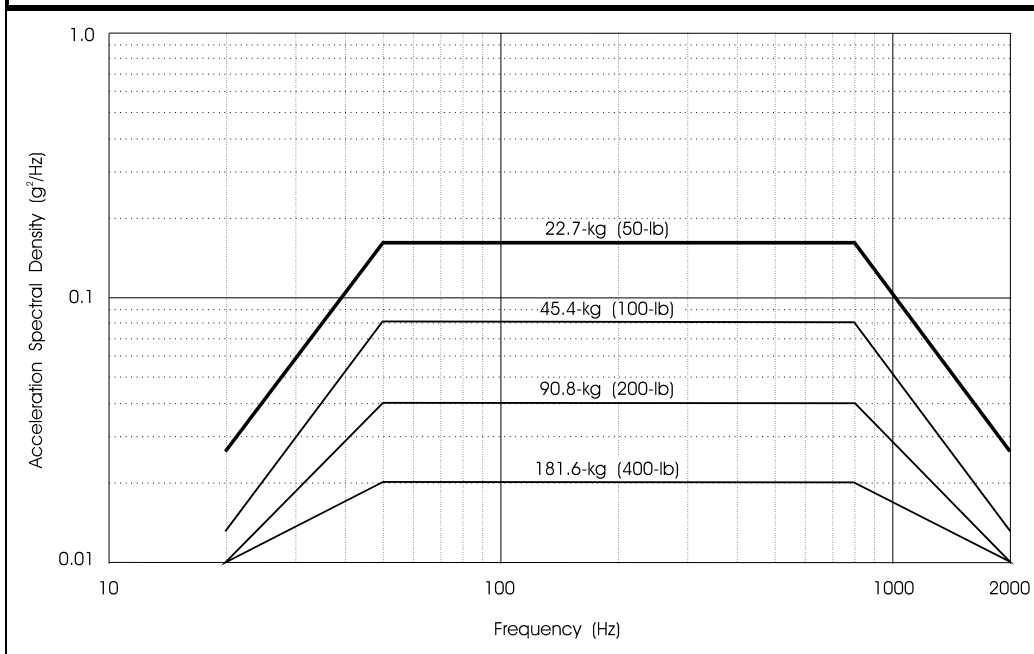
The acceleration spectral density level may be reduced for components weighing more than 22.7-kg (50 lb) according to:

	<u>Weight in kg</u>	<u>Weight in lb</u>	
dB reduction	= $10 \log(W/22.7)$	$10 \log(W/50)$	
ASD(50-800 Hz)	= $0.16 \cdot (22.7/W)$	$0.16 \cdot (50/W)$	for protoflight
ASD(50-800 Hz)	= $0.08 \cdot (22.7/W)$	$0.08 \cdot (50/W)$	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).



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Table 2.4-4  
 Component Minimum Workmanship  
 Random Vibration Test Levels  
 45.4-kg (100-lb) or less

Frequency (Hz)	ASD Level (g <sup>2</sup> /Hz)
20	0.01
20-80	+3 dB/oct
80-500	0.04
500-2000	-3 dB/oct
2000	0.01
Overall	6.8 grms

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:

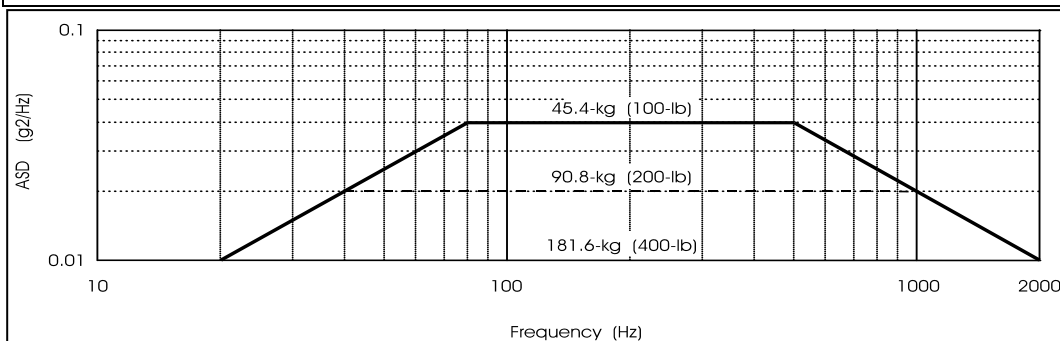
	<u>Weight in kg</u>	<u>Weight in lb</u>
dB reduction	= 10 log(W/45.4)	10 log(W/100)
ASD <sub>(plateau)</sub> level	= 0.04•(45.4/W)	0.04•(100/W)

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:

$F_L = 80 (45.4/W) \text{ [kg]}$       $F_L = \text{frequency break point low end of plateau}$   
 $= 80 (100/W) \text{ [lb]}$

$F_H = 500 (W/45.4) \text{ [kg]}$       $F_H = \text{frequency break point high end of plateau}$   
 $= 500 (W/100) \text{ [lb]}$

The test spectrum shall not go below 0.01 g<sup>2</sup>/Hz. For components whose weight is greater than 182-kg or 400 pounds, the workmanship test spectrum is 0.01 g<sup>2</sup>/Hz from 20 to 2000 Hz with an overall level of 4.4 g<sub>rms</sub>.



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.

2.4.2.8 Retest of Reworked Hardware – In many cases it is necessary to make modifications to hardware after a unit has been through a complete mechanical verification program. For example, replacing a capacitor on a circuit board in a electronics box that has already been through protoflight vibration testing. For this type of reworked hardware, the amount of additional mechanical testing required depends on the amount of rework done and the amount of disassembly performed as part of the rework. The primary objective of post-rework testing is to ensure proper workmanship has been achieved in performing the rework and in reassembling the component. As a minimum, the reworked component shall be subjected to a single axis workmanship random vibration test to the levels specified in Table 2.4-4. The determination of axis shall be made based on the direction necessary to provide the highest excitation of the reworked area. Testing may be required in more than one axis if a single axis test cannot be shown to adequately test all of the reworked area. If the amount of rework or disassembly required is significant, then 3-axis testing to acceptance levels may be necessary if they are higher than workmanship levels.

#### 2.4.3 Sinusoidal Sweep Vibration Qualification

Sine sweep vibration tests are performed to qualify prototype/protoflight hardware for the low-frequency 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, such as wiring harnesses and stowed appendages.

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.

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 start be limited to the



frequency range in which coupled loads results are applicable and may be used for notching test responses. The typical frequency range of the sine test is 5 to 50 Hz but the range of the test may be extended depending on the specific launch vehicle and the frequency content of the coupled loads analysis. 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.

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, 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.

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 frequency range of the sine test shall be consistent with the frequency content of the launch vehicle coupled loads analysis. 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.

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.

Coupled loads analysis results should be utilized to develop mission specific sinusoidal vibration test levels based on acceleration-response time histories or processed shock response spectra (SRS) data at the interface of the test article for all significant flight event loading conditions. Equivalent sine sweep vibration test input levels can be developed by processing the interface time history data using SRS techniques and then dividing the resulting SRS by the assumed Q (where  $Q=C_c/2C$ ).

It should be noted that, in developing equivalent test input levels by dividing the SRS by Q, the assumption of a lower Q is more conservative. In the absence of test data, typical assumed values of Q are from 10 to 20. For pogo-like flight events, the use of SRS techniques is not generally required.

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 for spacecraft base drive analysis, with notching levels based on net loads equivalent to the user manual c.g. load factor loads. The base-drive analysis shall be truncated to a frequency range consistent with the launch vehicle coupled loads analysis. 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 consistent with the launch vehicle coupled loads analysis and shall be the same as the frequency range defined for payload testing. 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 specified 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 or processed shock response spectra (SRS) data 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 (SRS) techniques for transient flight events using the methods defined in 2.4.3.1.b.

- (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 most shakers 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 with a duration comparable to the actual shock pulse shall be applied at the test item interface along each of the three axes. The shock spectrum of

the generated waveform (positive and negative) shall match the desired spectrum within the tolerances specified for mechanical shock in Section 1.13. Equalization of the shock spectrum is performed at a maximum resolution of one-sixth octave. The fraction of critical damping ( $c/c_c$ ) used in the shock spectral analysis of the test pulse should equal the fraction of critical damping 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 fraction of critical damping 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.

The decision to perform component shock testing to is typically based on an assessment of the shock susceptibility of the component and the expected shock levels. If the component is not considered shock sensitive and if there is low potential for damage due to the shock environment, then the project may choose to defer shock testing to the payload level of assembly. The potential for damage due to shock can be quantified based on Figure 2.4-1. Two curves are shown in the figure; one for standard aerospace electronics and one for all other hardware. If the flight shock environment as shown on an SRS plot ( $Q=10$ ) is enveloped by the appropriate curve shown in Figure 2.4-1, then the shock environment can be considered benign and there is low risk in deferring the shock test. For the case in which the shock levels are above the curve, then component level shock testing should be considered as the shock level may be high enough to cause damage. The curve provided in Figure 2.4-1 is intended as a guideline for determining whether component level shock testing should be performed. Each component should be evaluated individually to determine its susceptibility for damage due to the predicted shock environment.

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. Test Setup - 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.
- d. 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 specification.

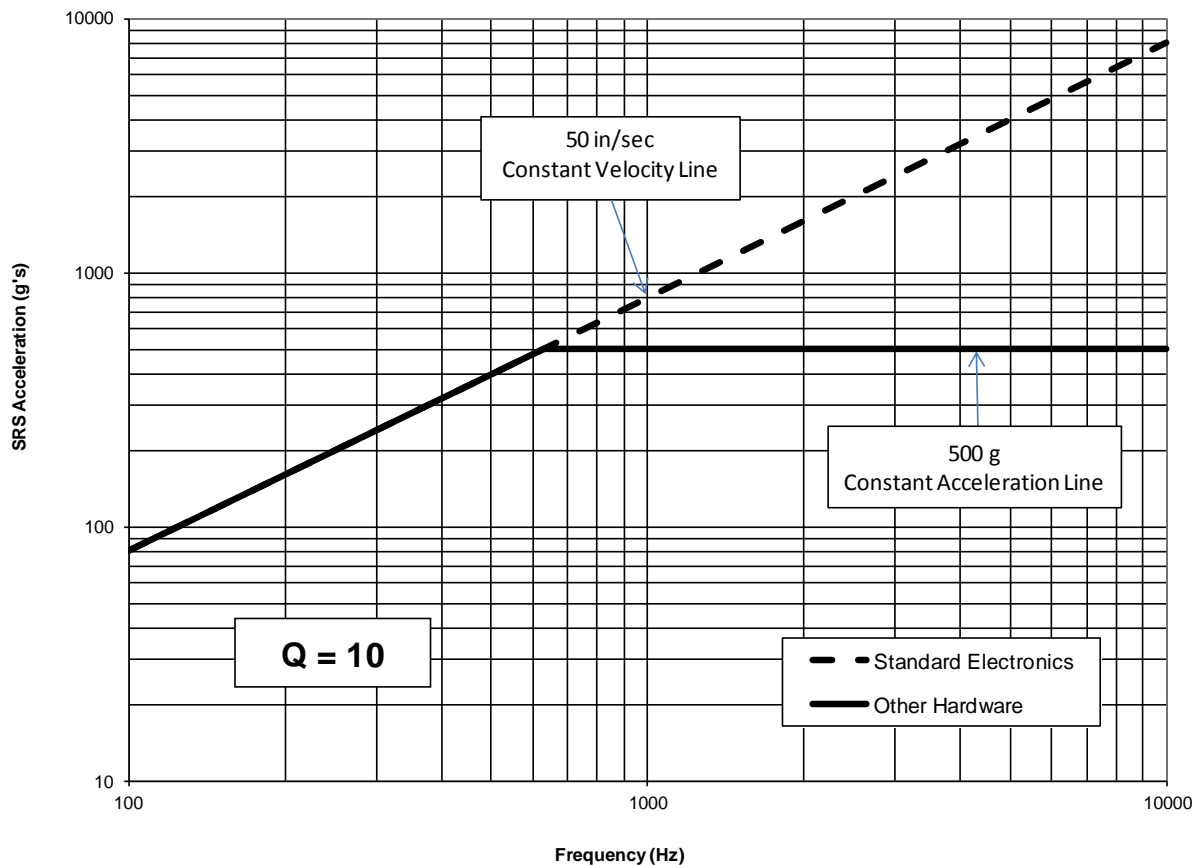


Figure 2.4-1 Shock Response Spectrum (SRS) for assessing Component Test Requirements

2.4.4.2 Payload (Spacecraft) Mechanical Shock Tests - 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. Other Payload (Spacecraft) Shocks - 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 in each of the three axes. The simulated shock spectrum (positive and negative) shall match the desired test spectrum within the tolerances for mechanical shock specified in Section 1.13. The analysis should be performed with a fraction of 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.

Lubricants susceptible to adverse effects from humidity, such as MoS<sub>2</sub> shall be given protection. Testing in a humid environment shall, where practicable, either be avoided or minimized.

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, and presented at each program system/peer review 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. Once the plan is finalized, an independent peer review of the procedure and criteria should be held.

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 or Qualification Unit. 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, both structural and thermal, that are anticipated for the flight units (protoflight or acceptance levels, 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 in a gaseous nitrogen environment as an inexpensive alternative to a long duration vacuum test, for example, may have a completely unexpected or unanticipated effect on lubricant tribology.

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, times of actuation and open/closed loop system responses.

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. Uninterruptible 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. Any autonomous data capture should include a time stamp to help diagnose the conditions present prior to a test shutdown.

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 and stop/start operations could have a significant effect on lubrication life, internal stresses, and, ultimately, the long term performance of the mechanism and therefore should be given priority in the development of the life test plan. Similarly, system dynamics effects due to inertial loads shall be considered in development of the plan and implemented where appropriate, such as in applications where normal operation includes multiple start / stop or acceleration / deceleration maneuvers.

The minimum requirement for demonstrated life test operation without failure shall be 2.0 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, even higher margins should be considered if time permits in order to establish greater confidence due to the limited number of life test units that are typically available. 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 mechanisms will be wear and lubricant breakdown, not fatigue. Unfortunately failure by wear is not an exact science; therefore, life test acceleration by increasing speed should be considered with caution. A mechanism that normally operates in these two regimes shall never be accelerated in a life test to a level where the lubrication system moves into the EHD regime for the test. Acceleration of a life test for systems in boundary or mixed lubrication regimes may be considered if it can be shown by analysis or test that the mechanism rotor oscillations for the accelerated operation are similar to that during normal operation. For example, in a step motor, it shall be shown that the rotor oscillations damp out to less than 10% of the peak overshoot amplitude prior to initiating the next accelerated step. Rationale for acceleration shall be presented in the initial test plan.

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. Additionally, thermal

issues may arise related to increased power dissipation for higher speed operation, like increased bearing gradients, which should be thoroughly evaluated.

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.

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 levels of inspection 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. Photographic documentation of the life test article should be made from incoming component inspection/acceptance through full assembly to act as a baseline for comparison.

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 testing. 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 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.

The selection of lubricant for use in critical moving mechanical assemblies shall be based upon development tests of the lubricant that demonstrate its ability to provide adequate lubrication under all specified operating conditions over the design lifetime. Since life testing cannot typically provide proof of lubricant availability based on evaporation over the required life of the mechanism, an analysis shall be performed to show that there is an adequate amount of lubricant in the system (not including degradation) for the duration of the mechanism life with a margin greater than 10. Lubricant availability analyses based on degradation rates should be proven through life testing (see section 2.4.5.1).

The design of each ball bearing installation shall be substantiated by analysis and either development tests or previous usage. The materials, stresses, stiffness, fatigue life, preload, and possible binding under normal, as well as the most severe combined loading conditions, and other expected environmental conditions shall be considered. Alignments, fits, tolerances, thermal and load induced distortions, and other conditions shall be considered in determining preload variations. Bearing fatigue life calculations shall be based on a survival probability of 99.95 percent when subjected to maximum time varying loads. For noncritical applications or deployables, if nonquiet running is acceptable, and the bearing material is 52100 Carbon Steel or 440C Stainless Steel, the mean Hertzian contact stress shall not exceed 2760 megapascals (400,000 psi) when subjected to the yield load. During operation, the mean Hertzian contact stress shall not exceed 2310 megapascals (335,000 psi). For materials other than these, a hertzian contact stress allowable shall be determined based on manufacturer recommendations with appropriate reduction factors for aerospace applications and approved by the responsible engineer.

In addition to the requirements stated above, bearing applications requiring quiet operation or low torque ripple shall be designed so that the bearing race and ball stress levels are below the levels that would cause unacceptable permanent deformation during application of ascent loads. Where bearing deformation is required to carry a portion or all of the vehicle ascent loads, and where smoothness of operation is required on orbit, the mean Hertzian stress levels of the bearing steel (52100 and 440C) shall not exceed 2310 megapascals (335,000 psi) when subjected to the yield load. The upper and lower extremes of the contact ellipses shall be contained by the raceways. The stress and shoulder height requirements of the races shall be analyzed for both nominal and off-nominal bearing tolerances. During operation, the mean Hertzian contact stress should not exceed 830 megapascals (120,000 psi) over the worst case environment. For materials other than 52100 carbon steel and 440C stainless steel, a Hertzian contact stress allowable shall be determined based on manufacturer recommendations with appropriate reduction factors for aerospace applications and approved by the responsible engineer.

- 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 torque or force ratio, a high-energy test and a low-energy test. The levels of these tests shall demonstrate margins beyond the nominal operational limits over the full range of motion at the worst case environments and the operating parameters of the system (rate, acceleration, etc.). The margins shall not be selected arbitrarily, but shall take into account all the uncertainties of operation, strength, and test. If a margin test cannot be conducted at the subsystem level due to its size and complexity these verification tests shall be performed at the highest level of assembly possible and the results combined to provide subsystem performance.

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 0.1 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.
- (4) Component Characterization and Testing – For applications where motor performance is critical to mission success, the design shall be based on a complete motor characterization at the minimum and maximum voltages from the spacecraft bus and motor driver and shall include as a minimum: rotor inertia, friction and damping parameters, back-EMF constant or torque constant, time constant, torque characteristics, speed versus torque curves, thermal dissipation, temperature effects, and where applicable, analysis to demonstrate adequate margin against back driving.

For applications where the motor is integrated into a higher assembly, the motor characterization shall be performed at the motor level prior to integration.

After initial functional testing, a run-in test shall be performed on each moving mechanical assembly before it is subjected to further acceptance testing, unless it can be shown that this procedure would be detrimental to performance and would result in reduced reliability. The primary purpose of the run-in test is to detect material and workmanship defects that occur early in the component life. Another purpose is to wear-in parts of the moving mechanical assembly so that they perform in a consistent and controlled manner. Satisfactory wear-in may be manifested by a reduction in running friction to a consistent low level. The run-in test shall be conducted for a minimum of 50 hours except for items where the number of cycles of operation, rather than hours of operation, is a more appropriate measure of the capability to perform in a consistent and controlled manner. For these units, the run-in test shall be for at least 15 cycles or 5% of the total expected life cycles, whichever is greater. The run-in test conditions should be representative of the operational loads, speed, and environment; however, operation of the assembly at ambient conditions may be conducted if the test objectives can be met and the ambient environment will not degrade reliability or cause unacceptable changes to occur within the equipment such as generation of excessive debris. During the run-in test, sufficient periodic measurements shall be made to indicate what conditions may be changing with time and what wear rate characteristics exist. Test procedures, test time, and criteria for performance adequacy shall be in accordance with an approved test plan. All gear trains using solid or liquid lubricants shall, where practicable, be inspected and cleaned following the run-in test.

2.4.5.3 Torque/Force Margin - The torque or force margin shall be demonstrated by test to be sufficiently large to guarantee system-performance under worst-case conditions throughout its life by fully accommodating the uncertainty in the resisting forces or torques and in the source of energy.

The Torque Margin (TM) is a measure of the degree to which the torque available to accomplish a mechanical function exceeds the torque required. The torque margin is generally the ratio of the driving or available torques times an appropriate Factor of Safety (FS) minus one.. The torque margin 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; however, all torque increases due to life test results should be included in the final TM calculation and verification. Positive margin (>0) using the TM equation and FS stated herein must be shown for worst case EOL predicted conditions and at the extreme operating parameters of the system (rate, acceleration, etc.). For linear devices, the term "force" shall replace "torque" throughout the section.

For final design verification, the torque margin shall be verified by testing the qualification (or protoflight) unit both before and after exposure to qualification level environmental testing. The torque margin on all flight units shall also be verified by testing when possible (without breaking the flight hardware configuration), both before and after exposure to acceptance level environmental testing. All torque margin testing should be performed at the highest possible level of assembly, throughout the mechanism's range of travel, under worst-case predicted EOL environmental conditions, representing the worst-case combination of maximum and/or minimum predicted (not qualification) temperatures, gradients, positions, acceleration/ deceleration of load, rate, voltage, vacuum, etc. As the deviation from these worst case conditions increases, a higher Factor of Safety than that stated below shall be used.

Along with system level test, available torque ( $T_{avail}$ ) and resistive torque ( $T_r$ ) under worst case conditions should be determined, whenever possible, through component, 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 that do not operate over the entire range of the mechanical function shall be neglected when computing available torque over the full range. However, the use of kick-off spring forces in the Torque Margin calculation at the beginning of travel or initial separation is acceptable. A Factor of Safety of at least 1.5 over inertial driven or known quantifiable resistive torques (that do not change over the operating life of the unit) shall be used in the final computing of torque margin as indicated in the table below. FS requirements for parasitic forces dominated by a combination of variable items should be determined based on the program phase as indicated in the table below. The final test verified Torque Margin shall be greater than zero (>0) based on the FS listed for the Acceptance / Qualification Test phase.

<b>Program Phase</b>	<b>Known Torque Factor of Safety (FS<sub>k</sub>)</b>	<b>Variable Torque Factor of Safety (FS<sub>v</sub>)</b>
Preliminary Design Review	2.00	4.0
Critical Design Review	1.50	3.0
Acceptance / Qualification Test	1.50	2.0

For those cases where high confidence does not exist in determination of worst case load or driving capability, a Safety Factor higher than that stated above may be appropriate. Factors of Safety should be based on a confidence level determined from the quantity and fidelity of heritage and program test data. At the program PDR, a detailed plan to determine torque margin shall be presented. By CDR, it shall be demonstrated (see GEVS section 2.4.5.2) that the detail design complies with the program requirements as outlined in this section.

The required Factors of Safety 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 margin testing is not performed in the required environmental conditions or is not repeatable and has a large tolerance band.
- c. The torque margin testing is performed only at the component level.

It is important to note that this torque margin requirement relates to the verification phase of the hardware in question. Conservative decisions should be made during the design phase to ensure adequate margins will be realized. However, it is recognized that under some unique circumstances these specified Factors of Safety might be detrimental (excessive) to the design of a system. For these specific cases which require approval of a waiver, appropriate Factors of Safety shall be determined based on the design complexity, engineering test data, confidence level, and other pertinent information.

The minimum available driving torque for the mechanism shall be determined based on the FS listed above. The Torque Margin (TM) shall be greater than zero and shall be calculated using the following formula:

$$TM = \{T_{avail} / (FS_k \Sigma T_{known} + FS_v \Sigma T_{variable})\} - 1$$

Where:

Driving Torques:

$T_{avail}$  = Minimum Available Torque or Force generated by the mechanism at worst case environmental conditions at any time in its life. If motors are used in the system,  $T_{avail}$  shall be determined at the output of the motor, not including gear heads or gear trains at its output based on minimum supplied motor voltage.  $T_{avail}$  similarly applies to other actuators such as springs, pyrotechnics, solenoids, heat actuated devices, etc.

Resistive Torques:

$\Sigma T_{known}$  = Sum of the fixed torques or forces that are known and quantifiable such as accelerated inertias ( $T=I\alpha$ ) and not influenced by friction, temperature, life, etc. A constant Safety Factor is applied to the calculated torque.

$\Sigma T_{variable}$  = Sum of the torques or forces that may vary over environmental conditions and life such as static or dynamic friction, alignment effects, latching forces, wire harness loads, damper drag, variations in lubricant effectiveness, including degradation or depletion of lubricant over life, etc.

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 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  $\pm 5\%$  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

This section establishes interface and associated verification requirements for the control of electromagnetic interference (EMI) emission and susceptibility characteristics of electronic, electrical, and electromechanical equipment and subsystems designed or procured for use on Goddard Space Flight Center (GSFC) Spacecraft.

This section is intended to provide guidance to each project for creating its own dedicated EMC Control Plan (EMCCP) or equivalent document that defines its electromagnetic environment and its requirements that have been tailored to that environment. The project EMCCP should be updated and maintained throughout the program.

Additional EMC documentation is provided on the GSFC EMC Working Group wiki site, maintained by Code 565, Electrical Systems Branch, GSFC:

<https://aetdwiki.gsfc.nasa.gov/display/EMCWG/Home>

The GSFC EMC Working Group wiki page includes all of the reference documents listed in Section 2.5.1.1 as well as documentation of lessons learned from previous programs. Organizations outside of GSFC should contact their GSFC EMC representative in order to obtain the information available on this wiki site.

This section of the document is organized as follows:

- Section 2.5.1 defines the general requirements, including test facility and procedure requirements;
- Section 2.5.2 defines the detailed requirements;
- Section 2.5.3 provides an Application Guide, which includes a discussion and rationale of the general and detailed requirements

### 2.5.1 General Requirements

The general requirements for electromagnetic compatibility (EMC) are as follows:

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

### 2.5.1.1 Documentation

All of the reference documents listed in this section are available on the “Documents” page of the GSFC EMC Working Group wiki site (<https://aetdwiki.gsfc.nasa.gov/display/EMCWG/Home>).

The EMI test requirements in this document are based primarily on the requirements and test methods of the following document:

MIL-STD-461F Requirements for the Control of Electromagnetic Interference Characteristics of Subsystems and Equipment, 10 December 2007

As needed, this document also references requirements, test methods, and tailoring guidelines provided in the following documents:

MIL-STD-461C Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference, 4 August 1986

MIL-STD-462, Notice 2 Electromagnetic Interference Characteristics, Measurement of, 1 May 1970

MIL-STD-464C Electromagnetic Environmental Effects Requirements for Systems, 1 December 2010

SL-E-0002, Book 3 Space Shuttle Specification, Electromagnetic Interference Characteristics, Requirements for Equipment, 10 August 2001

AIAA S-121-2009 Electromagnetic Compatibility Requirements for Space Equipment and Systems

(no document number) The Design, Construction and Test of Magnetically Clean Spacecraft - A Practical Guide (Mario H. Acuña)

Guidelines for grounding, bonding, and shielding are provided in the following documents:

NASA-HDBK-4001 Electrical Grounding Architecture for Unmanned Spacecraft

NASA-STD-4003 Electrical Bonding for NASA Launch Vehicles, Spacecraft, Payloads, and Flight Equipment

MIL-HDBK-419A Grounding, Bonding, and Shielding for Electronic Equipments and Facilities (Volumes 1 & 2, 29 December 1987)

MIL-HDBK-1857 Grounding, Bonding and Shielding Design Practices

Guidelines for addressing Space Charging effects are provided in the following document:

NASA-HDBK-4002A Mitigating In-Space Charging Effects - A Guideline

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

Guidelines for assessing Launch Vehicle & Launch Site Electromagnetic Environments are provided in the following document:

ELVL-2010-0042300      Electromagnetic Environments – A Guideline for Spacecraft Launching from Eastern, Western and Pacific Ranges

Guidelines for assessing on-orbit RF environments are provided in the following document:

JSC-CR-06-070      Space Vehicle RF Environments

### 2.5.1.2 Requirements Overview

Requirements and tests are prescribed at the component, subsystem, and payload levels of assembly. Not all requirements 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. For example, a transmitter would require a different group of EMI tests than a receiver.

Once the program is selected, all flight hardware shall be tested at the component or subsystem level. Tests at this level are designed to assess the risk of interference at higher levels of assembly; as such, they are called electromagnetic interference (EMI) tests. The EMI test program is meant to uncover design flaws, workmanship defects, and unit-to-unit variations. These component/subsystem level EMI tests are the primary focus of this document.

The EMI tests are intended to verify that the design and workmanship of each component will be compatible with its intended/predicted electromagnetic environments. All tests must simulate the flight configuration to the extent feasible. 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 by the project EMC engineer(s) to accommodate mission-specific requirements, as explicitly recommended in MIL-STD-461F Section 1.2.2:

“Application-specific environmental criteria may be derived from operational and engineering analyses on equipment or subsystems being procured for use in specific systems or platforms. When analyses reveal that the requirements in this standard are not appropriate for that procurement, the requirements may be tailored and incorporated into the request-for-proposal, specification, contract, order, and so forth, prior to the start of the test program. The test procedures contained in this document should be adapted by the testing activity for each application.”

Such requirements, as defined in the appropriate mission-specific documentation, may include:

- platform-specific environments
- launch vehicle and launch site environments
- protection of sensitive detectors or instruments in the payload
- environments encountered during ground testing that may differ from expected on-orbit environments

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Thus tailored, the requirements envelope the worst-case environments encountered during all phases of the program. However, because some payloads may have sensors and devices that are particularly sensitive to the low-level EMI ground environment, special workaround procedures may have to be developed in order to meet individual payload needs. Testing at the payload, spacecraft, and observatory levels are designed to assess compatibility at these higher levels of assembly; as such, they are called electromagnetic compatibility (EMC) tests.

Guidelines for defining system level EMC testing are provided in MIL-STD-464C, Electromagnetic Environmental Effects Requirements for Systems. Such EMC testing may include, but is not necessarily limited to, the following tests:

- Aggregate conducted emissions, frequency domain
- Aggregate ripple, voltage and/or current, time domain
- Aggregate radiated emissions, electric field (on-orbit and launch levels)
- Radio-frequency (RF) self-compatibility (verify no interference to on-board RF receivers and from on-board RF transmitters)
- System self-compatibility (verify no crosstalk between subsystems; generally included in system level comprehensive performance test)

Test programs at these higher levels of assembly must be developed on a case-by-case basis according to the needs of each platform. Additional guidelines are provided on the GSFC EMC Working Group wiki site (<https://aetdwiki.gsfc.nasa.gov/display/EMCWG/Home>).

#### 2.5.1.3 Testing at Lower Levels of Assembly

EMI testing must be performed at the component and subsystem levels of assembly, followed by EMC testing at the payload, spacecraft, and observatory levels. In addition, it is recommended that diagnostic testing be performed at the circuit board level to the extent feasible. If possible, such testing should be performed at the breadboard level in order to identify and correct problems as early as possible before finalizing the flight design.

Testing at lower levels of assembly has many advantages, including:

- Detection of problems early in the program when they are less costly to correct, less disruptive to the program schedule, and more easily diagnosed and addressed than at higher levels of assembly;
- Providing a baseline and troubleshooting aid that can be used to alert the project to potential problems at higher levels of assembly

#### 2.5.1.4 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 EMI testing. Operational control procedures should also be instituted for EMI testing during prerelease checkout to minimize interference with other equipment as appropriate.

The requirements documented within the EMCCP and/or levied in component/sub-system specifications represent the environment in which the equipment must function and meet its performance requirements. All requirements, without exception, shall be satisfied. Spurious emissions or susceptibilities determined non-compliant with requirements will be eliminated or reduced to sufficient levels to satisfy original requirements with adequate margin. All data/analysis presented for component/sub-system qualification will be subject to technical board review by GSFC EMC engineers and evaluated for compatibility within the intended payload system. GSFC EMC Engineering reserves the right to make final determination of qualification. Any Equipment Under Test (EUT) judged non-compliant by this review board is required either to make corrective actions or to follow the waiver process described in section 2.5.1.6.4.

#### 2.5.1.5 Test Facility and Procedure Requirements

The test facility and procedure requirements defined in MIL-STD-461F Section 4.3 shall apply with modifications to the following requirements as discussed in the following subsections:

- Ambient electromagnetic level (MIL-STD-461F Section 4.3.4)
- Power source impedance (MIL-STD-461F Section 4.3.6)
- Construction and arrangement of EUT cables (MIL-STD-461F Section 4.3.8.6)
- Susceptibility testing (MIL-STD-461F Section 4.3.10.4)
- General Test Setup (MIL-STD-461F Figure 2)

Additional discussions of these requirements are provided in Section 2.5.3.

Any deviations from these requirements must be approved by the project EMC engineer(s) and documented in the project EMC Control Plan (EMCCP).

##### 2.5.1.5.1 Ambient Electromagnetic Level

The ambient electromagnetic level shall be measured with the Equipment Under Test (EUT) de-energized and all Electrical Ground Support Equipment (EGSE) powered on. Ambient conducted levels on power leads shall be measured with the leads disconnected from the EUT and connected to a resistive load which draws the same rated current as the EUT. The ambient level shall be verified to be at least 6 dB below the specified limits and recorded in the EMI Test Report (EMITR) prior to performing any emissions measurements on the EUT.

These requirements override the following statement from MIL-STD-461F Section 4.3.4: "When tests are performed in a shielded enclosure and the EUT is in compliance with required limits, the ambient profile need not be recorded in the EMITR."

Guidelines for complying with the ambient electromagnetic level are provided in Section 2.5.3.2.1.

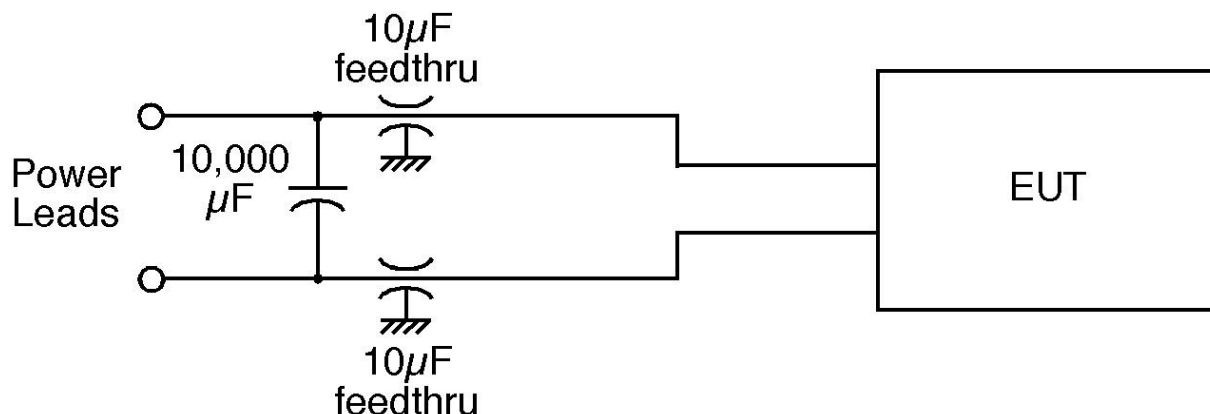
##### 2.5.1.5.2 Power Source Impedance



The impedance of power sources providing input power to the EUT shall be controlled by the capacitor network shown in Figure 2.5-1 for all measurement procedures of this document unless otherwise stated in a particular test procedure. This capacitor network simulates a battery dominated power bus typical of GSFC platforms.

For all MIL-STD-461F based test methods specified in this document, the Line Impedance Stabilization Networks (LISNs) will be replaced by this capacitor network.

Additional discussions of power source impedance and LISNs are provided in Section 2.5.3.2.2.



**Figure 2.5-1. Capacitor Network Simulating Battery Dominated Bus Impedance**

2.5.1.5.3 Construction and Arrangement of EUT Cables

The signal cables connecting to the EUT shall comply with the requirements defined in MIL-STD-461F Section 4.3.8.6. In particular:

- Electrical cable assemblies shall simulate actual installation and usage.
- Cables shall be checked against installation requirements to verify proper construction techniques such as use of twisted pairs, shielding, and shield terminations.
- Individual leads shall be grouped into cables in the same manner as in the actual installation.

The power cables connecting to the EUT shall comply with the requirements defined in MIL-STD-461F Section 4.3.8.6.2 with the following modification:

One (1) meter of input power leads (including neutrals and returns) shall be routed parallel to the front edge of the setup in the same manner as the interconnecting leads.

Additional discussion of construction and arrangement of EUT cables is provided in Section 2.5.3.2.3.

2.5.1.5.4 Susceptibility Testing

Susceptibility testing in MIL-STD-461F Section 4.3.10.4 addresses the following topics:

- Frequency scanning (MIL-STD-461F Section 4.3.10.4.1)
- Modulation of susceptibility signals (MIL-STD-461F Section 4.3.10.4.2)
- Thresholds of susceptibility (MIL-STD-461F Section 4.3.10.4.3)

Each of these topics will be discussed in turn below.

2.5.1.5.4.1 Frequency scanning (MIL-STD-461F Section 4.3.10.4.1)

For swept frequency susceptibility testing, frequency scan rates and frequency step sizes of signal sources shall not exceed the values listed in Table 2.5-1. This table replaces MIL-STD-461F Table III. All of the updates are in the last row as noted below.

The upper frequency for susceptibility scans is limited to 18 GHz instead of 40 GHz. The scan from 18 – 40 GHz requires considerably more test time with little added benefit in most cases. If the EUT shows no susceptibilities up to 18 GHz, then the risk of susceptibility between 18 GHz and 40 GHz is very small.

In the 1 GHz to 18 GHz frequency range, the maximum scan rate and maximum step size are increased by a factor of 4 over the values in MIL-STD-461F Table III. Scans in this frequency range take a significant amount of test time; the values in Table 2.5-1 will decrease test time while still providing sufficient frequency resolution to address susceptibility concerns in this frequency range.

**Table 2.5-1. Susceptibility Scanning (Replacement for MIL-STD-461F Table III)**

Frequency Range	Analog Scans Maximum Scan Rates	Stepped Scans Maximum Step Size
30 Hz - 1 MHz	0.0333 $f_o$ /sec	0.05 $f_o$
1 MHz – 30 MHz	0.00667 $f_o$ /sec	0.01 $f_o$
30 MHz - 1 GHz	0.00333 $f_o$ /sec	0.005 $f_o$
1 GHz - 18 GHz	0.00667 $f_o$ /sec	0.01 $f_o$

2.5.1.5.4.2 Thresholds of susceptibility (MIL-STD-461F Section 4.3.10.4.3)

Should an EUT fail to meet its susceptibility requirements at the full test level, the threshold(s) of susceptibility shall be determined in accordance with MIL-STD-461 procedures.

Should the supplier propose to deliver the EUT as prototype flight equipment without further modification and re-test, sufficient test data shall be gathered to support the processing of a waiver using the program's established process. Unless deemed different by the procuring agency (see MIL-STD-461F, Section 4.3.9.3), the following information shall be provided:

- Onset of susceptibility, defined as degradation in one or more EUT performance parameters beyond product specification limits or levels.
- EUT transition from soft upset (EUT returns to normal operation upon removal of the immunity stimulus) to hard upset (operator intervention required to reset to normal operation after removal of immunity stimulus), should it occur at or below the full required susceptibility test level.
- Description of degradation of performance with full test level applied.

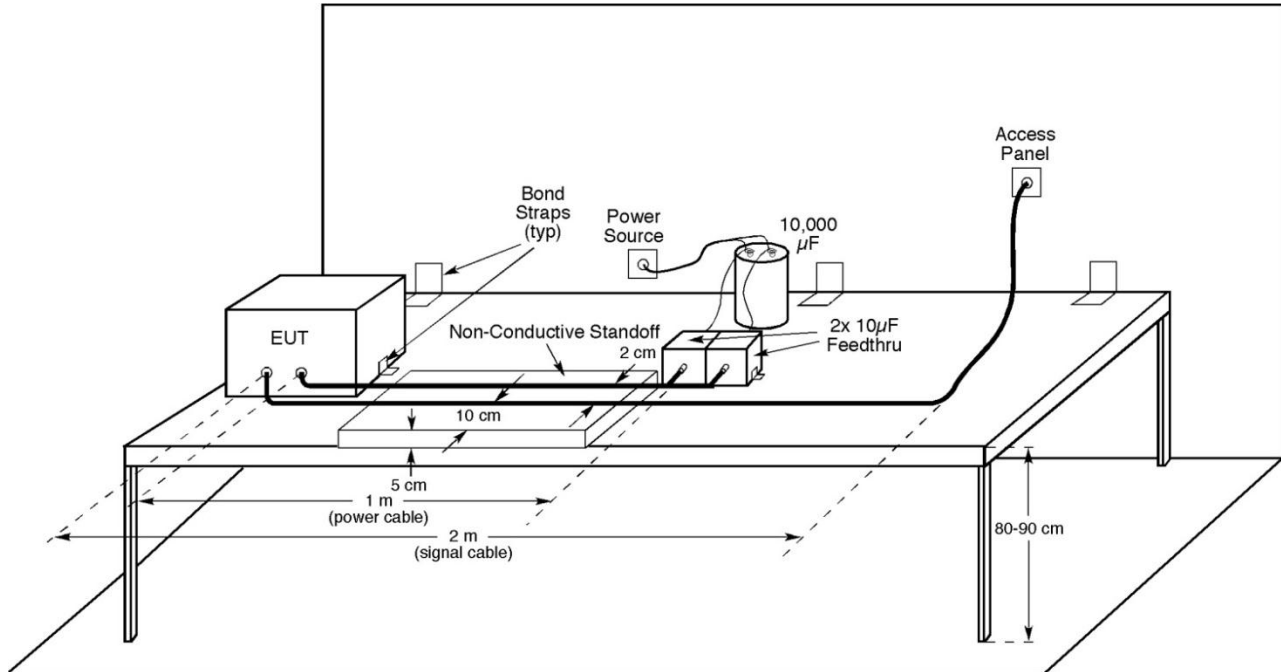
MIL-STD-461F specifies thresholds of susceptibility in terms of signal amplitude. It may also be desirable to determine the threshold of susceptibility to signal modulation as follows:

- When a susceptibility condition is detected, reduce the modulation duty cycle until the EUT recovers.
- Reduce the modulation duty cycle by an additional 10% or by half, whichever is less.
- Gradually increase the modulation duty cycle until the susceptibility condition reoccurs. The resulting level is the threshold of susceptibility.
- Record this modulation duty cycle, frequency range of occurrence, frequency and level of greatest susceptibility, and other test parameters, as applicable.

If there is susceptibility found at a frequency where a known modulation is used by a specific transmitter, the threshold(s) of susceptibility should also be determined as described above using that modulation scheme at those frequencies of interest. For equipment that is not designed to detect phase modulation, phase modulation may be simulated using no modulation, i.e. continuous wave (CW).

2.5.1.5.5 General Test Setup

The general setup for the EMI tests in this document is shown in Figure 2.5-2. Note that the figure is a modified version of MIL-STD-461F Figure 2 showing the capacitor network specified in Section 2.5.1.5.2 and the 1 meter power cables specified in Section 2.5.1.5.3 of this document.



**Figure 2.5-2. General Test Setup Diagram (Modified MIL-STD-461F Figure 2)**

### 2.5.1.6 Documentation Guidelines

It is strongly recommended that each project create a dedicated EMC Control Plan (EMCCP), either as a stand-alone document or as part of another project document. In addition, an EMI Test Procedure (EMITP) and EMI Test Report (EMITR) must be created for each EUT. The contents of each of these documents will be defined in the following subsections.

#### 2.5.1.6.1 EMC Control Plan (EMCCP)

The EMCCP shall provide the following:

- 1) Requirements with appropriate tailoring and documentation references (e.g. GEVS, MIL-STD-461F, etc.)
- 2) Management areas, including:
  - a. Specific organizational responsibilities, lines of authority and control, and program planning, including milestones and schedules.
  - b. Role in program of Government Furnished Equipment (GFE) and subcontractor items.
  - c. Description of the equipment or subsystem, its function, characteristics, and intended installation.
  - d. Plans and procedures for identifying and resolving potential EMI problems, implementing solutions, and verifying solutions through analysis and testing.
  - e. Point of contact for EMI technical issues.
- 3) Design techniques, including:
  - a. Spectrum management techniques (antenna locations, patterns, operating frequency ranges, sensitivity levels for receivers, power levels for transmitters, etc.)
  - b. EMI mechanical design, including the following:
    - i. Type of metals, casting, finishes, and hardware employed in the design.
    - ii. Construction techniques, such as isolated compartments; filter mounting, isolation of other parts; treatment of openings (ventilation ports, access hatches, windows, metal faces and control shafts), and attenuation characteristics of Radio Frequency (RF) gaskets used on mating surfaces.
    - iii. Shielding provisions and techniques used for determining shielding effectiveness.
    - iv. Corrosion control procedures.

- v. Methods of bonding mating surfaces, such as surface preparation and gaskets.
- 4) Electrical wiring design, including cable types or characteristics, cable routing, cable separation, grounding philosophy, and cable shielding types, transmission lines, and termination methods.
- 5) Electrical and electronic circuit design, including the following:
  - a. Filtering techniques, technical reasons for selecting types of filters, and associated filter characteristics, including attenuation and line-to-ground capacitance values of AC and DC power line filters.
  - b. Part location and separation for reducing EMI.
  - c. Location, shielding, and isolation of critical circuits.
- 6) Analysis. The EMCCP shall provide analysis results demonstrating how each applicable requirement is going to be met.
- 7) Developmental testing. The EMCCP shall include a discussion of testing to be performed during development (such as evaluations of breadboards, prototypes, and engineering models).

#### 2.5.1.6.2 EMI Test Procedure (EMITP)

The EMITP describes the measurement procedures that will be used to demonstrate that the EUT complies with its contractual EMI requirements defined in the project EMCCP (based on this document), including how the general test procedures will be applied to the specific equipment or subsystem.

The EMITP shall be provided for review by the project EMC engineer(s) and the EMI test facility at least 2 weeks prior to the start of EMI tests in order to verify adequate specification of test requirements and procedures; ensure that all required test equipment is available and to give the test facility adequate time to prepare.

The project shall provide the following information to be included in the EMITP (may be in the form of a test plan provided to the EMI test facility at least 6 weeks prior to the scheduled start date of the tests):

- 1) Documentation references (e.g. project EMCCP, GEVS, MIL-STD-461F, etc.).
- 2) A table describing all the tests to be performed, the applicable section within the EMITP (or test plan), the corresponding test procedure from this document or MIL-STD-461F, and any associated tailoring of the limits and/or test methods.
- 3) Description of the EUT, including its function, characteristics, intended installation, and power usage.
- 4) Description of platform's power distribution architecture and definition of capacitor network or LISN to be used for tests.
- 5) Approved exceptions or deviations from contractual test requirements, if any.

- 6) EUT setup. A description and diagram of the EUT test setup for each test shall cover the following:
  - a. Physical layout of the cables and EUT (see Figure 2.5-2).
  - b. Cable types, characteristics, and construction details (see Section 2.5.1.5.3 of this document)
  - c. Interface at chamber penetration, e.g. bulkhead feedthrough connectors, stuffing tube, etc.
  - d. Position of the capacitor network/LISN on the ground plane.
  - e. Use of bond straps and loads.
  - f. Test simulation and monitoring equipment.
- 7) EUT operation. A description of the EUT operation shall cover the following:
  - a. Modes of operation for each test, including operating frequencies (where applicable), and rationale for selection.
  - b. Control settings on the EUT.
  - c. Control settings on any test stimulation and monitoring equipment and characteristics of input signals.
  - d. Operating frequencies (such as oscillator and clock frequencies) which may be expected to approach limits.
  - e. Performance checks initiated to designate the equipment as meeting minimal working standard requirements.
  - f. Enumeration of circuits, outputs, or displays to be monitored during susceptibility testing, including the specific definition of acceptance criteria for monitoring the functional degradation of performance during exposure to an electromagnetic disturbance.
  - g. Description of longest machine cycle or response time of critical circuits; used for defining dwell time for susceptibility tests

The EMI Test Facility shall provide the following information for the EMITP:

- 1) Test site description, covering the following:
  - a. Test facility and shielded enclosure or anechoic chamber, including size, characteristics (e.g. shielding effectiveness), and placement of radio frequency (RF) absorbers.
  - b. Description of feedthrough filters, if used, at chamber feedthrough panel
  - c. Ground plane (size and type) and methods of grounding or bonding the EUT to the ground plane to simulate actual equipment installation.

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- d. Implementation of test facility and procedure requirements as specified in Section 2.5.1.5 of this document
- 2) Test instrumentation. Test instrumentation to be used shall be described as follows:
  - a. Equipment nomenclature.
  - b. Characteristics of coupling transformers and band-reject filters.
  - c. Antenna factors of specified antennas, transfer impedances of current probes, and LISN impedance(s), if used.
  - d. Description of the operations being directed by software for computer-controlled instrumentation, the verification techniques used to demonstrate proper performance of the software, and the specific versions of the software to be used. In addition, sweep times, correction factors and how are they used, how final data are determined and presented, and an audit trail that provides details on what part of the software controls each function shall be described.
  - e. Bandwidth (resolution and video) and scanning speeds of measurement receivers.
  - f. Modulation characteristics and scan rates of the susceptibility test signals.
- 3) Measurements. The following shall be described for each test.
  - a. Block diagram depicting test setup, including all pertinent dimensions.
  - b. Step-by-step procedures.
  - c. Test equipment used in performance of the test and the methods of grounding, bonding, or achieving electrical isolation of the measurement instrumentation.
  - d. Selection of measurement frequencies.
  - e. Information to be recorded during the test, including frequency and units of recorded information. Sample data sheets, test logs and graphs, including test limits, may be shown.

#### 2.5.1.6.3 EMI Test Report (EMITR)

The EMITR provides the data and information necessary to evaluate compliance of the EUT with its EMI control requirements based on the project EMCCP, including the discussion of recommended corrective actions, if needed.

The EMITR shall provide the following:

- 1) Documentation references (e.g. project EMCCP, EMITP, GEVS, MIL-STD-461F, etc.).
- 2) Administrative data. The EMITR shall contain an administrative section covering the following:
  - a. Contract number.

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- b. Authentication and certification of performance of the tests by a qualified representative of the procuring activity.
  - c. Disposition of the EUT.
  - d. Description of the EUT, including its function, characteristics, intended installation, actual cable types (characteristics and construction details - see Section 2.5.1.5.3 of this document and Figure 2.5-2), and electrical current usage on each power input line.
  - e. List of tests performed with pass/fail indications.
  - f. Any approved deviations from contractual test procedures or limits previously authorized.
  - g. Identification of Non-Developmental Items (NDI) and Government Furnished Equipment (GFE) that may be part of the EUT.
  - h. Traceability of test equipment calibration.
  - i. A reference to the approved EMITP.
- 3) Detailed results. A separate appendix shall be prepared for each test. If deviations from an approved test procedure occurred during the test program, an additional appendix shall be provided with the "as run" procedures showing all red-lines and procuring activity concurrence. A separate appendix shall be provided for log sheets. Each test appendix shall contain the following factual data:
- a. Test equipment nomenclature, serial numbers, version of software used (if any), and calibration due date.
  - b. Photographs or diagrams of the actual test set up and EUT, with identification.
  - c. Transfer impedance of current probes.
  - d. Antenna factors.
  - e. LISN impedance(s), if used.
  - f. Identification of any suppression devices used to meet the contractual requirements, including schematics, performance data, and drawings.
  - g. Sample calculations, such as conversions of measured levels for comparison against the applicable limit.
  - h. The ambient radiated and conducted electromagnetic emission profile of the test facility, when necessary.
  - i. Data, and data presentation, as specified in the "data presentation" sections of the individual test procedures of MIL-STD-461. Emissions data cannot be provided on one single sheet (plot). Provide a minimum frequency resolution of 1% or twice the measurement receiver bandwidth, whichever is less stringent,

and a minimum amplitude resolution of 1 dB for each plot. Data files for specified scans shall be provided upon request in a standardized electronic format.

- j. Scan speeds.
  - k. Measurement receiver bandwidths.
  - l. Antenna polarization.
  - m. Power line voltages, frequencies, and power factor.
  - n. Low-noise amplifiers (LNA) compression points.
  - o. Test susceptibilities and deviations from normal performance are not acceptable. However, when susceptibility indications or deviations from normal performance are noted in the EUT operation, a threshold level shall be determined where the susceptible condition or deviation is no longer present. These thresholds need to be documented for further assessment.
- 4) Conclusions and recommendations, including results of the tests in brief narrative form, a discussion of any remedial actions already initiated, and proposed corrective measures required (if necessary) to assure compliance of the equipment or subsystem with the contractual EMI requirements.

#### 2.5.1.6.4 Waiver Requests

Each non-compliance must be brought to the project to be assessed for corrective actions needed to eliminate the non-compliance and for potential impacts to the program. If the technical impacts to the platform are determined to be acceptable, the non-compliance must be submitted as a waiver request. If the board determines that the risk of the non-compliance is acceptable to the program and that the EUT does not require corrective action, then the waiver request will be granted.

Each waiver request shall contain:

- 1) Documentation references (e.g. project EMCCP, EMITP, EMITR, GEVS, MIL-STD-461F, etc.).
- 2) Identification of specific requirement(s) affected.
- 3) Detailed technical description of non-compliance, including:
  - a. Identification of the source
  - b. Detailed description of potential approach to fix or mitigate the non-compliance.
- 4) Assessment of the cost, schedule impact and any other impacts to the program (e.g., mass and space allocations) due to implementing, or not implementing, the fix.

2.5.2 Detailed Requirements

The EMI requirements are divided into four categories: Conducted Emissions (CE), Conducted Susceptibility (CS), Radiated Emissions (RE), and Radiated Susceptibility (RS). Table 2.5-2 provides the summary list of specific requirements defined in this document. For each requirement, the equivalent MIL-STD-461F designator is identified, and GEVS tailoring of the limit(s) and/or test method is indicated.

The details of the requirements are provided in the following subsections.

**Table 2.5-2. EMI Requirements Summary**

Test Description	MIL-STD-461F Designator	GEVS Tailoring	
		Limit(s)	Test Method
Conducted Emissions, Power Leads, 30 Hz to 150 kHz	CE101	tailored	DM
Conducted Emissions, Power Leads, 150 kHz to 50 MHz	CE102	tailored	CE03* (DM)
Conducted Emissions, Common Mode, Power and Signal Lines	no equivalent	new	new
Conducted Emissions, Turn-on Transients	no equivalent	new	new
Conducted Emissions, Antenna Terminal	CE106	tailored	no change
Conducted Susceptibility, Power Leads	CS101	tailored	alternate method available
Conducted Susceptibility, Antenna Port, Intermodulation	CS103	tailored	no change
Conducted Susceptibility, Antenna Port, Rejection of Undesired Signals	CS104	tailored	no change
Conducted Susceptibility, Antenna Port, Cross-Modulation	CS105	not applied	not applied
Conducted Susceptibility, Transients, Power Leads	CS106	CS06*	CS06*
Conducted Susceptibility, Structure Current	CS109	not applied	not applied
Conducted Susceptibility, Bulk Cable Injection, 10 kHz to 200 MHz	CS114	tailored	no change
Conducted Susceptibility, Bulk Cable Injection, Impulse Excitation	CS115	tailored	no change
Conducted Susceptibility, Damped Sinusoidal Transients	CS116	not applied	not applied
Radiated Emissions, Magnetic Field	RE101	tailored	alternate method available (RE04*)
Radiated Emissions, Electric Field	RE102	tailored	alternate methods available
Radiated Emissions, Antenna Spurious and Harmonic Outputs	RE103	not applied	not applied
Radiated Susceptibility, Magnetic Field	RS101	tailored	no change
Radiated Susceptibility, Electric Field	RS103	tailored	no change
Radiated Susceptibility, Transient Electromagnetic Field	RS105	not applied	not applied

\* MIL-STD-462 Test Method

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2.5.2.1 Conducted Emissions Requirements

Conducted emission requirements on power leads, signal leads, and antenna terminals shall be applied to payload and spacecraft hardware as defined in the following subsections.

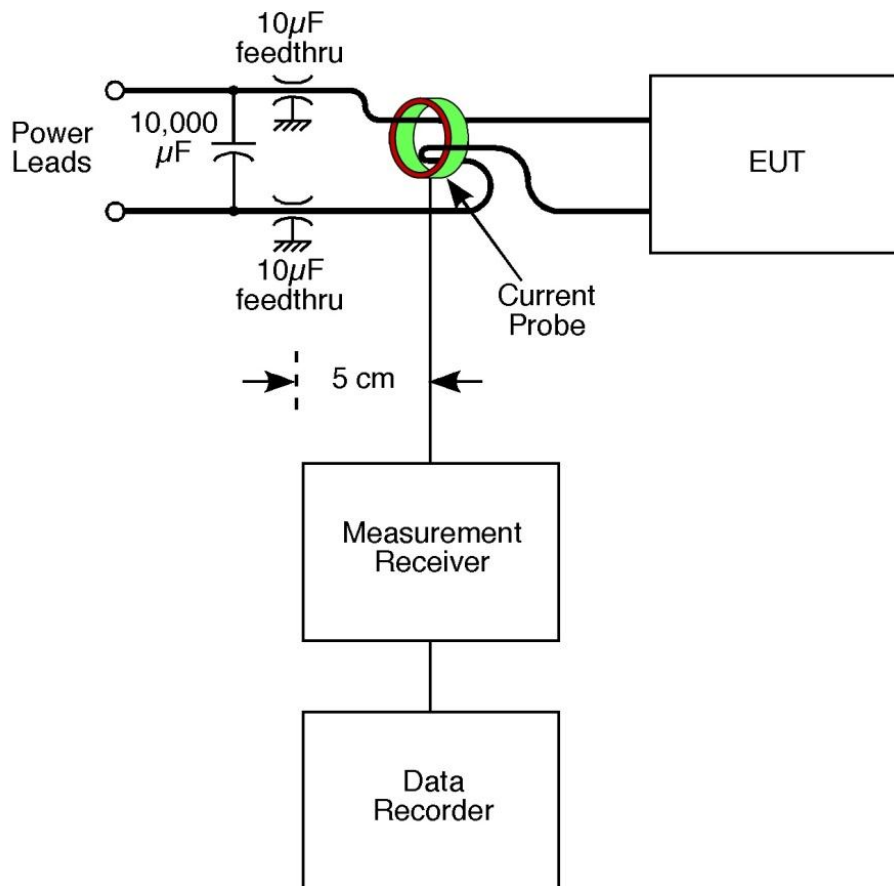
2.5.2.1.1 Conducted Emissions, Power Leads, Differential Mode

Applicability: This requirement is applicable to equipment power leads, including returns, that operate from a spacecraft primary power bus that may be shared with other loads.

Purpose: The purpose of this requirement is to limit the conducted emissions from the EUT in order to control load-induced effects on power quality.

Test method: From 30 Hz to 150 kHz, the basic MIL-STD-461F CE101 test method shall apply with the send and return wires run through the current probe for a differential mode (DM) measurement as shown in Figure 2.5-3.

From 150 kHz to 50 MHz, the test method shown in Figure 2.5-3 shall apply using a current probe that is suited to the higher frequency range. This is effectively the basic MIL-STD-462 CE03 test method with the send and return wires run through the current probe for a DM measurement.

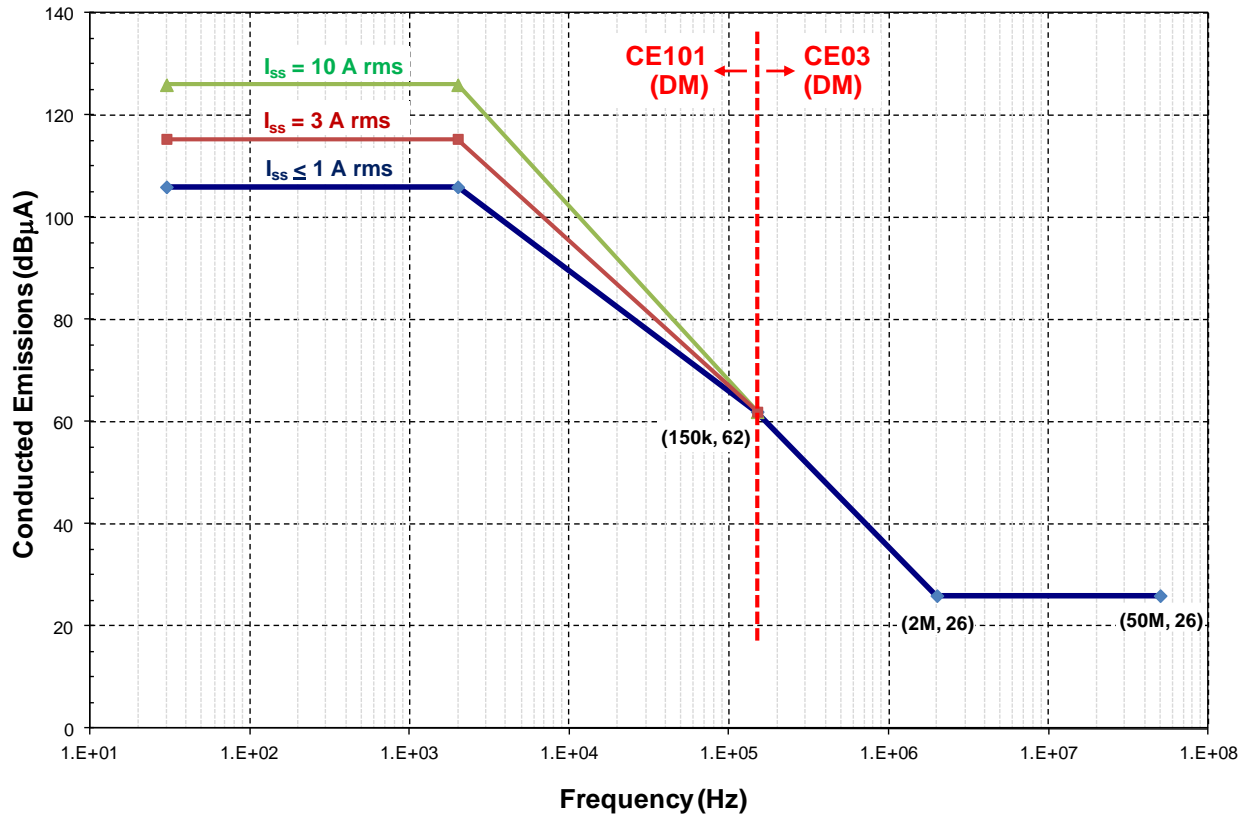


**Figure 2.5-3. Conducted Emissions Test Setup - Differential Mode**

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

Limit: The DM mode current emissions limits are shown in Figure 2.5-4. Because the (DM) test method measures twice the true DM mode current (6 dB above the true value), the measured limit must be set 6 dB above the true maximum DM mode current.

The default limit (blue curve) is for loads with a steady state current of 1 Amp DC or less. The low frequency plateau shifts upward for higher currents by adding a factor of  $20 \cdot \log(\text{load current in Amperes})$ . Example curves corresponding to load currents of 3 A rms and 10 A rms are shown.



**Figure 2.5-4. Conducted Emissions Differential Mode Limits**

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

2.5.2.1.2 Conducted Emissions, Common Mode, Power and Signal Lines

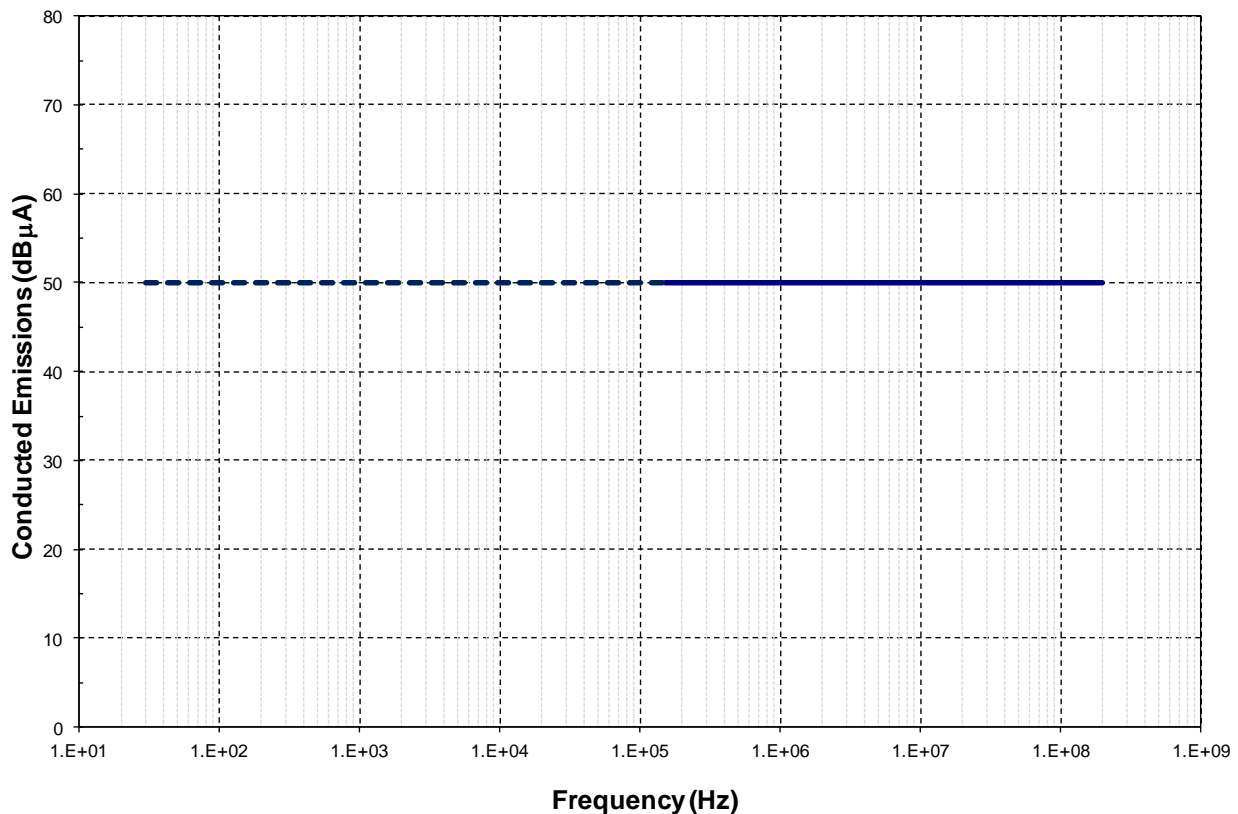
Applicability: This requirement is applicable to power and signal cables.

Purpose: The purpose of this requirement is to limit the common mode (CM) conducted emissions from bulk cables in order to protect against cable-to-cable crosstalk. This requirement is intended to be a companion to the Common Mode Conducted Susceptibility requirements (CS114 & CS115).

Limit: There is no MIL-STD-461F equivalent for this requirement.

The default limit is shown in Figure 2.5-5. The default frequency range defined by the solid line, 150 kHz to 200 MHz, is determined largely by the frequency response of the absorbing clamp used for the test. If the platform's magnetic requirements necessitate control of common mode currents below 150 kHz, it may be desirable to extend the limit down to 30 Hz as shown by the dashed line.

Discussion of this limit and tailoring guidelines are provided in Section 2.5.3.3.2.



**Figure 2.5-5. Common Mode Conducted Emissions Default Limit**

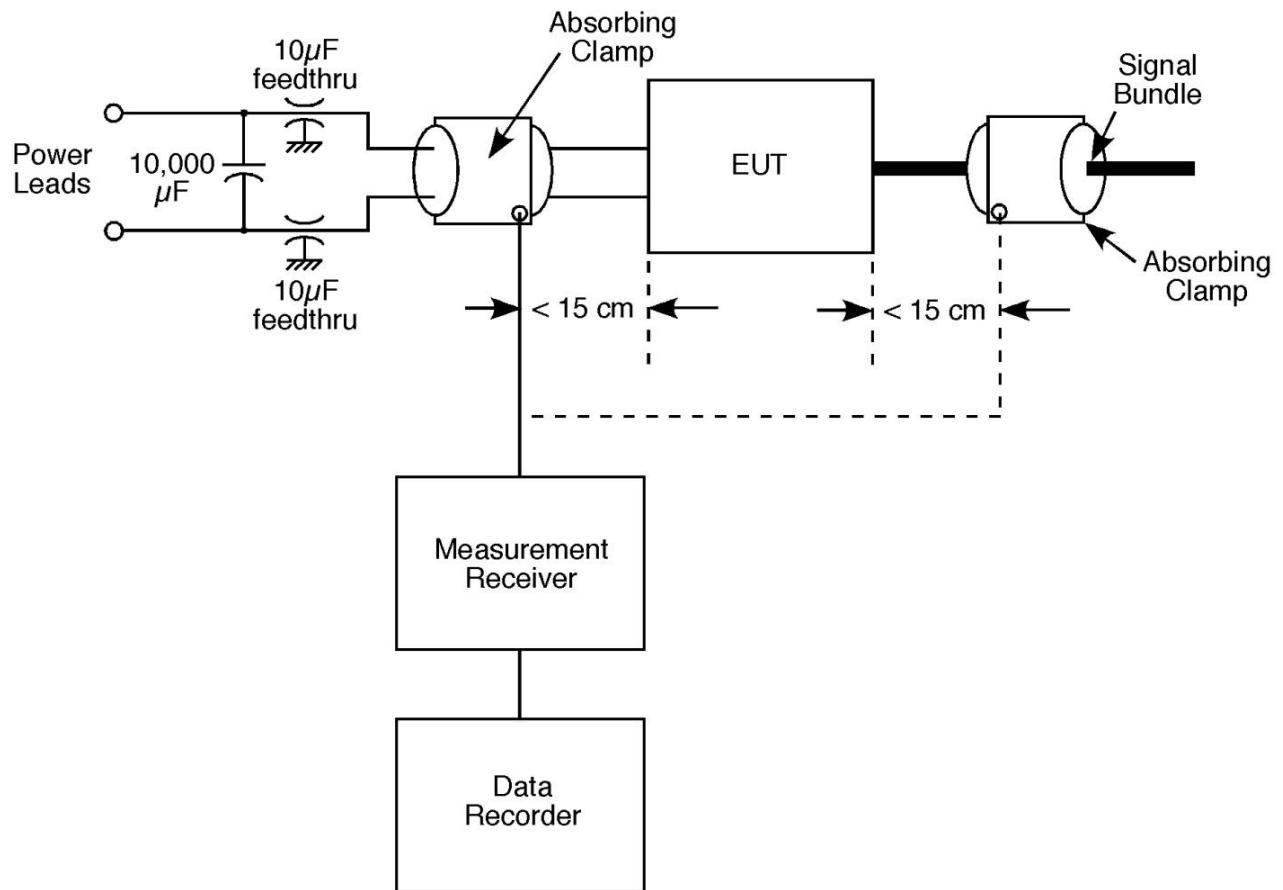
Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

Test Method: There is no MIL-STD-461F equivalent for this requirement.

The test method from 150 kHz to 200 MHz is adapted from the alternate RE102 test method below 200 MHz for Bulk Cable Emission (BCE) defined in SL-E-0002, Space Shuttle Specification, Electromagnetic Interference Characteristics, Requirements for Equipment. The basic test setup is shown in Figure 2.5-6; details are provided in Section 2.5.3.3.2.

The test method from 30 Hz to 150 kHz, if used, is the same as that shown in tailored CE101 test method use for the Conducted Emissions, Differential Mode requirement in section 2.5.2.1.1, except that the current probe is placed around each individual bulk cable under test for a common mode measurement. The basic test setup is still as shown in Figure 2.5-6 with the CE101 current probe replacing the absorbing clamp.

Additional discussion of this test method is provided in Section 2.5.3.3.2.



**Figure 2.5-6. Common Mode Conducted Emissions Test Setup**

### 2.5.2.1.3 Conducted Emissions, Time Domain, Transients

Applicability: This requirement applies to the following:

- 1) Turn-on transients generated by equipment that is connected to the spacecraft primary power bus through an external relay, solid-state switch or circuit breaker located in the bus power system.
- 2) Operational transients generated by equipment that enables and disables secondary loads (e.g. secondary power converters, motors, reaction wheels, cryocoolers, etc.)

Purpose: The purpose of this requirement is to ensure that equipment inrush current due to turn-on and operational transients will neither adversely affect the power bus nor overstress its circuit protection device (e.g. fuse).

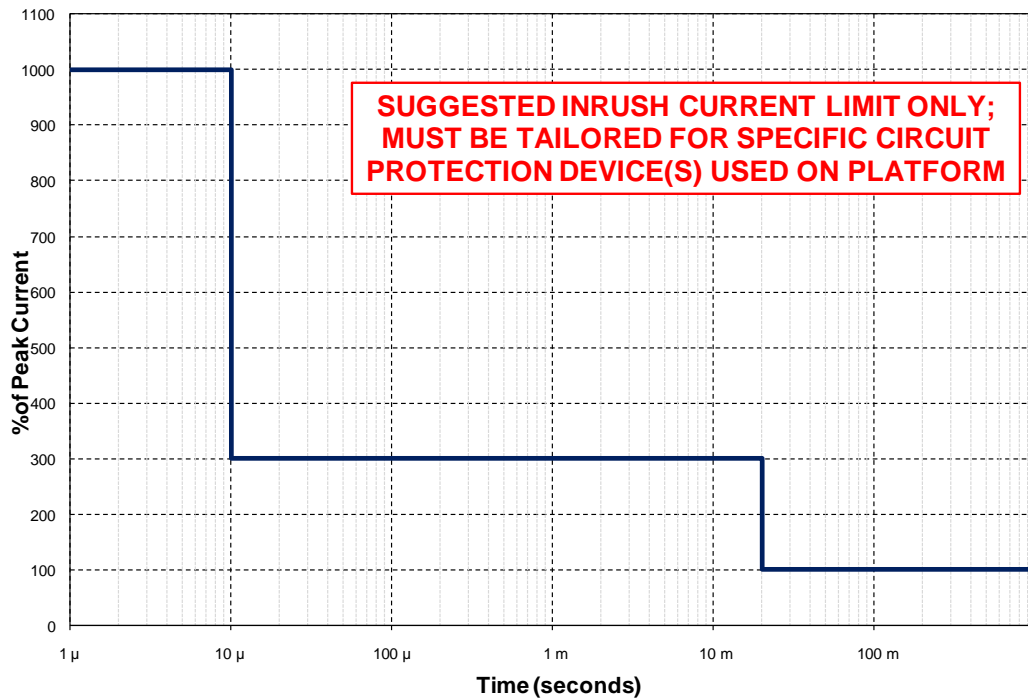
Limit: There is no MIL-STD-461F equivalent for this requirement. The default limit is shown in Figure 2.5-11 and is described as follows:

The peak inrush current caused by turn-on or operational transient shall not exceed 1000% of the unit peak operational current for the first 10 microseconds after the start of the transient, shall not exceed 300% of the unit peak operational current from 10 microseconds to 20 milliseconds after the start of the transient, and shall return to peak operational current within 20 milliseconds of the start of the transient. The requirement shall be met when the unit input voltage is the maximum steady-state operational voltage of the unit.

The default limit must be compared against the characteristics of the specific circuit protection device(s) used on the platform and tailored as necessary. Additional discussion is provided in Section 2.5.3.3.2.

It is recommended that the inrush current measurement be repeated at payload and/or spacecraft level as part of each unit's integration procedure.





**Figure 2.5-7. Inrush Current Transient Default Limit**

Test Method: There is no MIL-STD-461F equivalent for this requirement.

The basic test setup using a current probe is shown in Figure 2.5-8; an alternate test setup that measures the voltage drop across a series resistor is shown in Figure 2.5-9.

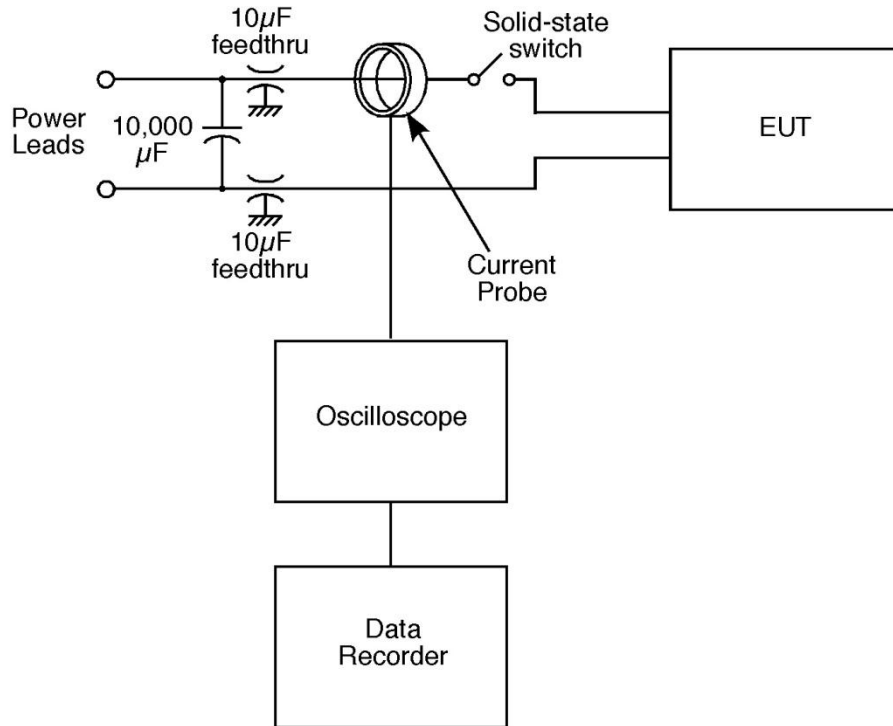
EUT Test Procedure:

- 1) Configure the test equipment as shown in Figure 2.5-8 if using the current probe method or Figure 2.5-8 if using the series resistor method.
- 2) Set up oscilloscope initially as defined in the calibration procedure with the time scale set to 50 μs/div for the first sweep.
- 3) Energize switch and record waveform of turn-on transient. If entire waveform is not recorded on-screen, due to either amplitude or duration, readjust vertical sensitivity and/or horizontal time base as necessary to yield on-screen trace.
- 4) Perform additional sweeps with the oscilloscope time scale set to 500 μs/div and 2 ms/div.
- 5) Repeat measurement for all operational transients in addition to the turn-on transient. When making multiple inrush measurements, ensure that test sample power supply filters have time to adequately discharge between measurements.

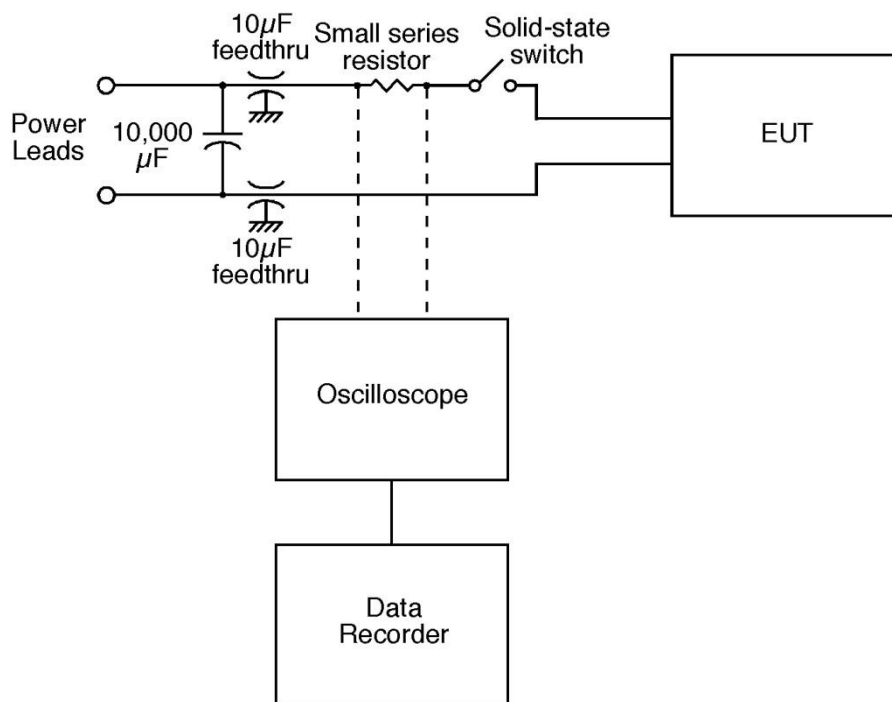
Data presentation:

Print out or screen dump of actual oscilloscope display, both for calibration and actual test waveforms. Entire waveform shall be displayed from one division before turn-on event to complete return to nominal conditions.

Additional procedure details and discussions are provided in Section 2.5.3.3.2.



**Figure 2.5-8. Inrush Current Emissions Test Setup (Current Probe)**



**Figure 2.5-9. Inrush Current Emissions Alternate Test Setup (Series Resistor)**

2.5.2.1.4 Conducted Emissions, Antenna Terminal

Applicability: This requirement is applicable to the antenna terminals of RF transmitters, receivers, and amplifiers.

Purpose: This purpose of this requirement is to ensure that emissions from the EUT do not interfere with other antenna-connected receivers on the platform.

Limit and Test Method: MIL-STD-461F CE106 limits and test method shall apply in frequency bands used by other antenna-connected receivers on the platform. For most GSFC platforms, this will consist of frequencies above 200 MHz (see section 2.5.2.3.2, RE102).

### 2.5.2.2 Conducted Susceptibility Requirements

Conducted susceptibility requirements on power leads, signal leads, and antenna terminals shall be applied to payload and spacecraft hardware as defined in the following subsections.

#### 2.5.2.2.1 Conducted Susceptibility, Power Leads, 30 Hz to 150 kHz

Applicability: This requirement is applicable to equipment operating from spacecraft primary power.

Purpose: The purpose of this requirement is to verify the ability of the EUT to withstand voltage ripple on the spacecraft primary power bus.

Limit: The voltage limit is shown in Figure 2.5-10. The corresponding power limit into a 0.5  $\Omega$  load from 30 Hz to 150 kHz is shown in Figure 2.5-11. These limits must be compared against the ripple characteristics of the spacecraft power subsystem and adjusted as necessary.

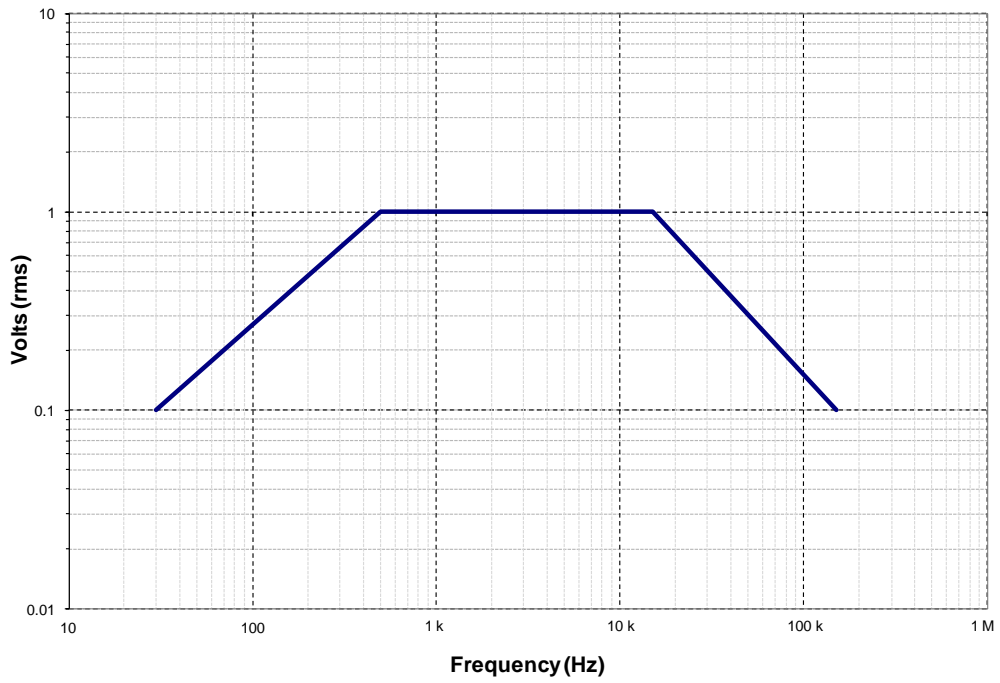
The limit for conducted susceptibility on power leads from 150 kHz to 50 MHz is defined in section 2.5.2.2.4.2 (CS114 test method).

Test method: The MIL-STD-461F CS101 test method may be applied.

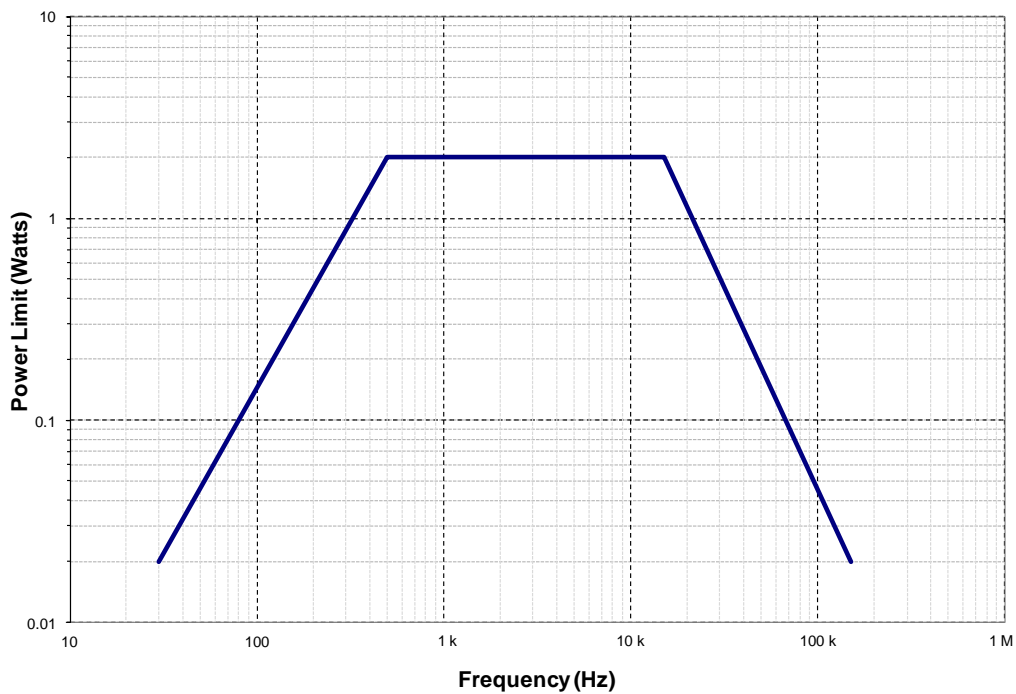
An alternate test method is discussed in Section 2.5.3.3.2.

**CAUTION!!!** When using the standard CS101 test method, the power amplifier driving the coupling transformer (shown in MIL-STD-461F Figure CS101-4) **MUST** be powered up and allowed to stabilize prior to applying power to the EUT. Failure to do so has been demonstrated to cause instability and damage to EUTs.

The requirement for conducted susceptibility on power leads from 150 kHz to 200 MHz is defined in section 2.5.2.2.4.2 (CS114 test method).



**Figure 2.5-10. Conducted Susceptibility Voltage Limit, (30 Hz to 150 kHz)**



**Figure 2.5-11. Conducted Susceptibility Power Limit (30 Hz to 150 kHz)**

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

#### 2.5.2.2.2 Conducted Susceptibility, Antenna Terminals

Applicability: These requirements are applicable to antenna terminals of radio frequency (RF) receivers as specified in the individual procurement specification.

Purpose: The purpose of these requirements is to verify that the EUT rejects out-of-band RF signals from other antenna-connected transmitters on the platform.

Limit and Test Methods: MIL-STD-461F CS103 (intermodulation) and CS104 (rejection of undesired signals) limits and test methods shall be applied. Where appropriate, testing may be limited to specific frequency bands used by antenna-connected systems on the spacecraft and launch vehicle.

Additional guidelines for defining mission-specific limits and test methods are provided on the [GSFC EMC Working Group wiki site](https://aetdwiki.gsfc.nasa.gov/display/EMCWG/Home) (<https://aetdwiki.gsfc.nasa.gov/display/EMCWG/Home>).

#### 2.5.2.2.3 Conducted Susceptibility, Transients, Power Leads

Applicability: This requirement is applicable to equipment operating from spacecraft primary power. It applies only to equipment operating from buses where equipment may be switched on and off the bus at full potential. This requirement does not apply to platforms where all loads are hardwired to the bus without a switch and are first energized as the bus is energized.

Purpose: The purpose of this requirement is to verify the ability of the EUT to withstand transients coupled onto its input power leads. Such transients include:

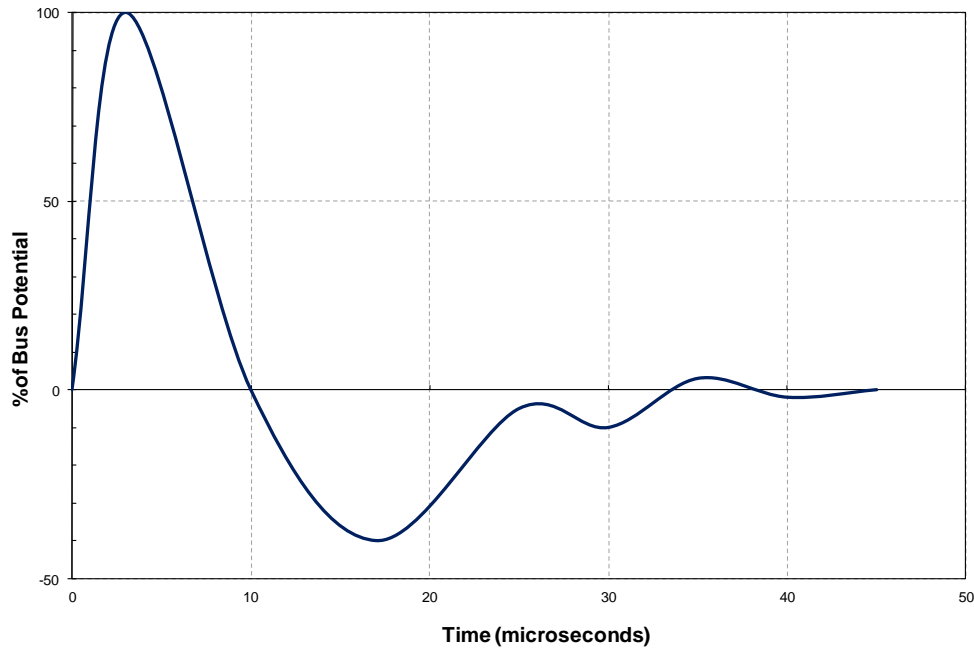
- Negative transients, which are generally due to equipment being switched onto the bus, in which the bus is pulled instantaneously to near zero volt potential and returns to nominal potential after a recovery time. Such transients are common to all platforms.
- Positive transients, which are generally associated with a back electromotive force (emf) from a rapidly changing inductive load (e.g. motor).

Limit: The limit is shown in Figure 2.5-12. This is the CS06 waveform defined by MIL-STD-461C. The pulse shall be applied at a minimum of once per second for one minute.

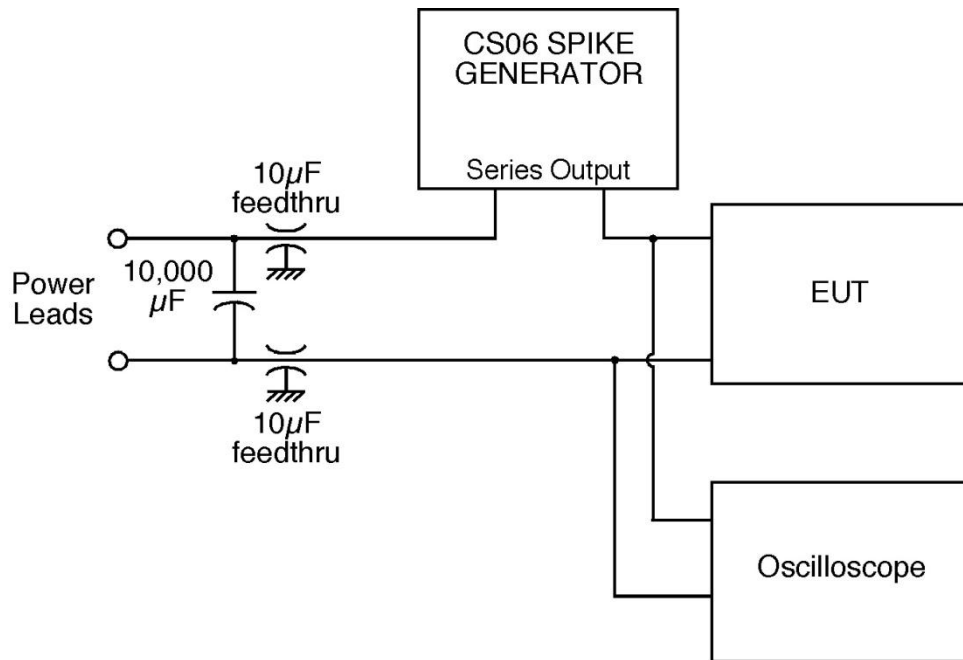
For all platforms, the negative transient shall be applied. For platforms that include rapidly changing inductive loads that are expected to generate significant back emf onto the bus, the positive transient shall be applied as well.

Test Method: The MIL-STD-462 CS06 test method may be applied as shown in Figure 2.5-13 and Figure 2.5-14. The series injection setup is recommended for most applications; for higher power loads, the parallel injection setup may be used. Note that if the parallel method is used, the standard source impedance capacitors should be removed in order to avoid loading down the CS06 generator.

Section 2.5.3.3.3 provides a discussion of the CS06 limit and test method as compared to the CS106 limit and test method defined by MIL-STD-461F.



**Figure 2.5-12. Conducted Susceptibility Transient Waveform**



**Figure 2.5-13. Conducted Transient Susceptibility Test Setup – Series Injection**

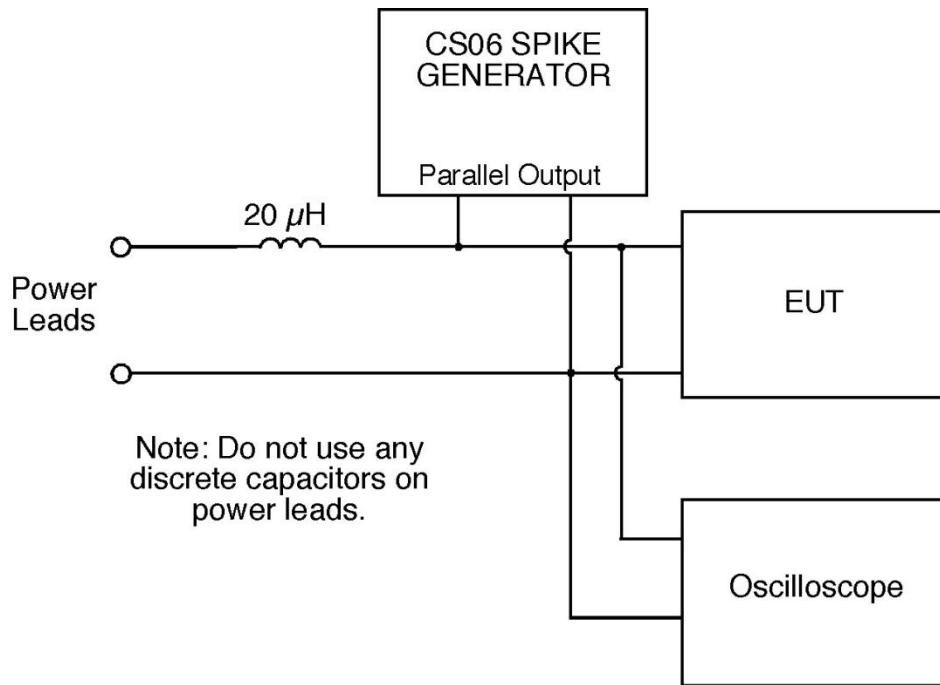


Figure 2.5-14. Conducted Transient Susceptibility Test Setup – Parallel Injection



2.5.2.2.4 Conducted Susceptibility, Bulk Cable Injection, 10 kHz to 200 MHz

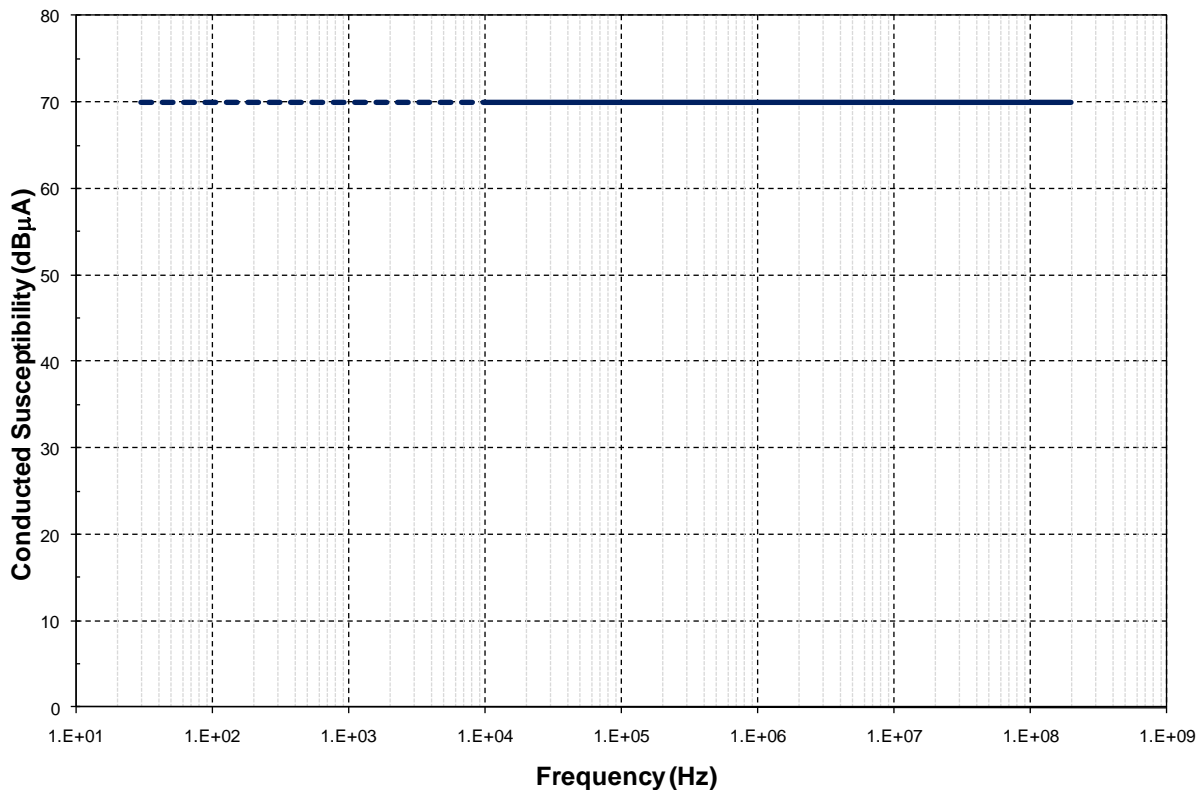
2.5.2.2.4.1 Conducted Susceptibility, Power and Signal Cables, Common Mode

Applicability: This requirement is applicable to all interconnecting power and signal cables.

Purpose: The purpose of this requirement is to verify the ability of the EUT to withstand common mode currents coupled onto its interface cables. This requirement is intended to be a companion to the Common Mode Conducted Emissions Requirement defined in Section 2.5.2.1.2 in order to mitigate the risk of crosstalk.

Limit: The calibration limit is shown in Figure 2.5-15. Note that per the MIL-STD-461F CS114 test method (see below), the injected level during EUT testing is 6 dB above this level. The default frequency range defined by the solid line, 10 kHz to 200 MHz, corresponds to the frequency range of the MIL-STD-461F CS114 limit curves. On some platforms, it may be desirable to extend the limit down to 30 Hz as shown by the dashed line. This would be particularly true for platforms using equipment sensitive to magnetic fields below 10 kHz. In addition, it may be desirable to evaluate susceptibilities to test environments that typically use many potential sources of noise at these lower frequencies (e.g. vacuum pumps, valves, vibration tables, etc.).

Test Method: The MIL-STD-461F CS114 test method for common mode injection shall apply from 10 kHz to 200 MHz. If the lower frequency limit is extended to 30 Hz as shown in Figure 2.5-15, the basic CS114 test method may still be used with an injection clamp capable of injecting the necessary current levels from 30 Hz to 10 kHz.



**Figure 2.5-15. Common Mode Conducted Susceptibility Calibration Limit**

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

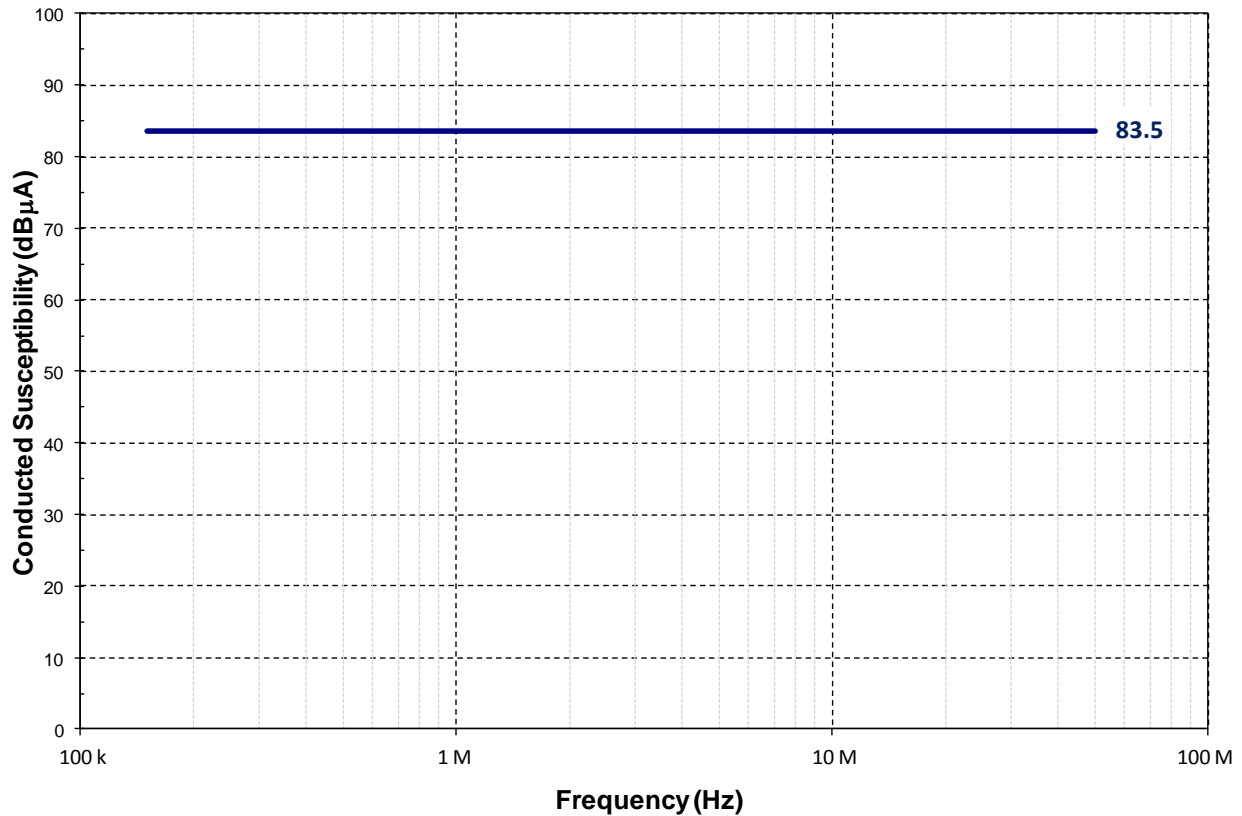
2.5.2.2.4.2 Conducted Susceptibility, Power Leads, 150 kHz to 50 MHz

Applicability: This requirement is applicable to equipment operating from spacecraft primary power.

Purpose: The purpose of this requirement is to verify the ability of the EUT to withstand ripple on the spacecraft primary power bus.

Limit: The calibration limit is 83.5 dB $\mu$ A (15 mA rms) from 150 kHz to 50 MHz as shown in Figure 2.5-16. Note that per the MIL-STD-461F CS114 test method (see below), the injected level during EUT testing is 6 dB above this level.

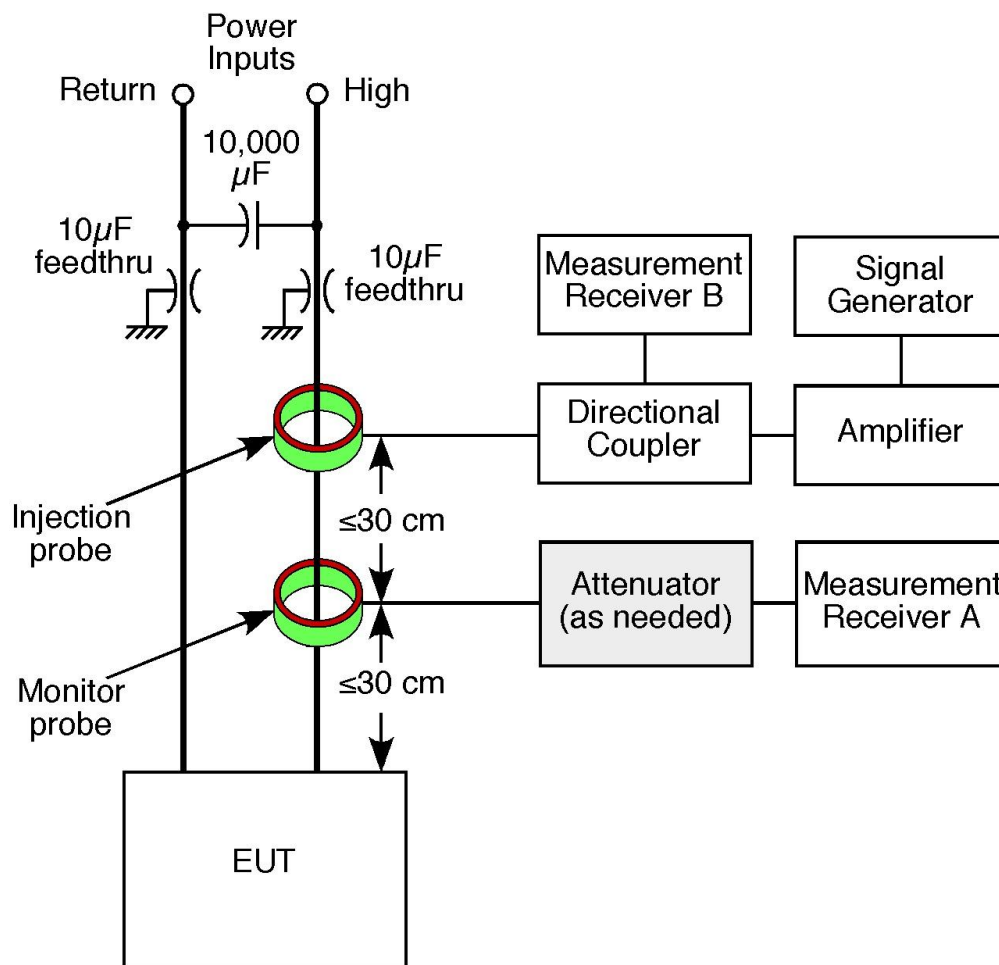
Conducted Susceptibility on power leads from 30 Hz to 150 kHz is addressed in Section 2.5.2.2.1.



**Figure 2.5-16. Conducted Susceptibility, Power Leads, 150 kHz to 50 MHz Limit**

**Test Method:** The MIL-STD-461F CS114 test method for injection onto power leads shall apply (per MIL-STD-461F section 5.13.3.4.c) with the following modifications:

- Replace the LISN with the capacitor network described in section 2.5.1.6.2 and as shown in Figure 2.5-17.
- Place the monitor probe  $\leq 30$  cm from the EUT, and place the injection probe  $\leq 30$  cm of the monitor probe as shown in Figure 2.5-18. This places both probes within  $\lambda/10$  of the EUT at the upper frequency limit of 50 MHz.
- Perform the scan from 150 kHz to 50 MHz.



**Figure 2.5-17. Conducted Susceptibility, Power Leads Test Setup**

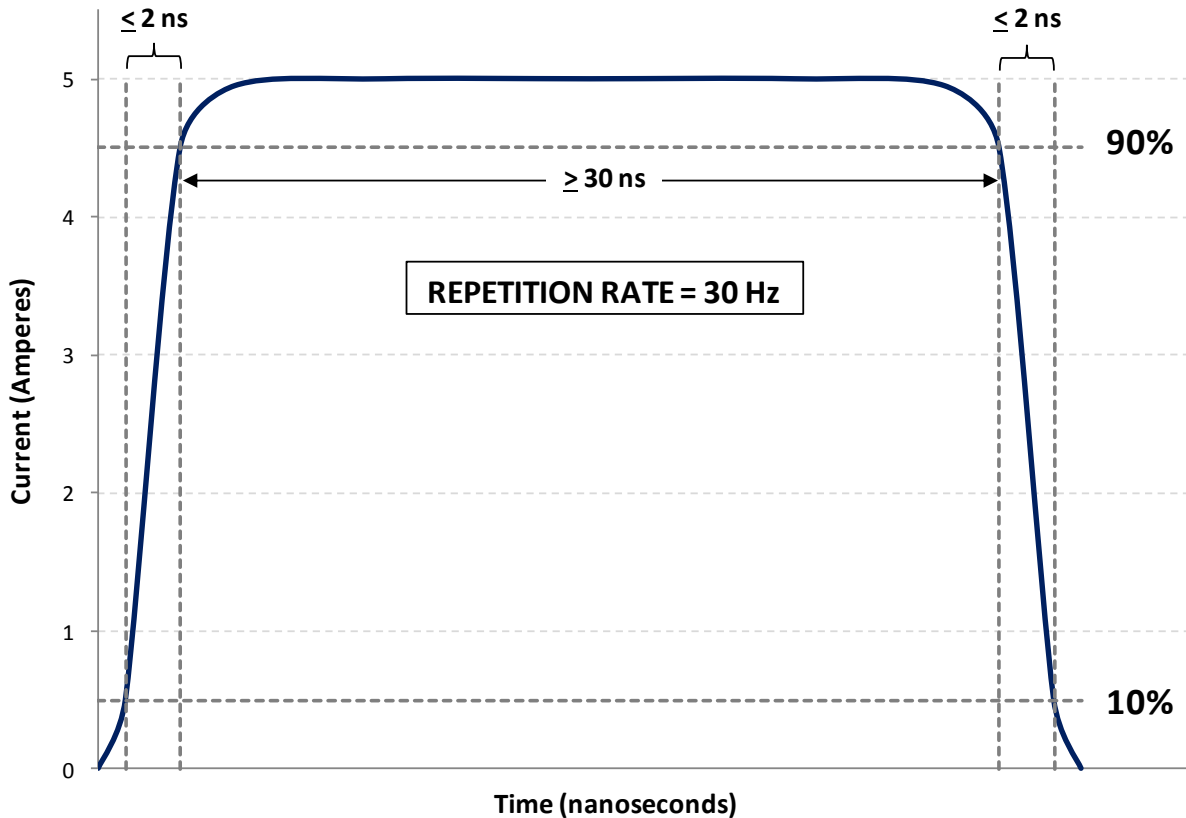
2.5.2.2.5 Conducted Susceptibility, Bulk Cable Injection, Impulse Excitation

Applicability: This requirement is applicable to all interconnecting power and signal cables that may be subjected to impulse signals.

Purpose: This test procedure is used to verify the ability of the EUT to withstand impulse signals coupled onto EUT associated cabling. This test may be used to provide time domain information in addition to the frequency domain information provided by the tailored CS114 test described in the previous section.

Test Method: MIL-STD-461F CS115 test method shall apply.

Limit: The CS115 impulse defined in MIL-STD-461F may be applied. This waveform is reproduced in Figure 2.5-18. The amplitude, rise time, fall time, duration, and repetition rate of the applied impulse should be tailored according to the expected impulse amplitudes on the platform.



**Figure 2.5-18. CS115 Default Limit**

### 2.5.2.3 Radiated Emissions Requirements

#### 2.5.2.3.1 Radiated Emissions, Magnetic Field

Applicability: This requirement is applicable to equipment on platforms using devices that are inherently susceptible to magnetic fields either by design or because they process very low-level signals.

Purpose: The purpose of this requirement is to verify that the magnetic field emissions from the EUT and its associated electrical interfaces do not exceed specified requirements.

Test Method: MIL-STD-461F RE101 test method, which measures the magnetic field at a distance of 7 cm from the EUT, may be applied. For some platforms, it may be desirable to also measure the magnetic field at a distance of 1 m from the EUT as specified by the RE04 test method from MIL-STD-462, Notice 2. The 1 m measurement should be performed with a sensor with comparable sensitivity and frequency range to that of the most sensitive potential victim on the platform.

Limit: The limit must be defined and tailored in order to protect equipment on the platform with known sensitivities to magnetic fields. For platforms using such equipment, it is recommended that the project create and maintain a Magnetics Control Plan that defines its magnetic environment and its requirements that have been tailored to that environment. The project may elect to include the Magnetics Control Plan as part of its EMCCP or provide it as a separate document. This plan should be updated and maintained throughout the program.

Guidelines for defining magnetic requirements are provided in “The Design, Construction and Test of Magnetically Clean Spacecraft - A Practical Guide (Mario H. Acuña)”.

#### 2.5.2.3.2 Radiated Emissions, Electric Field

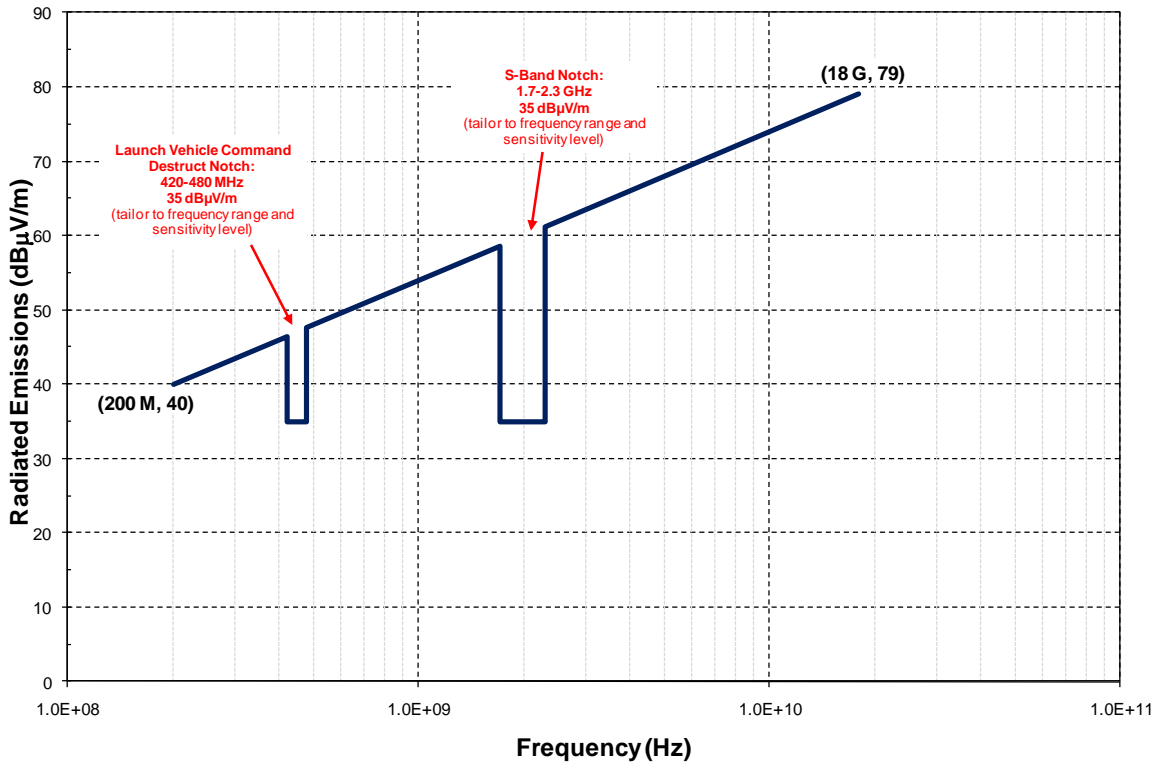
Applicability: This requirement is applicable for radiated emissions from equipment and subsystem enclosures, all interconnecting cables, and antennas designed to be permanently mounted to EUTs (receivers and transmitters in standby mode). The requirement does not apply at the transmitter fundamental frequencies and the necessary occupied bandwidth of the signal.

Purpose: The purpose of this requirement is to verify that electric field emissions from the EUT and its associated cabling do not interfere with on-board RF receivers. This requirement is not intended to verify compatibility with neighboring non-RF equipment that has been tested for RS103 that addresses radiated susceptibility to electric fields. While RE102 is concerned with potential effects with antenna-connected receivers, RS103 simulates fields resulting from antenna-connected transmitters. Compatibility between non-RF components and subsystems (i.e. crosstalk) is addressed by the Common Mode Conducted Emissions requirements defined in Section 2.5.2.1.2 and Common Mode Conducted Susceptibility requirements defined in Section 2.5.2.2.4 (CS114) and Section 2.5.2.2.5 (CS115).

Limit: The default limit is shown in Figure 2.5-19.

Test Method: The test method of MIL-STD-461F RE102 may be applied.

Guidelines for tailoring the limit and test method are provided in Section 2.5.3.3.4.



**Figure 2.5-19. Radiated Emissions Electric Field Limit with Suggested Notches**

2.5.2.4 Radiated Susceptibility Requirements

2.5.2.4.1 Radiated Susceptibility, Magnetic Field

Applicability: This requirement is applicable to equipment on platforms using devices that are inherently susceptible to magnetic fields because they process very low-level, low frequency signals.

Purpose: The purpose of this requirement is to verify the ability of the EUT to withstand radiated magnetic fields.

Limit and Test Method: MIL-STD-461F RS101 test method may be applied for AC magnetic fields. The limits must be defined and tailored according to the equipment used on the platform and documented in the project Magnetics Control Plan (see Section 2.5.2.3.1).

For platforms using equipment sensitive to DC magnetic fields, the test methods as well as the limits defined and documented in the project Magnetics Control Plan.

Guidelines for defining magnetic requirements are provided in “The Design, Construction and Test of Magnetically Clean Spacecraft - A Practical Guide (Mario H. Acuña)”.

#### 2.5.2.4.2 Radiated Susceptibility, Electric Field

Applicability: This requirement is applicable to equipment and subsystem enclosures and all interconnecting cables.

Purpose: The purpose of this requirement is to verify the ability of the EUT and associated cabling to withstand electric fields generated from intentional RF transmitters during all phases of the project, including on orbit, on the launch vehicle, at the launch site, and all test environments. This requirement is not intended to verify compatibility with neighboring non-RF equipment that has been tested for RE102 that addresses electric field radiated emissions. While RE102 is concerned with potential effects with antenna-connected receivers, RS103 simulates fields resulting from antenna-connected transmitters. Compatibility between non-RF components and subsystems is addressed by the Common Mode Conducted Emissions and Bulk Cable Injection Conducted Susceptibility (CS114 test method) requirements.

Test Method: The MIL-STD-461F RS103 test method shall apply.

Limit: Equipment that must operate at launch shall be tested, at minimum, to the following levels:

- 20 V/m from 2 MHz to 18 GHz

These limits are based on typical fields from transmitters operating on the launch vehicle and at the launch site. Specific launch vehicles and launch sites may impose different limits. Each project must assess its compatibility to these environments once a launch vehicle and launch site are selected. Further guidance relevant to Launch Vehicle & Launch Site Electromagnetic Environments can be found in the following document:

- ELVL-2010-0042300 Electromagnetic Environments – A Guideline for Spacecraft Launching from Eastern, Western and Pacific Ranges

In addition, the Interface Control Document (ICD) and/or User's Guide for the appropriate launch vehicle must be consulted for further guidance.

Equipment that will not be powered on during launch shall be tested, at minimum, to the following levels:

- 2 V/m from 2 MHz to 18 GHz

These limits are based on the typical on-orbit environment created by the transmitters on board the spacecraft. Specific platforms may impose different limits. Each project must assess its compatibility to the environments defined in the appropriate platform-specific documentation, and the limits must be tailored accordingly.

Based on data and analysis compiled by the Department of Defense Joint Spectrum Center, the on-orbit RF environment can likely exceed 2 V/m in specific frequency bands due to fields transmitted from high gain antennas on the ground or on communications satellites. Susceptibility test levels must be derived from the expected mission environment. Guidance relevant to on-orbit RF environments are provided in the following document:

- JSC-CR-06-070 Space Vehicle RF Environments

Equipment that will not be powered on during launch, but that may be powered on during any portion of the test campaign at the launch site, should be tested to the launch site levels in addition to the on-orbit levels. If no susceptibility is observed at the launch site levels, then the equipment may be powered on during the launch site test campaign without restrictions. However, if a susceptibility is observed at the launch site levels but not at the on-orbit levels, the project must assess the risk of interference at the launch site (i.e. fix the interference problem) vs. the need to power up the equipment during the test campaign (i.e. keep the equipment powered off at the launch site).

### 2.5.3 Application Guide

This section, along with the Appendix of MIL-STD-461F, provides background information for each emission and susceptibility and associated test requirement in the previous sections of this document. This information includes rationale for requirements, guidance in applying the requirements, and lessons learned from platform and laboratory experience. This information should help users understand the intent behind the requirements, should aid the procuring activity in tailoring emission and susceptibility requirements as necessary for particular applications, and should help users develop detailed test procedures in the EMITP based on the general test procedures in this document. This section is provided for guidance purposes and, as such, should not be interpreted as providing contractual requirements.

#### 2.5.3.1 Engineering Concerns for EMI Testing

Each EMI test addresses at least one of the following concerns:

- Compatibility with the spacecraft power subsystem
- Compatibility with the spacecraft communications subsystem
- Compatibility between subsystems (crosstalk)

The EMI tests specified by MIL-STD-461 primarily address the first two concerns with little to no emphasis on the third. Compatibility with the spacecraft power subsystem is addressed through Conducted Emissions (CE101, CE102) and Conducted Susceptibility (CS101) requirements; compatibility with the spacecraft communications subsystem is addressed through Electric Field Radiated Emissions (RE102) and Electric Field Radiated Susceptibility (RS103) requirements.

Regarding RE102 and RS103, Section A.5.17 of MIL-STD-461F states: "There is no implied relationship between (RE102) and RS103 that addresses radiated susceptibility to electric fields. Attempts have been made quite frequently in the past to compare electric field radiated emission and susceptibility type requirements as a justification for deviations and waivers. While RE102 is concerned with potential effects with antenna-connected receivers, RS103 simulates fields resulting from antenna-connected transmitters." In other words, RE102 and RS103 are specifically designed to assess compatibility with the communications subsystem. They are not designed to address crosstalk between subsystems.

When an EUT meets these requirements on a military platform, there is generally low risk of crosstalk. This is primarily because RE102 limit corresponds to common mode current levels on culprit cables that are too low to pose a significant risk. Outside the frequency bands of antenna-connected receivers, the RE102 levels (and those of its MIL-STD-461C



predecessor, RE02) are significantly more stringent than necessary to control crosstalk. This is precisely the reason that RE02 and RE102 exceedances have historically been the largest sources of waivers on many GSFC platforms.

Section A.5.17 of MIL-STD-461F provides the following tailoring guideline: "It may be desirable to tailor the frequency coverage of the limit to include only frequency bands where antenna-connected receivers are present." Most GSFC platforms do not use the antenna-connected receivers that operate at frequencies below 200 MHz. Following this guideline, RE102 is only applicable above 200 MHz on most GSFC platforms as discussed in Section 2.5.2.3.2.

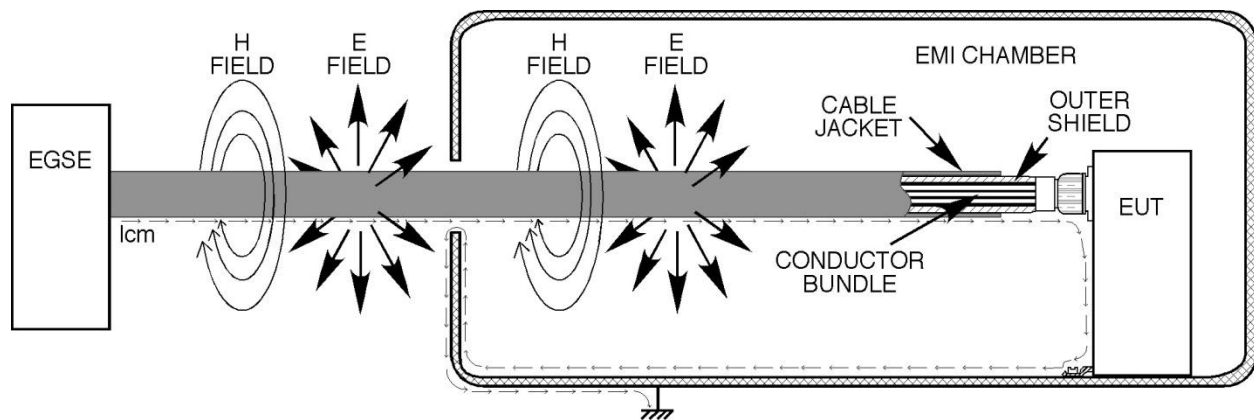
Below 200 MHz, RE102 is replaced with a measurement of Common Mode Conducted Emissions (CMCE, no MIL-STD-461F equivalent) on power and signal cables as defined in Section 2.5.2.1.2. This requirement is intended to be a companion to the Common Mode Conducted Susceptibility (CMCS, MIL-STD-461F CS114 test method) requirement as defined in Section 2.5.2.2.4. The results of these two tests enable a significantly more direct and efficient assessment of cable-to-cable crosstalk than is possible using RE102 and RS103.

The approach outlined above is adapted from the tailored RE102 requirements and test methods defined in SL-E-0002, Space Shuttle Specification, Electromagnetic Interference Characteristics, Requirements for Equipment. This specification replaces the traditional RE102 test method below 200 MHz with a Bulk Cable Emission (BCE) measurement for all of the reasons stated above. This approach has been demonstrated to control cable-to-cable crosstalk more effectively and efficiently while at the same time eliminating the paperwork associated with waivers against an inapplicable and unnecessarily stringent RE102 limit below 200 MHz.

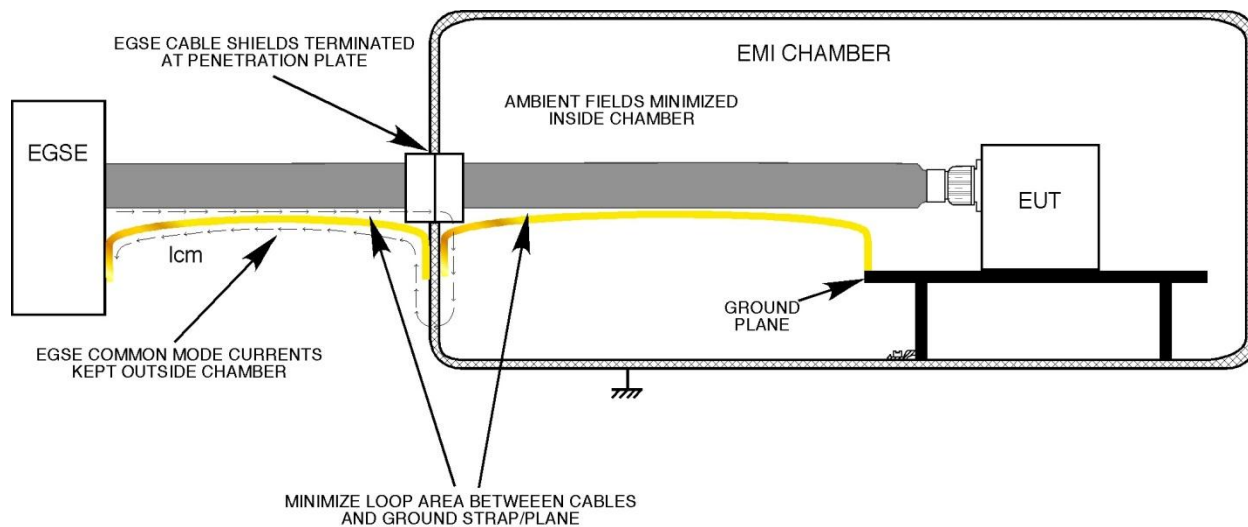
2.5.3.2 Test Facility and Procedure Requirements

2.5.3.2.1 Ambient Electromagnetic Level

Noise from Electrical Ground Support Equipment (EGSE) is one of the most common contributors to high ambient levels. If the EGSE cables penetrate the chamber wall in an uncontrolled manner as shown in Figure 2.5-20, common mode currents are allowed to enter the chamber along with their resulting electric and magnetic fields. These fields will raise the ambient electromagnetic field level, quite possibly to a level that violates the requirement stated in Section 2.5.1.5.1.



**Figure 2.5-20. Impacts of Uncontrolled EGSE Common Mode Currents**



**Figure 2.5-21. Proper EGSE Shielding and Grounding Configuration**

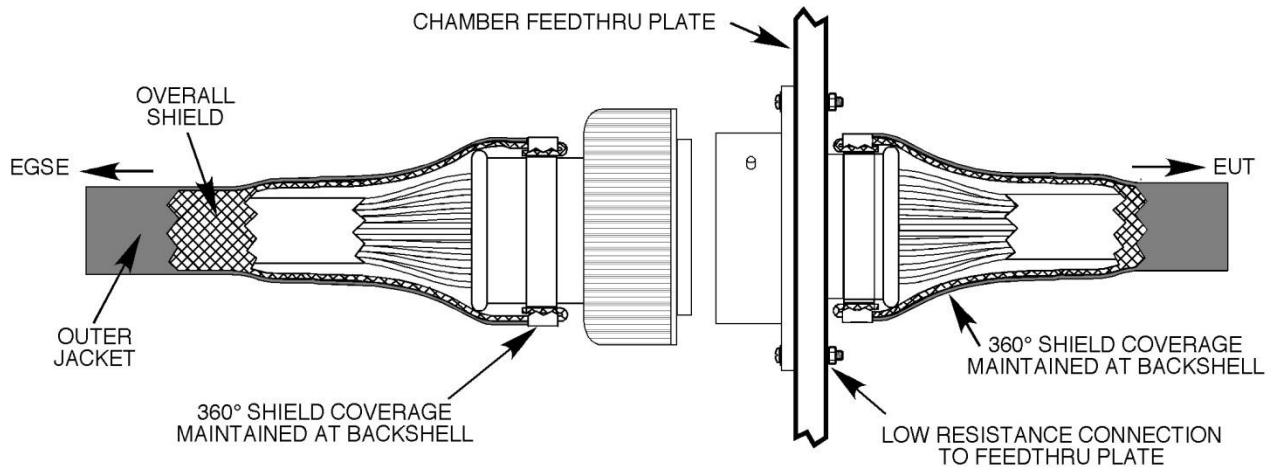
In order to prevent these uncontrolled currents and fields from entering the chamber, all outer shields of all EGSE cables should be terminated at the chamber wall, and a single point ground should be established at the feedthrough panel as shown in Figure 2.5-21. Common mode currents on the EGSE cables are thereby prevented from entering the chamber and do not contribute to the ambient level.

The preferred method for terminating the shields is to provide a connector break at the chamber feedthrough panel as shown in Figure 2.5-22. The outer shield of each cable is connected directly to its backshell with 360-degree coverage, and the mating connector is connected directly to the feedthrough panel, preferably with an EMI gasket.

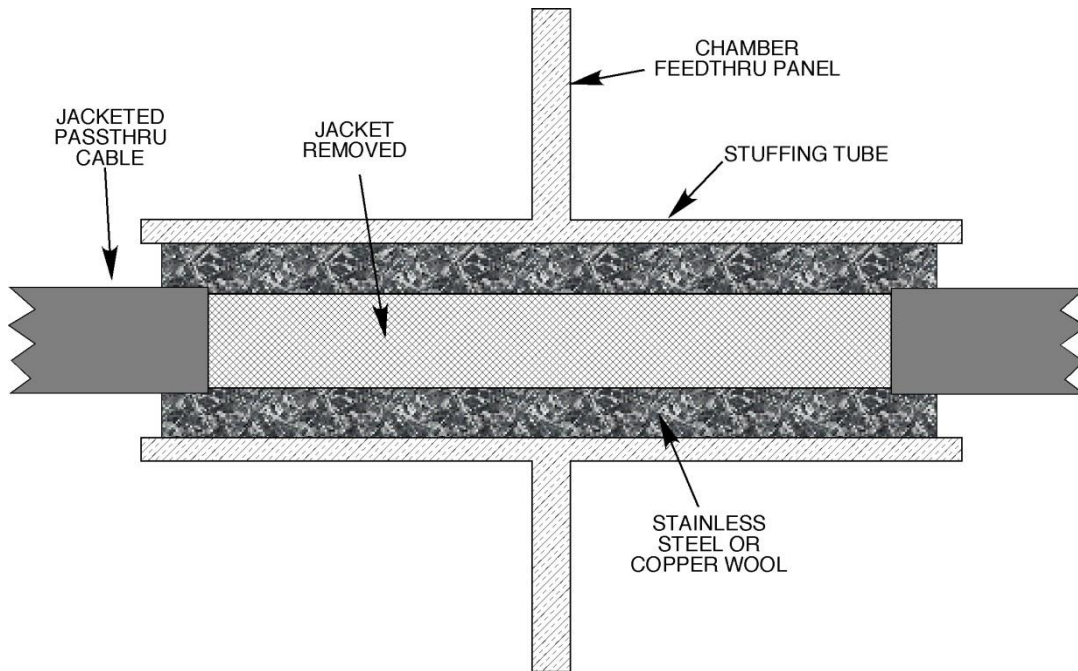
An alternate method is to run the cables through a stuffing tube as shown in Figure 2.5-23. The outer jacket of the cables inside the stuffing tube must be removed in order to expose the shield, and the tube should be filled with stainless steel, bronze, or copper wool in order to provide electrical continuity (bonding) between the shield braid(s) and the inside of the stuffing tube wall.

For EMI test facilities in which contamination and cleanliness are a significant concern, the stuffing tube may not be used. These facilities must implement the connector break and terminate the shields at the feedthrough panel.

Additional guidance for grounding, bonding, and shielding for facilities and test configurations is provided in MIL-HDBK-419A, Grounding, Bonding, and Shielding for Electronic Equipments and Facilities (Volumes 1 & 2, 29 December 1987).



**Figure 2.5-22. EGSE Cable Shield Termination – Connector Break (Preferred Method)**



**Figure 2.5-23. EGSE Cable Shield Termination – Stuffing Tube (Alternate Method)**

### 2.5.3.2.2 Power Source Impedance

MIL-STD-461F specifies the use of Line Impedance Stabilization Networks (LISNs) in series with the power leads to the EUT. The purpose of the LISNs is to provide a standardized power bus source impedance to represent expected impedances in actual installations and to ensure consistent results between different test facilities. The LISN is intended to simulate the common source impedance between the spacecraft power supply and the common distribution point as shown in Figure 2.5-24. It is not intended to include the harness impedance between the common distribution point and the load; this portion of the impedance is most effectively simulated using flight representative cables. LISNs are discussed in further detail in MIL-STD-461F section A.4.3.6.

MIL-STD-461F Section 4.3.6 specifies a default 50  $\mu\text{H}$  LISN. This LISN simulates worst case power distribution wiring on large military platforms as shown in Figure 2.5-25. On such platforms, power is distributed as a single wire 5 cm above structure (chassis), and structure is used as the current return path. This wiring configuration has an inductance on the order of 1  $\mu\text{H}/\text{m}$ . In the worst case, the wiring may run as much as 50 meters to the common distribution point, which gives the worst case total inductance of 50  $\mu\text{H}$  specified in the default MIL-STD-461F LISN.

On the majority of GSFC platforms, the power distribution wiring more closely resembles that shown in Figure 2.5-26. The power source impedance consists of a battery-dominated bus charged from solar cells, with the point of distribution (fuse block or other circuit protection devices) mounted in the immediate vicinity of the battery (< 1 m), providing for very low wiring contribution to the common or shared bus impedance. In addition, the chassis is almost never used as a deliberate current return path. A dedicated return wire is generally twisted with the “send” or “hot” wire, which results in a lower inductance (< 0.5  $\mu\text{H}/\text{m}$  typical). For such platforms, the power source impedance is significantly less than that simulated by the 50  $\mu\text{H}$  LISN. The impedance is more accurately simulated by the capacitor network shown schematically in Figure 2.5-27, which consists of a 10,000  $\mu\text{F}$  line-to-line capacitor, shunted by a pair of 10  $\mu\text{F}$  feedthrough capacitors installed between each power conductor and the ground plane.

The 10  $\mu\text{F}$  feedthrough capacitors form an integral part of the standard setup specified by MIL-STD-462 and are part of the standard suite of equipment available in any EMI laboratory. These feedthrough capacitors provide a standard source impedance at frequencies above approximately 20 kHz. The 10,000  $\mu\text{F}$  line-to-line capacitor extends the standardized source impedance down to 30 Hz in order to cover the full range of frequencies specified by the tests in this document. For all MIL-STD-461F based test methods specified in this document, the LISNs will be replaced by this capacitor network.

If a given GSFC program identifies a need to use a LISN, it is recommended to follow the tailoring guidelines of MIL-STD-461F section A.4.3.6, which specifies a 5  $\mu\text{H}$  LISN as shown in MIL-STD-461F Figure A-2 (schematic) and MIL-STD-461F Figure A-3 (impedance). If the flight power distribution scheme is well defined and the 5  $\mu\text{H}$  LISN is also unrealistically high, a custom LISN or flight representative cables may be used. The specific LISN or test cables used shall be specified in the project EMCCP along with supporting rationale.

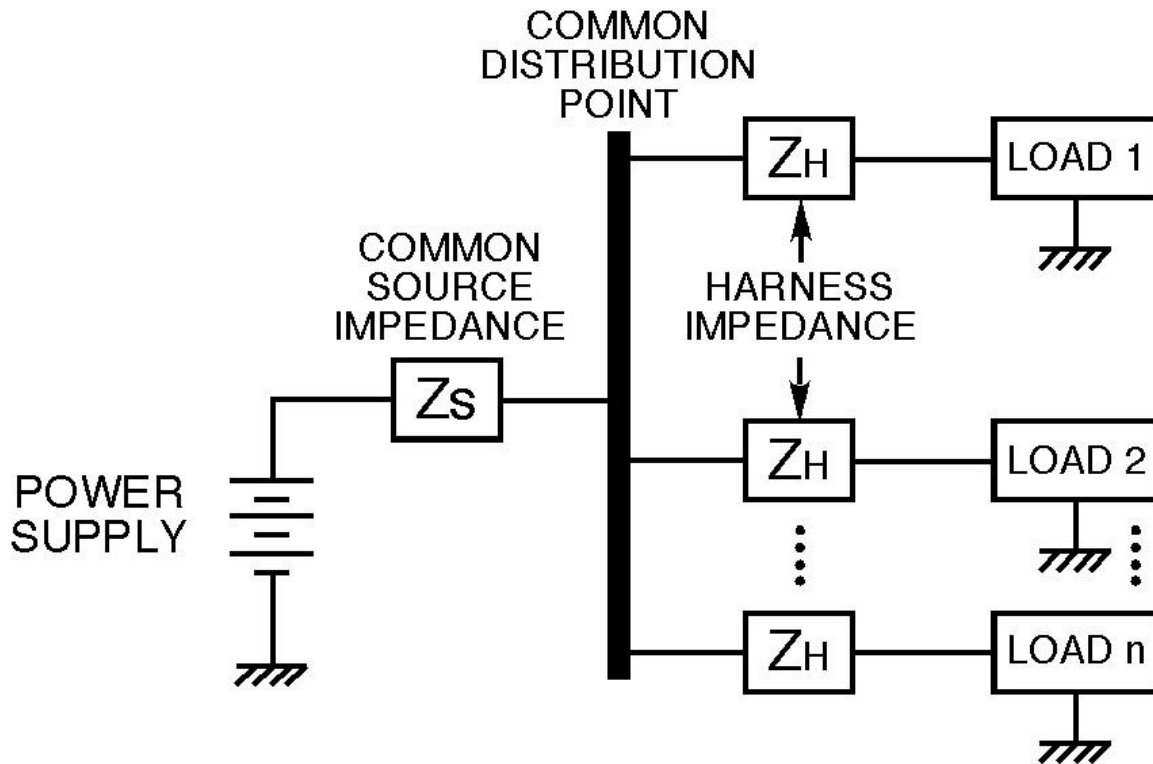


Figure 2.5-24. Common Source Impedance and Harness Impedance (General)

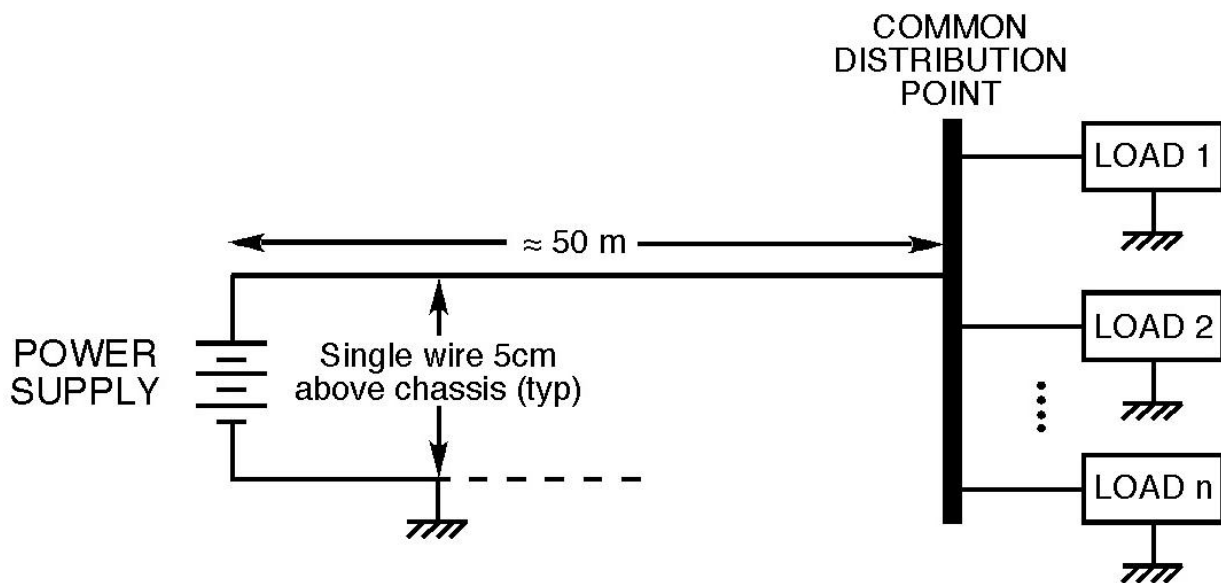


Figure 2.5-25. Power Distribution Wiring, Large Military Platforms (Worst Case)

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

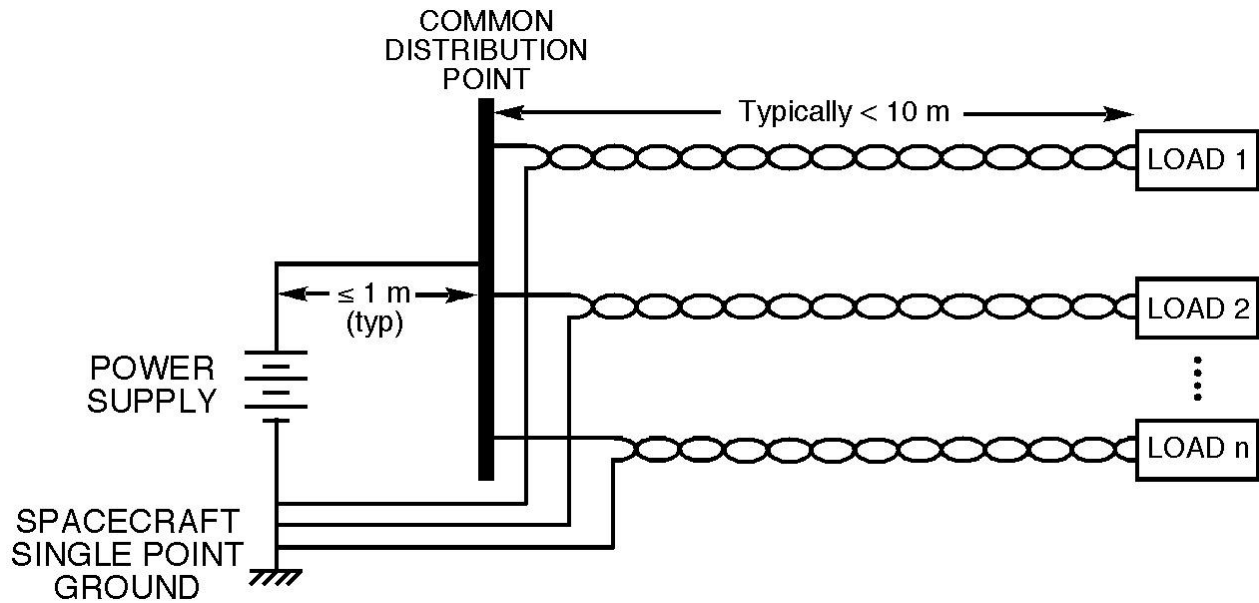


Figure 2.5-26. Power Distribution Wiring, GSFC Platforms (Typical)

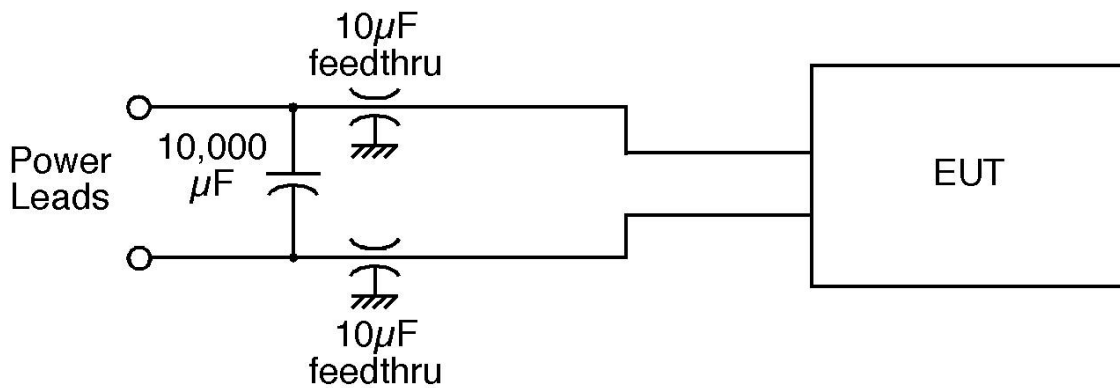


Figure 2.5-27. Capacitor Network Simulating Battery Dominated Bus Impedance

2.5.3.2.3 Construction and Arrangement of EUT Cables

As stated in Section 2.5.1.5.2 of this document, the power source impedance for the tests in this document will be simulated by the capacitor network shown in Figure 2.5-1. The 10  $\mu$ F feedthrough capacitors are an integral part of the standard setup specified by MIL-STD-462. The CE01 and CE03 test methods of MIL-STD-462, the predecessors of CE101 and CE102, specify that the “length of power lead from the test sample to the feedthrough capacitor shall not exceed 1 meter.” This clearly conflicts with the following requirement in MIL-STD-461F Section 4.3.8.6.2: “Two meters of input power leads (including neutrals and returns) shall be routed parallel to the front edge of the setup in the same manner as the interconnecting leads.”

In order to minimize the number of power cables needed for the EMI tests, it is recommended to use the 1 meter length as specified by MIL-STD-462. The rationale for this recommendation is provided below.

The purpose of the 1 m maximum length specified by MIL-STD-462 is to minimize standing waves by ensuring that the cables are electrically short (less than a significant fraction of a wavelength) over the CE01 and CE03 frequency range. The wavelength  $\lambda$  in meters of an electromagnetic wave is given by:

$$\lambda = \frac{c}{f}$$

where:

$c$  = speed of light =  $3 \times 10^8$  m/s

$f$  = frequency

Wavelength is alternately expressed as follows:

$$\lambda = \frac{300}{f_{MHz}}$$

At the CE03 upper frequency limit of 50 MHz,  $\lambda = 6$  m. The 1 m cable is a sixth of a wavelength ( $\lambda/6$ ), which is sufficiently electrically short through this frequency range.

The purpose of the 2 m minimum length specified by MIL-STD-461F is to ensure that radiated emissions and susceptibility testing properly assesses the performance of the EUT along with all of its interconnecting cables. In order for the interconnecting cables to be properly assessed, they must be electrically long (greater than a significant fraction of a wavelength) over most of the frequency range covered by the tests. This is particularly important for the radiated electric field emissions (RE102) and susceptibility (RS103) tests, for which the nominal frequency range specified by MIL-STD-461F is 2 MHz to 18 GHz.

A 1 m cable is a tenth of a wavelength ( $\lambda/10$ ) at 30 MHz, which means that it is electrically long over most of the nominal RE102 and RS103 frequency range. Moreover, as specified in section 2.5.2.3.2 of this document, radiated electric field measurements on most GSFC platforms will be performed only above 200 MHz (see discussion in Section 2.5.3.3.5). The 1 m cables will certainly be electrically long at these frequencies. Radiated electric field



emissions below 200 MHz are addressed by the Common Mode Conducted Emissions test on power and signal cables defined in section 2.5.2.1.2.

### 2.5.3.3 Detailed Requirements

#### 2.5.3.3.1 Conducted Emissions, Power Leads, Differential Mode

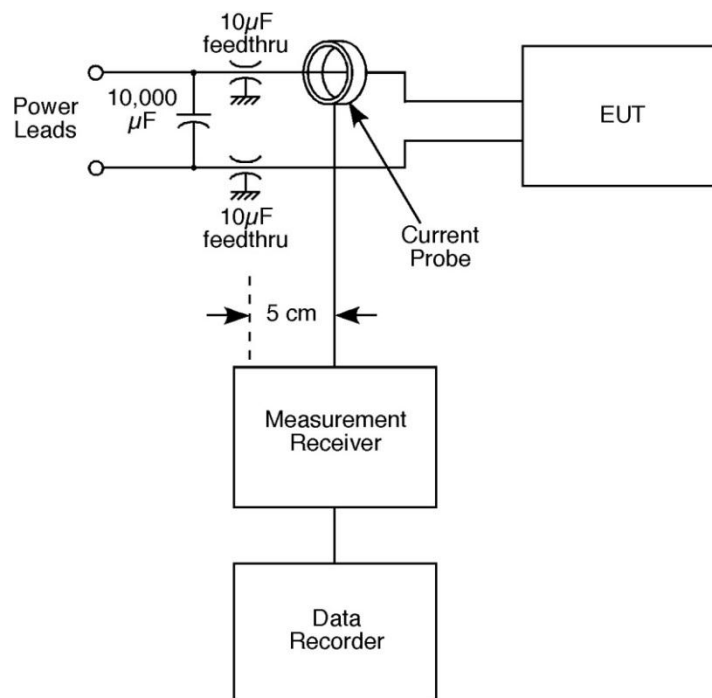
The CE101 test method (and its CE01 predecessor) is a single-ended line-to-ground current measurement as shown in Figure 2.5-28. This test method originates from military architectures in which power is distributed to each load in a single-ended manner and uses structure as the return path. This scheme is almost never used on GSFC platforms; dedicated return wires are almost always provided in present GSFC power distribution schemes.

These single-ended measurements combine differential mode (DM) and common mode (CM) information, which unnecessarily complicates the process of diagnosing and addressing the effects of the measured emissions. Separating the DM and CM measurements provides more directly meaningful limits and more useful information for diagnosing the source of a given emission and for assessing its potential impacts.

The MIL-STD-461F test method CE102 specifies a voltage measurement across a LISN. When conducted emissions are measured as a voltage, the results depend significantly on the power bus source impedance, which can vary from platform to platform. For this reason and for the reasons discussed above, a DM current measurement across the capacitor network is preferred as specified in Section 2.5.2.1.1. This test method provides a worst-case measurement of DM current emissions that is largely independent of the source impedance. The measured DM current may be directly compared to the platform's power bus impedance in order to assess the contribution to power bus voltage ripple.

This is effectively the basic MIL-STD-462 CE03 test method with the send and return wires run through the current probe for a DM measurement. It is also very similar to the tailored CE101 test method described above; the primary difference consists of the selection of a current probe that is suited to the higher (CE03) frequency range.

The Common Mode Conducted Emissions (CMCE) measurements are defined in Section 2.5.2.1.2.



**Figure 2.5-28. Conducted Emissions Test Setup – Single-Ended (Traditional)**

Because the DM test method measures twice the true DM mode current (6 dB above the true value), the measured limit must be set 6 dB above the true maximum DM mode current. The limits below 150 kHz are based on the CE101 tailoring guidelines in MIL-STD-461F Section A.5.4; the limits above 150 kHz are based on the CE03 limits of MIL-STD-461C.

The frequency breakpoint between the low and high frequency portions is 150 kHz. The limit at 150 kHz is fixed at 62 dBμA, which is 6 dB above the tailored CE101 limit at 150 kHz as specified in MIL-STD-461F Section A.5.4.

#### 2.5.3.3.2 Conducted Emissions, Common Mode, Power and Signal Lines

This test method addresses crosstalk at frequencies up to 200 MHz. In order to define a meaningful limit, the susceptibility of potential victim cables must be characterized first as specified by the Common Mode Conducted Susceptibility test (CS114) defined in Section 2.5.2.2.4. The Common Mode Conducted Emissions limits for neighboring culprit cables must be defined accordingly in order to protect against crosstalk to the most sensitive victim. If the susceptibility thresholds of the victim cables are not known at the time of the test, the default limit is shown in Figure 2.5-5.

Emissions above 200 MHz are addressed by the Radiated Electric Field Emissions (RE102) requirement defined in section 2.5.2.3.2. A strict comparison of these two limits reveals a discontinuity at 200 MHz. The common mode current corresponding to the RE102 limit at 200 MHz would be approximately 26 dB more stringent than that shown in Figure 2.5-5, which would be unnecessarily stringent for controlling crosstalk. The design of the EUT must account for this discontinuity and it must meet the more stringent RE102 limit at 200 MHz.

The test method from 150 kHz to 200 MHz requires the following test equipment:

- 1) Measurement receivers
- 2) Data recording device
- 3) Signal generators
- 4) Capacitor network consisting of two 10  $\mu$ F feedthrough capacitors and one 10,000  $\mu$ F capacitor (replacing the LISNs specified by SL-E-0002)
- 5) Current probe sensitive to frequencies below 150 kHz if the platform's magnetic requirements necessitate control of common mode currents at those frequencies
- 6) Absorbing clamp (per CISPR 16, Specification for radio disturbance and immunity measuring apparatus and methods). NOTE: Absorbing clamp must be calibrated as a current probe, not per CISPR 16.

The absorbing clamp (fully described in CISPR 16) is the central piece of equipment for this test. It is a current probe followed by a series of ferrite ring absorber elements. These act to isolate the rest of the cable, minimizing the standing waves associated with signals on an electrically long mismatched transmission line. If a traditional current probe were used for this test instead of an absorbing clamp, then at frequencies above which the Cable-Under-Test (CUT) were greater than one-tenth wavelength long (a 2 meter cable is  $\lambda/10$  at 15 MHz), it would be necessary to physically scan the probe the length of the CUT at each frequency in order to find a peak. The absorbing clamp can be set up once and left in place for the duration of the test, which will significantly reduce test time and produce more reliable and repeatable results. In order to get the full value of the absorbing clamp, it must be set up with the current probe end of the device within 15 cm ( $\lambda/10$  at 200 MHz) of the EUT.

In order to use the absorbing clamp for this test, with the current CE limit of this requirement, it must be calibrated as a current probe, not as an absorbing clamp per CISPR 16. The manufacturer may be requested to do this, or it can be done in-house.

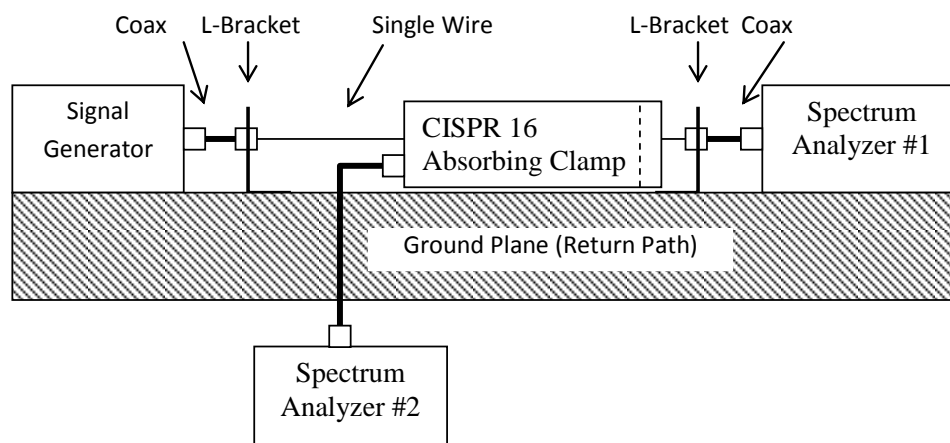
Calibration Procedure:

- 1) Set up the equipment as shown in Figure 2.5-29. This method uses two L-brackets with bulkhead mount BNC or N-type connectors on either side of the CISPR 16 absorbing clamp. The signal generator and spectrum analyzer attach to the connectors with coax. The center pins of the connectors are connected by a solid copper wire, 12 to 20 AWG in size, approximately one meter in length. The absorbing clamp is placed around the wire, with the current probe side of the clamp (indicated by dashed line) as close as possible to the spectrum analyzer, but in no case more than 15 cm away from the terminating end of the fixture where spectrum analyzer #1 is located. The outer conductor of the connector is bonded to the L-bracket, which is in turn bonded to the ground plane beneath the absorbing clamp.
- 2) Set the terminating spectrum analyzer (#1) to sweep from 150 kHz to 200 MHz, continuous sweep, max hold, and so that the data is measured in dB $\mu$ A, if possible. Otherwise, set it to read in dB $\mu$ V.

- 3) Connect the output of the absorbing clamp into spectrum analyzer #2 50 ohm input, and set this spectrum analyzer to sweep from 150 kHz to 200 MHz, continuous sweep, max hold, and so that the data is measured in dB $\mu$ V.
- 4) Set the signal generator output to -10 dBm. Sweep the signal generator from 150 kHz to 200 MHz, using a sweep time of between 100 ms and 500 ms.
- 5) If the terminating spectrum analyzer #1 is not able to display the data in dB $\mu$ A, the current into spectrum analyzer #1, and thus through the absorbing clamp current probe, can be calculated from the voltage read at spectrum analyzer #1 using the following relationship:

$$I \text{ (dB}\mu\text{A)} = V \text{ (dB}\mu\text{V)} - 34 \text{ dB}\Omega$$

- 6) Verify that the voltage measured by the absorbing clamp and displayed on spectrum analyzer #2 (in dB $\mu$ V) minus the detected current displayed on spectrum analyzer #1 (in dB $\mu$ A) results in a value that closely approximates the transfer impedance given by the curve in Figure 2.5-30. Retain the derived transfer impedance data for use in later calculations during the testing phase of the BCE procedure.



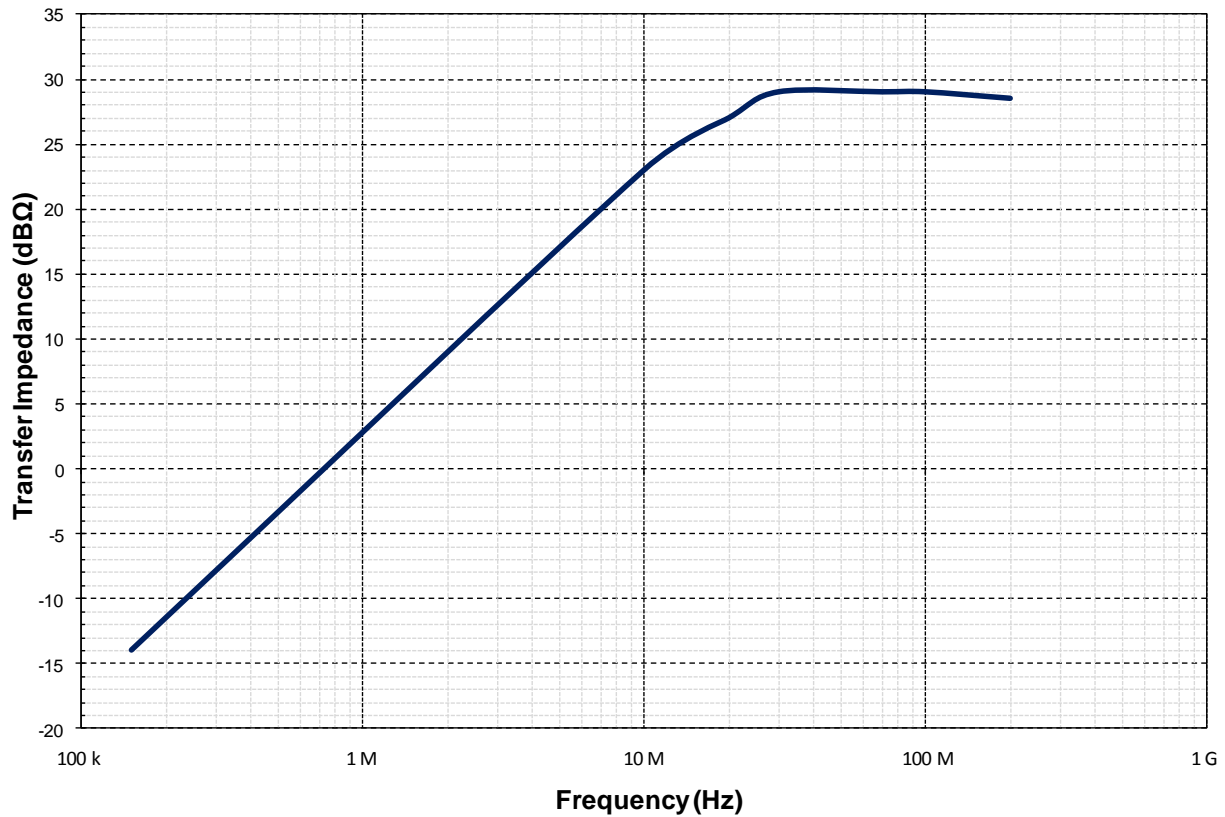
**Figure 2.5-29. Basic Common Mode Conducted Emissions Calibration Setup**

**Measurement System Integrity (MSI) Verification:**

Prior to performing the BCE test, it is necessary to verify measurement system integrity (MSI). MSI is checked at a minimum of three frequencies. The MSI check is conducted according to the steps below:

- 1) Set up the equipment in the same manner as the calibration procedure above. Spectrum Analyzer #2 is replaced by the measurement receiver and any controller/automation used during the BCE test. The measurement receiver is installed as it would be during the BCE test, with all interconnecting cables between it and the clamp as per the test set-up.
- 2) Apply a calibrated signal level, which is at least 6 dB below the applicable limit, at 3 frequencies in the range of the low frequency current probe and 3

- frequencies in the range of the absorbing clamp (150 kHz, 10 MHz, and 200 MHz).
- 3) Scan the measurement receiver for each frequency in the same manner as a normal data scan.
  - 4) Verify that the signal level indicated by the measurement system is within +/- 3 dB of the actual current level. If readings are obtained which deviate by more than  $\pm 3$  dB, locate the source of the error and correct the deficiency prior to proceeding with the testing.



**Figure 2.5-30. Nominal Transfer Impedance of CISPR 16 Absorbing Clamp**

#### Common Mode Conducted Emissions Testing:

- 1) Setup the test equipment as shown in Figure 2.5-6. Place the absorbing clamp around the cable, with the current probe side of the clamp as close as possible to the equipment under test but in no case more than 15 cm away from the EUT.
- 2) Scan the measurement receiver according to Table II of MIL-STD-461F. Take the voltage measured by the clamp (in dB $\mu$ V) minus the transfer impedance at that frequency (in dB $\Omega$ ) to determine the current level (in dB $\mu$ A). Compare this to the Common Mode Conducted Emissions limit. This may be the limit shown in Figure 2.5-5 or the tailored limit defined in order to protect potentially sensitive victim cables.

#### 2.5.3.3.3 Conducted Emissions, Time Domain, Transients

As stated in Section 2.5.2.1.3, one of the primary purposes of this requirement is to ensure that equipment inrush current due to turn-on and operational transients will not overstress its circuit protection device (e.g. fuse).

The spacecraft provider will allocate a size/value for each circuit protection device based on the peak power allocation of its corresponding unit. The intent of this requirement is to ensure that the unit's inrush transient is within the allowable envelope for the given circuit protection device, thus minimizing the likelihood of tripping the device during integration or during flight operations.

The envelope shown in Section 2.5.2.1.3 is a specific example based on the FM12 style fuse used on many platforms. This envelope must be compared against the characteristics of the specific circuit protection device(s) used on the platform and tailored as necessary.

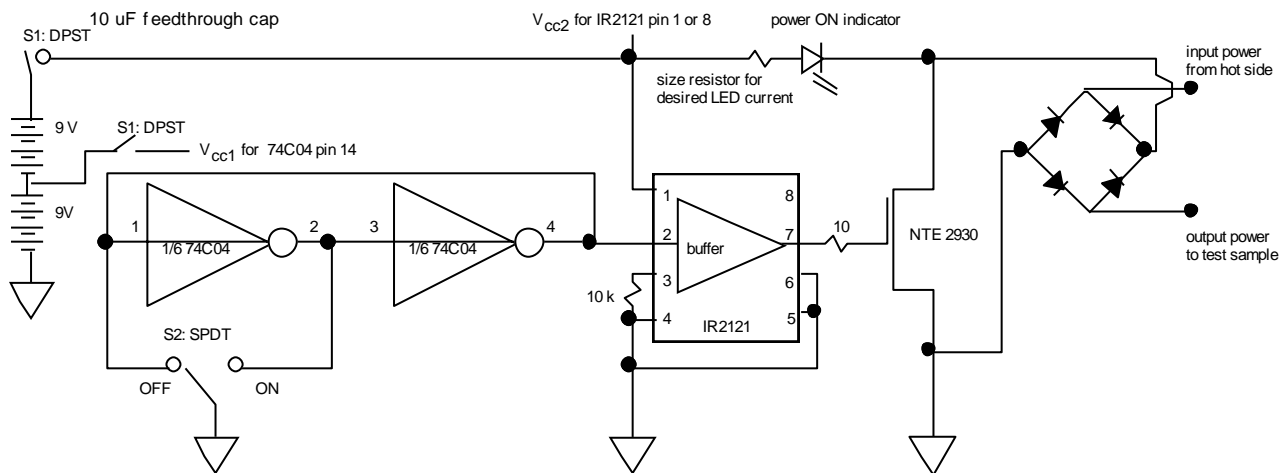
Given that this is a time domain measurement, a current probe test must be selected that has a flat transfer impedance at least from 20 Hz to 1 MHz in order to ensure accurate reproductions of waveforms with a resolution of approximately 1 microsecond.

Although Figure 2.5-7 shows the current probe located between the capacitor network and the switch, probe placement is not critical. The 1 m power cables are not a significant fraction of a wavelength at the frequencies of interest in this measurement (see Section 2.5.3.2.3 for the relationship between frequency and wavelength). At the upper frequency of 1 MHz indicated above,  $\lambda = 300$  meters. Even if a probe is used with flat response up to 10 MHz (giving a resolution of approximately 0.1 microsecond),  $\lambda = 30$  meters at this frequency. In either case, 1 m power cables are less than  $\lambda/10$  for all frequencies of interest, and the measurement will not be significantly affected by probe placement.

The test equipment required is as follows:

- 1) Storage/holdup capacitors: 10,000  $\mu$ F or ten times the hold-up cap in the test sample, whichever is larger
- 2) Oscilloscope (floated ground if small value series resistor is used)
- 3) Solid-state switch that closely simulates the characteristics of the switching device used on the platform (example design shown in Figure 2.5-31)

- 4) One of the following current measurement devices:
  - a. Current probe with flat transfer impedance from 20 Hz to 1 MHz and minimum sensitivity of -40 dBΩ
  - b. 10 mΩ resistor sized to handle steady-state test sample current



Note 1: CAUTION: Vcc potentials referenced to switched power potential, absolute Vcc potential is either 9 Volts or 18 Volts added to switched bus potential  
 Note 2: All resistor values in Ohms, all nominal

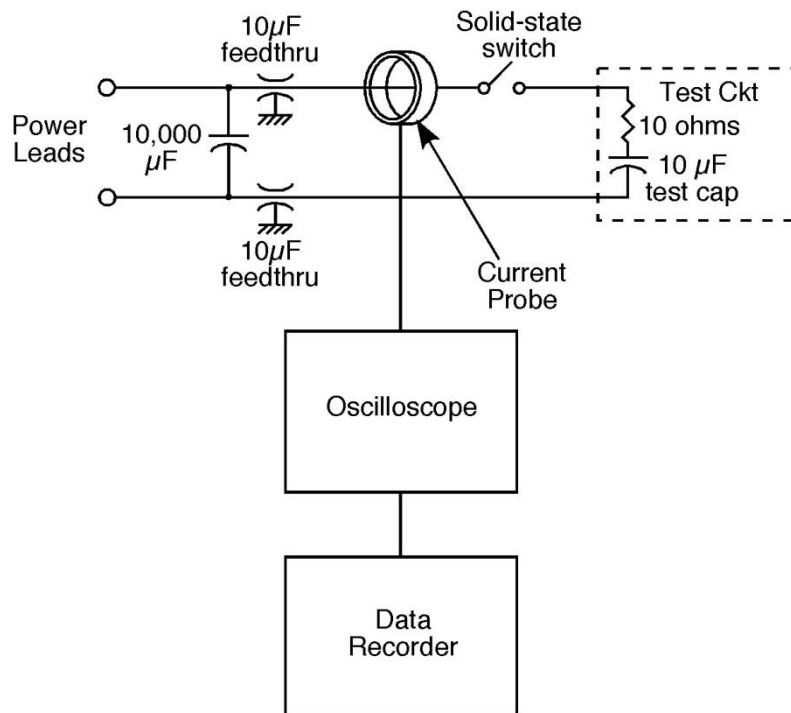
**Figure 2.5-31. Solid State Switch Example Design**

Measurement System Check procedure:

- 1) Configure the test setup for the measurement system check as shown in Figure 2.5-32 if using the current probe method or Figure 2.5-33 if using the series resistor method. Ensure that the power switch is turned off (open). If using a current probe, it must be oriented for a positive reading for the inrush current event. If using a resistor, the oscilloscope leads must be connected so that the potential drop across the resistor during the inrush event gives a positive reading.
- 2) Set up the oscilloscope as follows:
  - a. DC coupling
  - b. 5 mV/division sensitivity for resistor method; for current probe method, set sensitivity as appropriate for probe's transfer impedance
  - c. 50 μs per division time base
  - d. Single sweep after trigger, with hold
  - e. Display delay such that leading edge of waveform is one division right of the left side of display grid

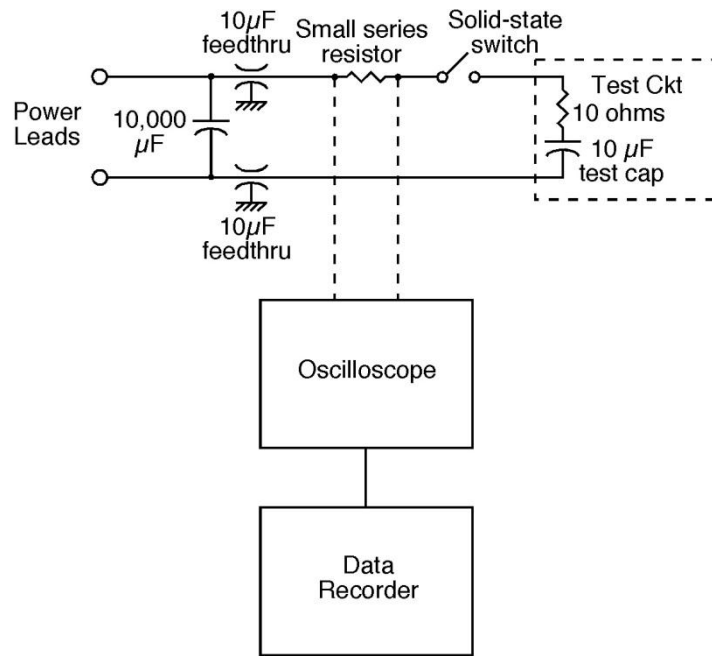
Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

- f. Trigger:
- level set to 10 mV for resistor method; for current probe method, set level as appropriate for probe's transfer impedance
  - positive (rising) edge
- 3) Turn on the measurement equipment and allow a sufficient time for stabilization.
  - 4) Energize switch and record waveform. Compare trace to the current profile shown in Figure 2.5-34. If the current probe method is used, the displayed voltage must be converted to current using the current probe transfer impedance. If the series resistor test method is used, the measured voltage must be converted to current using the value of the series resistor (nominally 10 mΩ).
  - 5) If waveform departs from the Figure 2.5-13 waveform by more than 20%, check set up for problems and rectify before proceeding.

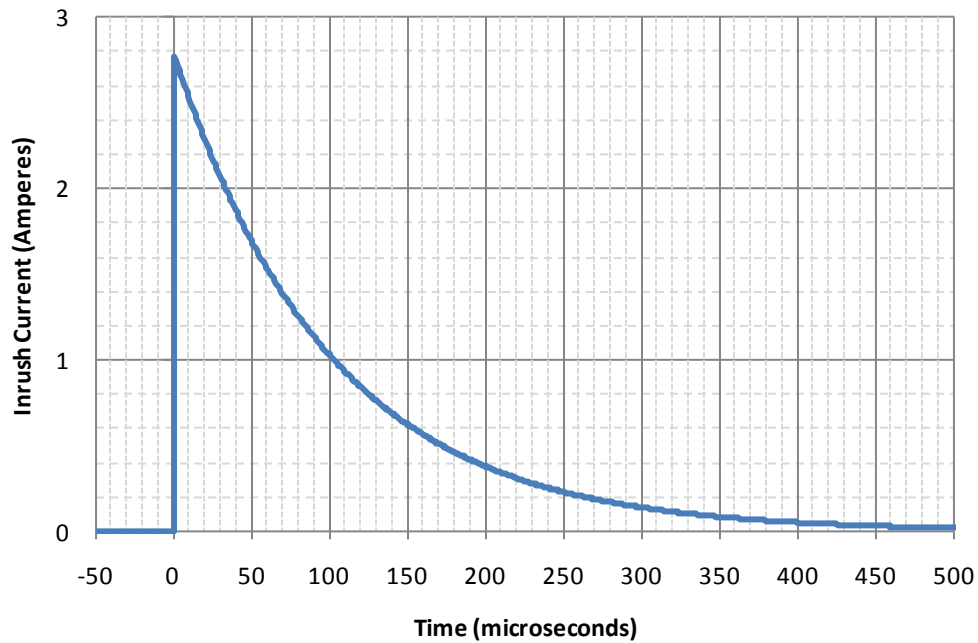


**Figure 2.5-32. Measurement System Check (Current Probe)**





**Figure 2.5-33. Measurement System Check (Series Resistor)**



**Figure 2.5-34. Inrush Current Calibration Waveform (20% Tolerance)**

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

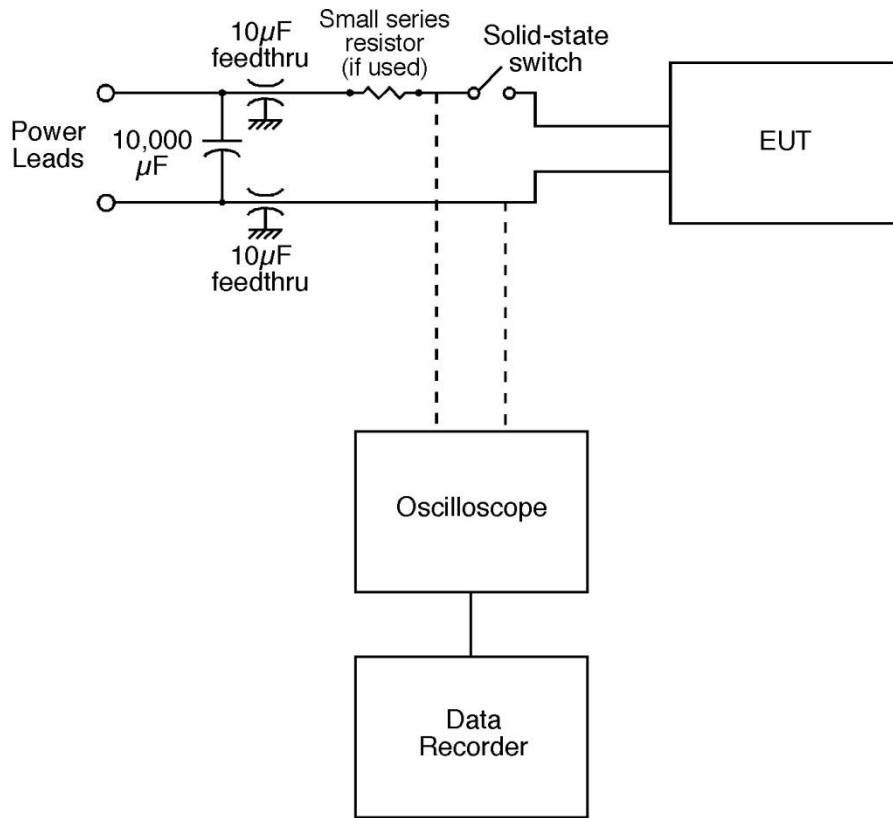


Figure 2.5-35. Input Power Voltage Sag Measurement Setup

Input Power Voltage Sag Measurement Procedure:

- 1) Configure the test setup for the power bus sag measurement as shown in Figure 2.5-35. Ensure that the power switch is in the open position.
- 2) Set up the oscilloscope as follows:
  - a. 5 V/division sensitivity with 0 Volts at bottom of display grid for nominal 28 Vdc bus.
  - b. 50  $\mu$ s per division time base
  - c. Single sweep after trigger, with hold
  - d. Display such that leading edge of waveform is one division left of the left side of display grid
  - e. Trigger:
    - level set to 1 – 2 Volts below nominal bus potential
    - looking for negative going waveform (negative trigger slope)
- 3) Energize switch and look for waveform that sags more than 2.8 Volts (or 10% of nominal). If sag is greater than 10%, then provide additional line-to-line capacitance and/or decrease the impedance of the current monitoring device (especially if using the series resistor) as necessary. Measure again and repeat process until bus sag is within tolerance. When making multiple inrush measurements, ensure that test sample power supply filters have time to adequately discharge between measurements.

#### 2.5.3.3.4 Conducted Susceptibility, Power Leads, 30 Hz to 150 kHz

The MIL-STD-461F CS101 limit is defined in order to provide a margin with respect to the power generation characteristics defined in MIL-STD-704 for aircraft. The ripple levels in MIL-STD-704 are based on electromechanical power sources, i.e. rotating machinery that turns a shaft that provides motive power to an electrical generator. Such power sources have significant inherent ripple generated by the source itself.

GSFC platforms do not typically use electromechanical power sources. GSFC spacecraft operating in Earth orbit, or within Earth's orbit of the Sun, tend to use solar panel arrays that charge a battery and provide direct current to loads. The power system typically controls bus potential using a Pulse Width Modulation (PWM) system operating at a frequency determined by the platform, which can range from a few hundred Hz to a few hundred kHz. Such power sources will have source-generated ripple at the PWM frequency, but this ripple is typically less than that generated by the electromechanical sources specified in MIL-STD-704. Any ripple at frequencies other than the PWM frequency and its harmonics is dominated by load-induced effects, which are controlled by the conducted emissions requirements specified in Section 2.5.2.1.1.

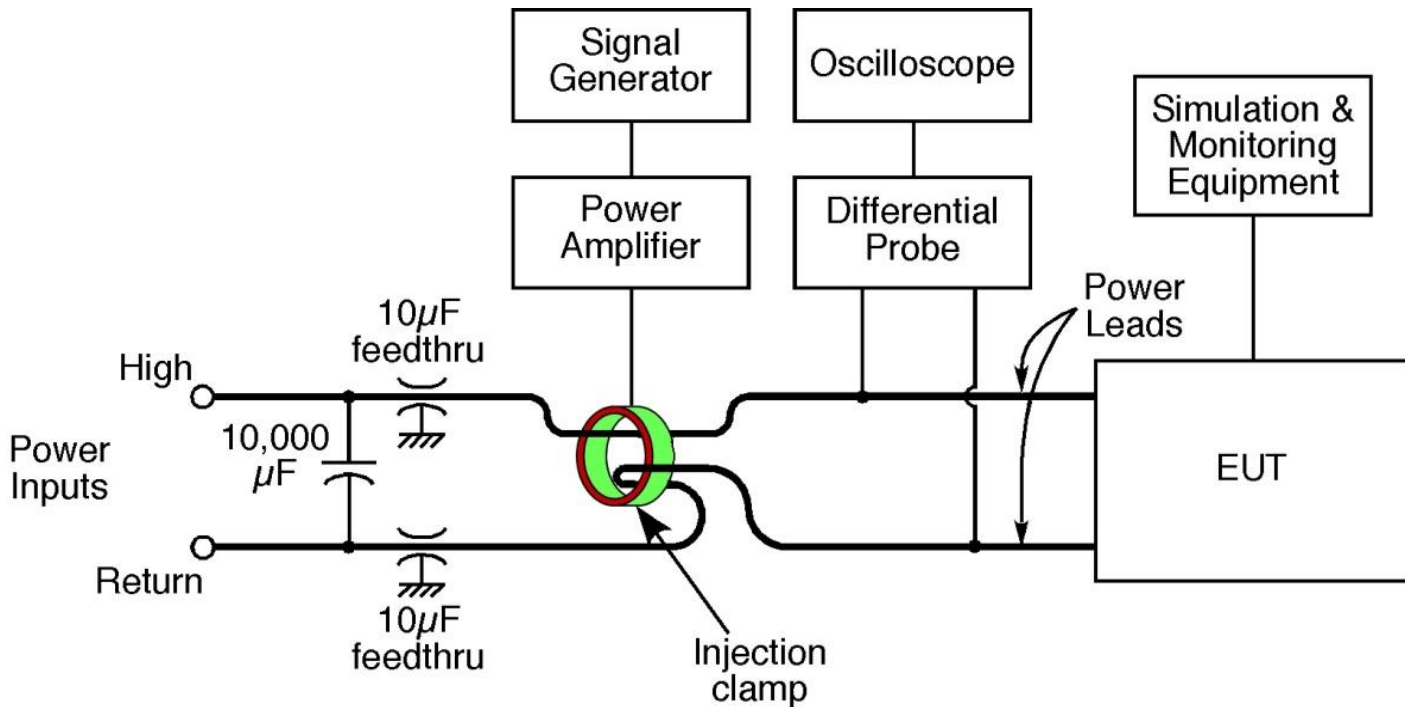
The MIL-STD-461F CS101 test method, along with its CS01 predecessor from MIL-STD-462, has been shown to pose a potential damage risk to the EUT if proper precautions are not followed. Both of these test methods insert the secondary windings of a coupling transformer with an inductance of approximately 1 millihenry in series with the power lead to the EUT. Such a large inductance in series with the power source can cause instability and even damage to a switched mode power supply lacking adequate decoupling from the power source. For these reasons, the power amplifier driving the coupling transformer (shown in MIL-STD-461F Figure CS101-4) **MUST** be powered up and allowed to stabilize prior to applying power to the EUT. Failure to do so is the primary cause of the instability and damage problems described above.

In order to mitigate these concerns, an alternative injection technique similar to modern bulk current injection technology was developed for the Apollo program Lunar Excursion Module (LEM) in the mid-1960s. This technique uses an injection clamp instead of a coupling transformer as shown in Figure 2.5-36. The recommended injection clamp is the Solar Electronics 6541-1 or equivalent. This device inserts approximately 0.01 ohm in series with each wire passing through it, which significantly reduces the risks associated with the 1 millihenry series inductance in the CS101 test method.

The Solar Model 6541-1 has a multiple pin connector instead of the more typical coaxial connector. The pins that connect to the power amplifier output are "C" and "D". The other pins are not connected for this test.

The balance of the test is identical to the MIL-STD-461F CS101 test method with the 6541-1 injection clamp taking the place of the coupling transformer. A 100 Watt audio amplifier is required to drive the injection clamp. Note from Figure 2.5-36 that the signal is injected as a Differential Mode signal.

If the actual ripple characteristics of the spacecraft power subsystem exceed the levels below 500 Hz shown in Figure 2.5-15, then the standard CS101 test method must be used.



**Figure 2.5-36. Conducted Susceptibility Alternate Test Setup (Differential Mode, 30 Hz to 150 kHz)**

2.5.3.3.5 Conducted Susceptibility, Transients, Power Leads

The MIL-STD-461F CS106 requirement was developed specifically for Navy applications, submarine and surface ship equipment in particular. The 5 microsecond pulse represents the typical transient observed on these platforms (shown in MIL-STD-461F Figure CS106-1).

The 10 microsecond pulse specified by the CS06 requirement of MIL-STD-461C is preferred because:

- it is more stringent limit than the CS106 pulse, and
- the CS06 transient generator is a standard piece of EMI test equipment in any EMI test lab

For all platforms, the negative transient shall be applied. For platforms that include rapidly changing inductive loads that are expected to generate significant back emf onto the bus, the positive transient shall be applied as well.

Although this limit is expected to be sufficient for most platforms, it must be compared against the worst-case expected transients generated by equipment on the platform. If significantly longer transients are expected, the limit must be tailored accordingly, and a different transient generator must be used.

2.5.3.3.6 Conducted Susceptibility, Bulk Cable Injection, 10 kHz to 200 MHz

2.5.3.3.6.1 Conducted Susceptibility, Power and Signal Cables, Common Mode

The MIL-STD-461F CS114 limits are specifically defined in order to simulate currents that will be developed on platform cabling from electromagnetic fields generated by antenna transmissions both on and off the platform. In MIL-STD-461F Section A.5.13, the ratio of common mode current to electric field intensity is given as 1.5 mA per V/m. Using this relationship, the CS114 limit curves in MIL-STD-461F correspond to radiated susceptibility (RS103) limits ranging from 5 V/m up to 200 V/m. While these types of levels may be expected at the launch site, they are significantly higher than typical on-orbit levels, and they are almost certainly unnecessarily stringent for controlling cable-to-cable crosstalk. A more typical on-orbit level is 2 V/m as discussed in Section 2.5.2.4.2, corresponding to a common mode current of 3 mA or 70 dB $\mu$ A. This forms the basis for the limit shown in Figure 2.5-15. In addition to protecting against crosstalk, testing to these levels will demonstrate the ability of the victim cables to withstand typical on-orbit electric fields.

Shielding options may be limited on some potential victim cables, such as those connecting to equipment operating at cryogenic temperatures. Such cables may be susceptible to levels lower than those shown in Figure 2.5-15. These cables must be identified and characterized as early as possible for susceptibility to common mode currents. The thresholds of susceptibility shall be documented in the project EMCCP, and the Common Mode Conducted Emissions limit for neighboring culprit cables must be defined accordingly in order to protect the victim cables from crosstalk.

Shielding options may also be limited on potential culprit cables. If so, the crosstalk concern must be addressed by properly separating victim and culprit cables.

2.5.3.3.6.2 Conducted Susceptibility, Power Leads, 150 kHz to 50 MHz

The basic engineering concern of this test is the need to evaluate the conducted susceptibility of equipment to ripple on its primary power inputs. For this purpose, MIL-STD-461C and MIL-STD-462 defined CS01 for low frequencies (< 50 kHz) and CS02 for high frequencies (> 50 kHz). Both of these tests specified voltage injection.

With the release of MIL-STD-461D in 1993 and continuing through -461E (1999) and -461F (2007), CS01 was replaced with CS101. CS101 is nearly identical to CS01 except that it goes up to 150 kHz and that it requires a higher powered amplifier than CS01.

MIL-STD-461F specifies the CS114 test method for evaluating conducted susceptibility on power lines at frequencies above 150 kHz. CS114 is primarily intended to evaluate susceptibility of bulk cables (power and signal) to common mode currents, but it is also intended for injection of current onto power leads as stated in MIL-STD-461F Section 5.13.3.4.c:

“Also perform the procedures on power cables with the power returns and chassis grounds (green wires) excluded from the cable bundle. For connectors which include both interconnecting leads and power, perform the procedures on the entire bundle, on the power leads (including returns and grounds) grouped separately, and on the power leads grouped with the returns and grounds removed.”

It is necessary to convert the CS02 voltage limit into a current limit for CS114. The CS02 limit defined as 1 Vrms from a 50 ohm source; CS114 also specifies a 50 ohm source. The CS02 voltage limit converts to a short circuit current of:

$$I_{sc} = \frac{1 V}{50 \Omega} = 20 mA$$

An injected current of 30 mA (89.5 dBμA) is specified in order to provide margin with respect to this value. Because the CS114 test method specifies that the injected current is 6 dB above the calibration current, the calibration level is defined as 15 mA (83.5 dBμA).

The 15 mA calibration level converts to 0.75 V across the 50 ohm calibration load as specified by CS114. The 50 ohm calibration load is part of a 100 ohm loop and is driven by a 50 ohm source impedance. This converts to an open circuit voltage of approximately 3 times the calibration voltage, or 2.25 V (this level will likely vary somewhat depending on the insertion loss of the injection clamp). This voltage will be applied to the EUT only when its input impedance is much higher than 50 ohms. Under these circumstances, the CS114 injection will be limited to the pre-calibrated power level, and the injected current will be much lower than the pre-calibrated current level.

The upper frequency limit is set to 50 MHz in order to be consistent with the upper frequency limit for the conducted emissions tests. This allows the injection probe to be placed up to 60 cm (λ/10 at 50 MHz) from the EUT instead of the default 10 cm specified by MIL-STD-461F.

#### 2.5.3.3.7 Radiated Emissions, Electric Field

The basic limit and test method of MIL-STD-461F RE102 apply while incorporating the following tailoring guidelines in section A.5.17 of the MIL-STD-461F Application Guide:

- “The limits could be adjusted based on the types of antenna-connected equipment on the platform and the degree of shielding present between the equipment, associated cabling, and the antennas. For example, substantial relaxations of the limit may be possible for equipment and associated cabling located totally within a shielded volume with known shielding characteristics.”
- “It may be desirable to tailor the frequency coverage of the limit to include only frequency bands where antenna-connected receivers are present.”

The RE102 limit specifies the full range of frequencies used by antenna-connected receivers on military platforms; not all of these frequencies are used on NASA platforms. Most GSFC spacecraft do not use the electromagnetic spectrum below 2 GHz for the purpose of receiving RF signals. 200 MHz has been selected as the low end of radiated emission control for GSFC platforms for the following reasons:

- Control of conducted emissions at ultra-high frequencies (UHF) is problematical. While shield room radiated measurements are notoriously inaccurate, UHF conducted emissions measurements are not going to be much better unless the absorbing clamp is used. Because it cannot be guaranteed that the absorbing clamp will be used in all cases, a radiated technique is preferred. In addition, above 400 MHz, equipment case leakage may be as significant as radiation from the interconnecting cables.
- Many satellites use the Global Positioning System (GPS). While this doesn't drive RE control down to 200 MHz, it does require control down to 1 GHz.
- All launch platforms require strict control of the command destruct signal band around 400 MHz. While many portions of a GSFC satellite may be powered off during the ascent portion of a mission, any that need to be powered will have to meet the stringent command destruct band RE limit. Therefore, RE control at 400 MHz is required.
- 200 MHz is a convenient breakpoint in the RE102 test method between different antenna types.

For all of the reasons given above, the RE102 test method shall apply between 200 MHz to 18 GHz with the limit shown in Figure 2.5-19. The figure includes a representative notch for the S-band receiver (1.7-2.3 GHz), which should apply to most GSFC platforms. For equipment that is powered on at launch, notches may also be applied to protect receivers on the launch vehicle and at the launch site. Figure 2.5-19 also includes a representative notch for the Launch Vehicle Command Destruct receiver (420-480 MHz). These notches, and any other applicable notches for other receivers, must be tailored for the specific receiver frequency ranges and sensitivity levels on the platform. In each notch, the measurement bandwidth that simulates the protected receiver must be used instead of the MIL-STD-461F prescribed bandwidth that would normally be used in that band.



Further guidance relevant to Launch Vehicle & Launch Site Electromagnetic Environments can be found in the following documents:

- ELVL-2010-0042300 Electromagnetic Environments – A guideline for Spacecraft launching from Eastern, Western and Pacific Ranges

Emissions below 200 MHz are addressed by the Common Mode Conducted Emissions (CMCE) test on power and signal cables defined in Section 2.5.2.1.2. The purpose of that test is to control cable-to-cable crosstalk while at the same time eliminating the paperwork associated with waivers against an inapplicable and unnecessarily stringent RE102 limit below 200 MHz. The CMCE test is adapted from the alternate RE102 test methods defined in SL-E-0002, Space Shuttle Specification, Electromagnetic Interference Characteristics, Requirements for Equipment.

The guidance above should apply to most GSFC platforms. However, if any equipment on the platform uses the electromagnetic spectrum below 200 MHz, then a radiated emission limit must be imposed at those frequencies. In addition, the antenna used for the measurement should simulate that used on the platform.

Test Procedure Guidelines:

The MIL-STD-461F RE102 test procedure specifies the use of double ridge horn antennas between 200 MHz and 18 GHz. This antenna type is acceptable if it is the only type available; however, a log-spiral or log periodic antenna is preferred. The double ridge horn has a constant aperture; this means the gain increases with frequency, which translates to a decreased field-of-view as frequency increases. The log-spiral and log periodic antennas have a lower and constant gain as a function of frequency, which translates into a wider and constant field of view. For these reasons, the log-spiral and log periodic antennas are better suited to capture all of the radiated emissions from a given test setup using fewer antenna positions, thus reducing total test time.

A few notes regarding log-spiral antennas:

- Log-spiral antennas are circularly polarized, which means their response to a linearly polarized field is 3 dB down from what it is for the proper circular polarization. Therefore, a 3 dB correction factor must be added to the manufacturer's antenna factor.
- The wide field-of-view (low gain) and circular polarization enable most measurements to be completed using only one position and one polarization.
- The log-spiral antenna only works up to 10 GHz; the double ridge horn must be used for measurements above 10 GHz.
- The low gain of the log-spiral antenna may not have the required sensitivity in the notches, depending on the level of the notch. If so, it is recommended that a basic sweep be performed using the log-spiral, then go back and scan the notch frequencies with the double ridge horn.

## SECTION 2.6

### THERMAL

## 2.6 VACUUM, THERMAL, AND HUMIDITY VERIFICATION REQUIREMENTS

The vacuum, thermal, and humidity requirements herein apply to ELV payloads. 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.
- b. The thermal design and the thermal control system shall maintain the affected hardware within the established mission thermal limits during planned mission phases, including survival/safe-hold, if applicable.
- c. The hardware shall withstand, as necessary, the temperature and/or humidity conditions of transportation, storage, launch, flight, and 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 the hardware can be shown by analyses to be insensitive to vacuum effects relative to temperature levels and temperature gradients.

### 2.6.1 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. Figure 2.6-1 shows the organization of the requirements and supporting information within this section of the GEVS.

While payloads mounted in pressurized compartments need not be qualified for the vacuum environment, the thermal cycling requirements of paragraph 2.6.2.4 do apply. These payloads must also be qualified for proper thermal performance. An ambient pressure test must be designed to prevent external natural convection, since there will be no natural convection in micro-gravity.

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 for passively controlled items are the same except that a 10°C temperature margin is added for qualification/protoflight testing and a 5°C margin is added for acceptance testing. For items controlled by active temperature control thermal systems, the margins are the same for qualification/protoflight and acceptance testing, as specified in 2.6.2.4.

Electronic card/piece part thermal analyses shall be performed to ensure that the GSFC Preferred Parts List (PPL) derated temperature limits and the allowable junction temperatures are not exceeded during qualification test conditions.

### 2.6.2 Thermal-Vacuum Qualification

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

Table 2.6-1  
 Vacuum, Thermal, and Humidity Requirements

Requirement	Payload or Highest Practicable Level of Assembly	Subsystem including Instruments	Unit/ Component
Thermal-Vacuum <sup>1,6</sup>	T	T	T <sup>2</sup>
Thermal Balance <sup>1,3,6</sup>	T and A	T,A	T,A
Temperature-Humidity <sup>3</sup> (Habitable Volumes)	T/A	T/A	T/A
Temperature-Humidity <sup>4</sup> (Transportation & Storage)	A	T/A	T/A
Leakage <sup>5</sup>	T	T	T

1. Applies to hardware carried in unpressurized spaces and to ELV-launched hardware.

2. Temperature cycling at ambient pressure may be substituted for thermal-vacuum temperature cycling if it can be shown by a comprehensive analysis to be acceptable. This analysis must show that temperature levels and gradients are as severe in air as in a vacuum.

3. Applies to flight hardware located in pressurized area.

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

5. 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.

6. Survival/Safehold testing is performed on that equipment which may experience (nonoperating) temperature extremes more severe than when operating. The equipment tested is not expected to operate properly within specifications until the temperatures have returned to qualification temperatures.

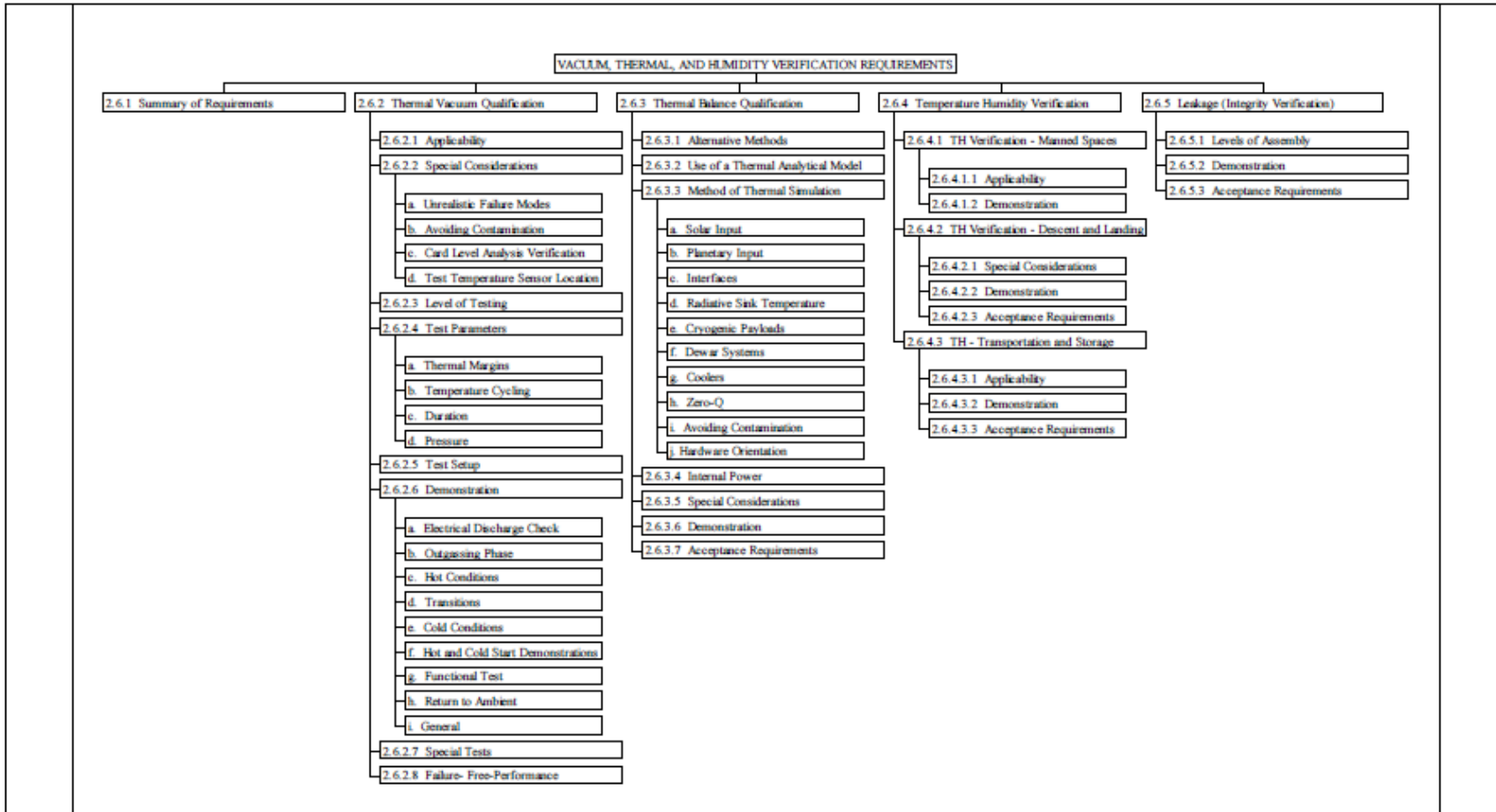
T = Test required.

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 this level of assembly if analysis verification is established for nontested elements.

Note: Card level thermal analysis using qualification level boundary conditions is required to insure derated temperature limits, for example, junction temperature limits, are not exceeded



**Figure 2.6-1 Section 2.6 Thermal Requirements**

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

### 2.6.2.1 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.

The Goddard Space Flight Center generally utilizes a protoflight qualification test program. Protoflight thermal test levels are the same as prototype. Figure 2.6-2 shows operational temperature test margins. Contingency margins required by design rules are included in the development of the expected flight temperatures. Unit survival limits should be defined by the hardware limits.

Spare components shall undergo a test program in which the number of thermal cycles is equivalent to the total number of cycles to which other flight components are subjected 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 the payload/spacecraft.

Redundant components shall be exercised sufficiently during the test program, including cold and hot starts, to verify proper orbital operations. Testing to validate all applicable operational modes shall be performed. 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).

For spare and redundant components, the duration and test temperature levels of the tests shall be the same as those for flight components.

For repaired equipment, usually a component, subsequent testing shall be sufficient to demonstrate flight worthiness. If additional testing is expected at either the Subsystem or the Payload level, the number of cycles can be reduced so long as the total number of cycles satisfies the 12 cycle requirement.

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 change 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.

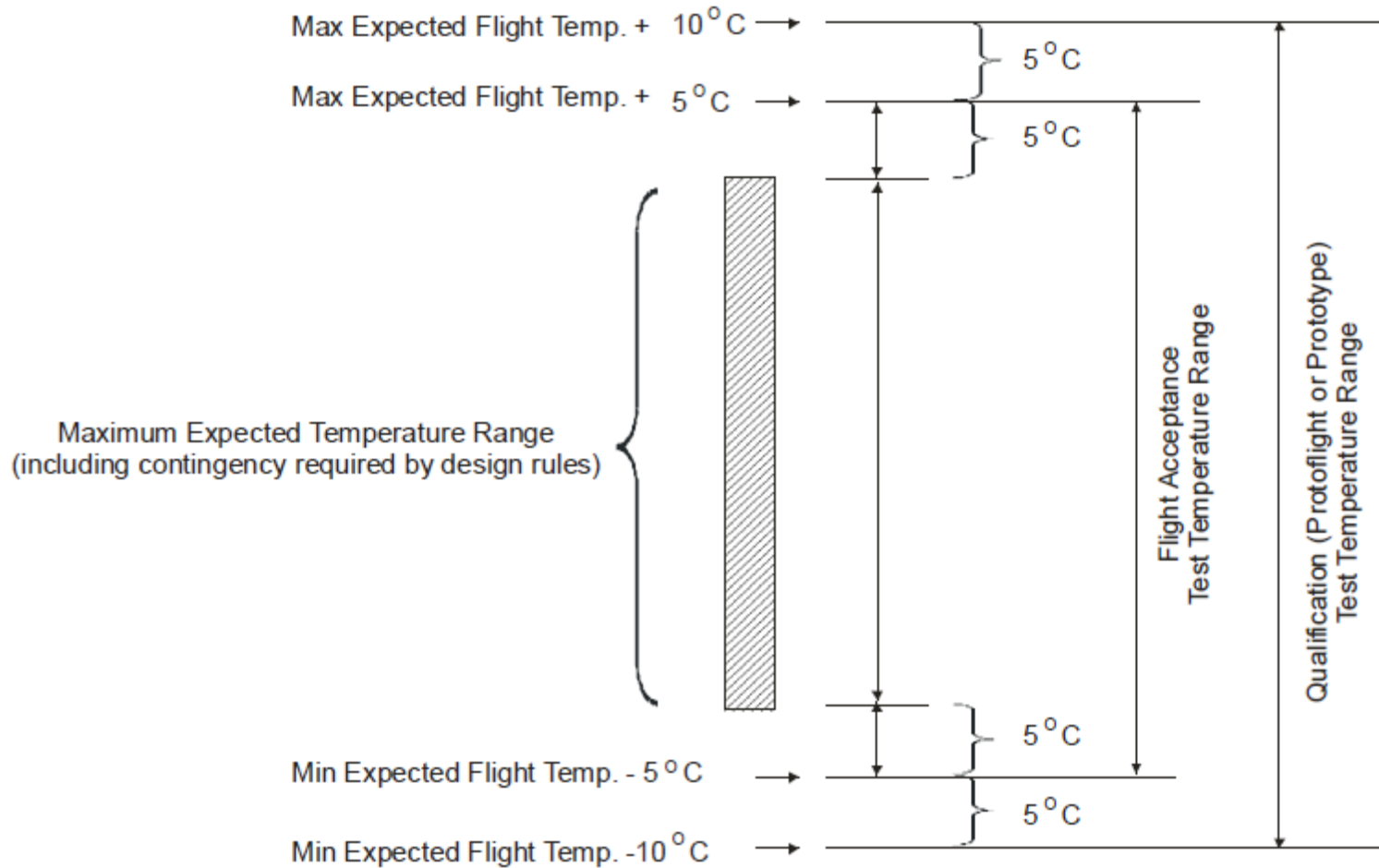


Figure 2.6-2 Qualification (Protoflight or Prototype) and Flight Acceptance Thermal-Vacuum Temperatures

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.



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), Gas Chromatographs/Mass Spectrometers (GC/MS), Fourier Transform Infrared Spectrometers (FTIS), Cryogenic QCM's, 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. It is recommended that testing 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 the critical period. In all cases, every effort should be made to keep the test article warmer than its surroundings during testing.

- c. Card Level Analyses Verification - During hot qualification testing, consideration should be given to monitoring temperature sensors placed at strategic points on electronic cards or piece parts to confirm that the detailed thermal analyses performed were conservative. These temperature monitors can either be flight sensors or test sensors.
- d. Test Temperature Sensor Location - 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. In some cases, the temperature sensors shall be attached to the component base plate or to the heat sink on which the component is mounted, if the temperature requirement is defined at the mounting interface. Temperature soaks and dwells shall begin when the "control" temperature is within  $\pm 2^{\circ}\text{C}$  of the proposed test temperature. The "control" temperature criterion for cryogenic systems should be determined by the thermal engineer and the Project as it may be significantly more stringent than  $2^{\circ}\text{C}$ .

#### 2.6.2.3 Level of Testing

There is a minimum of three levels of testing; the component, subsystem/instrument, and the payload/ spacecraft levels. 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

Workmanship margin, temperature cycling, soak duration, test chamber conditions, transition rates, temperature and pressure regimes, are some of the parameters that define key environmental conditions of the test:

- a. Workmanship Margins - Thermal margins shall be established to induce stress conditions to detect unsatisfactory performance that would not otherwise be

uncovered before flight. The workmanship test margin is defined as an increase in a condition beyond the range of conditions the hardware would experience over the expected lifetime. This could include temperature, heat loads, and/or environmental conditions.

The maximum and minimum temperatures to be imposed during unit level thermal vacuum testing shall represent, as indicated above, a temperature range large enough, including margins, to induce stress during temperature cycling. The basis for these test temperatures shall be established based on unit temperature requirements. For instrument/subsystem and spacecraft level testing, the minimum and maximum temperatures imposed shall be derived from the collection of various unit temperature ranges, while acknowledging the least robust range. For multiple instrument/spacecraft builds only, where a correlated model is developed on the first build, subsequent builds may base the thermal vacuum test temperatures on predicted temperatures derived analytically using the test verified model, only if these builds are identical in configuration. When a thermal balance test precedes the thermal vacuum test, results from that test may be used to refine the thermal vacuum test criteria, presuming that there is sufficient time to correlate the model and generate updated predictions prior to the thermal vacuum test. If predictions from a verified model are not available at the time of the thermal vacuum test, the basis shall be Project Office established on-orbit maximum and minimum allowable operating limits. This basis shall constitute the "flight" temperature range to which test margins shall be applied.

For passively controlled systems, a qualification temperature margin of no less than 10°C above the "flight" maximum operating temperature (as established above) and 10°C below the "flight" minimum operating temperature shall be used in establishing test temperatures. The margins for acceptance testing of previously qualified hardware may be reduced to 5°C, as long as testing to these levels does not preclude protoflight test levels from being achieved at higher levels of assembly.

The test margins for actively controlled hardware, as specified in the following three paragraphs, shall apply to both qualification/protoflight and to acceptance testing of those systems and components.

For actively controlled systems such as Heaters, ThermoElectric Coolers (TECs), Loop Heat Pipes (LHPs), Capillary Pumped Loops (CPLs), or other devices with selectable/variable set points, a test temperature margin of no less than 5°C shall be imposed on the respective set point band that is under control. The required range of temperature control shall be demonstrated in the worst hot and worst cold thermal environments during thermal balance testing.

For components/subsystems/payloads with operational heater circuits with fixed temperature setpoints, the cold end margin may be reduced from 10°C to 5°C.

If a component/subsystem/payload has an active control whose range is not selectable/ variable such that the control system will not allow the hardware to be stressed via temperature, then the stressing shall be induced by the increase or decrease of a heat load (internal or external) of at least 30 %. The active temperature control hardware shall maintain control under these stressed conditions. The goal of this testing is to create an environmental condition in excess of what the system will see on-orbit in order to stress the system and demonstrate its overall flightworthiness.

The 10°C thermal vacuum margin requirement may not apply to cryogenic systems. Obtaining “cold” margins may not be possible for some cryogenic systems, for example, an instrument inside a dewar. Also, operating the test article at temperatures 10°C above normal may be detrimental to performance testing. The following parameters define key environmental conditions of the test:

Cryogenic margins should be established by the thermal engineer and the Project based on the unique characteristics of the test article.

The survival/safehold thermal-vacuum test shall consist of driving the element, without any test margin, to the desired temperature, and then returning that element to the qualification temperature, if different, to functionally check the operation. No component shall be allowed to exceed the non-operating temperature limit with allowable tolerances.

Temperatures shall not exceed allowable qualification temperatures for extended periods of time. This may constrain the test to be driven by those components with the smallest allowable temperature range. Also, for testing at higher levels of assembly, the “red limits” (not-to-exceed temperatures) shall be established based on temperatures actually achieved during testing at lower levels of assembly.

- b. Temperature Cycling - Cycling between temperature extremes has the purpose of checking performance during both stabilized conditions and transitions thereby 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. During the cycling, the hardware shall be operating and its performance shall be monitored. Items not in the spacecraft thermal vacuum test (see Test Like You Fly exceptions list), must modify their lower level test plans to achieve 12 TV cycles prior to flight. It is strongly recommended that the SCTV configuration be as complete as possible to validate the system prior to flight.
  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.

If thermal cycle testing is being performed as a performance or workmanship screen for mechanical hardware and the hardware does not contain any heat generating components, it is acceptable to use the qualification margin of +/- 10 corresponding to vacuum testing. The number of cycles for this type of testing

may be less than the minimum of the 8 thermal-vacuum cycles required for qualification. Typically 2-3 thermal cycles are sufficient to demonstrate that the hardware can survive the predicted thermal environment without damage or degradation in performance. Examples of where these reduced requirements may be applied are when thermal-cycling a honeycomb panel with bonded inserts to verify integrity under thermal loading or during testing of a mechanism to verify operation at temperature extremes.

The required approach is to test in the expected environment (vacuum). Waivers to vacuum testing of flight hardware must be submitted and approved by AETD prior to PDR.

4. Cryogenic systems - The cycling requirement may not apply to cryogenic systems. For example, instruments inside a dewar may never see cycling in flight. Cycling them during ground testing may also be preclusive due to time constraints and may cause undue stress on flight systems. Operational conditions must be considered when determining cryogenic system cycling. The number of cycles shall be specified by the Project with inputs from the Experimenter and the Thermal Engineer.
- c. Duration - The total test duration shall be sufficient to demonstrate performance and uncover early failures. The duration varies with the time spent in flight 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 the required performance tests (functional, comprehensive, etc.) for all modes of operations including safehold/survival at the hot and cold extremes. Projects seeking to reduce durations or the number of cycles must submit deviations and receive approval from AETD prior to PDR. The test plan for unit/subsystem/instrument should then be adjusted to ensure 12 cycles before flight for all units.
  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 the required performance tests for all modes of operation including safehold/survival.
  3. Unit/Component - Components shall be exposed for a minimum of four (4) hours at each extreme of each temperature cycle. The thermal soaks must be of sufficient duration to allow time for the required performance tests for all modes of operation. Hot and cold start demonstrations shall be performed for each unit/component per section 2.6.2.6 f. If component testing is done at ambient pressure, the dwell time should be increased to six (6) hours.

The dwell time for cryogenic elements may be significantly longer than noted above. Times should be established by cognizant engineers based on the operational characteristics.

The dwell time for thermal cycling at ambient pressure of mechanical hardware which contains no heat generating devices and is being performed to demonstrate strength or performance at temperature does not require an

increase in dwell time from the four (4) hours specified for thermal-vacuum testing. It may be acceptable to reduce the dwell time to less than four (4) hours depending on the goals of the test.

The survival/safehold TV test shall consist of soaking the non-operating element for at least four (4) hours at proper temperature conditions.

- d. Pressure - The chamber pressure after the electrical discharge checks are conducted shall be less than  $1.33 \times 10^{-3}$  Pa. ( $1 \times 10^{-5}$  torr). The ability to function through the voltage breakdown region shall be demonstrated if applicable to mission requirements (those elements that are operational during launch).

#### 2.6.2.5 Test Setup

The setup for the test, including any instrument and/or component 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. Test heaters on the payload may be required to achieve proper and safe temperatures.

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 time 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 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 as specified in 2.6.2.4.c.
- d. 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 and/or unrealistic gradients. Violation of functional specifications is acceptable during transitions with the approval of the Project Office.

The rate of transition shall be specified to insure that stresses caused by thermal gradients will not damage the test article. Contamination effects may also be a factor. Care must be exercised with cryogenic systems where the thermal stresses can be severe. The cool-down and warm-up for cryogenic systems should be as flight like as possible.

STOP (Structural/Thermal/Optical) analyses with temperature variant properties should be performed to insure stresses and alignments are acceptable for the given transition rate.

- e. Cold Conditions - The temperature controls shall be adjusted to cause the test item to stabilize at the lowest test temperature. Cold turn-on capability shall be demonstrated at the start of the cold condition. The duration of the cold phase shall be sufficient to permit the performance of the functional tests with a minimum soak time as specified in section 2.6.2.4.c.
- f. 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. Turn-on capability shall be demonstrated under vacuum at least once, at both the low and high temperatures, on primary and redundant side, as applicable. Test turn-on temperatures are defined by the expected mission operations without any margin; that is, temperatures should be at either survival/safe-hold or qualification temperature conditions, whichever are more extreme, as appropriate. At the Unit/Component level, this demonstration shall consist of power-off, power-on cycles for each unit/component. At the Subsystem/Instrument level, and Payload/Spacecraft level, this demonstration shall be consistent with the scenario regarding which units/components are actually power cycled (off/on) in orbit, and also for recovery from a survival/safe-hold mode in orbit. For example, recovery from cold survival/safehold temperatures to cold operational temperatures may be accomplished either by using a flight heater, or alternately, by turning the units/components of the test item back on and allowing internal dissipation to warm temperatures. Proper operation is then checked after the component has returned to the qualification limit. The duration of the soak with the test item off, or in survival/safe-hold mode, shall be in accordance with section 2.6.2.4.c.
- g. Functional Test - Functional tests shall be performed at each hot and cold soak plateau and during transitions. A comprehensive performance test (CPT) shall be performed at least once during hot plateau(s) and once during cold plateau(s), exercising complete primary and redundant operations, unless it is determined to be impractical. In that case, 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 2.3.2 apply. Functionality of the thermal control system hardware shall be demonstrated during the Thermal-Vacuum Qualification test.
- h. 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. Before the chamber can be backfilled with air, all sensors should read above the dew point to insure that water does not condense on the payload.
- i. General - The margins, soak criteria, cycling, and duration guidelines listed above apply to primarily test articles around room temperature (except where noted). Test

parameters for high temperature and cryogenic systems should be based on flight operations. Parameters should be determined early in the program by the engineering and science teams.

#### 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 Failure-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. It is noted that a total of 350 hours of failure-free hours is a requirement of which 200 are to be in vacuum. (Refer to section 2.3.4).

#### 2.6.3 Thermal Balance Qualification

The adequacy of the thermal design and the capability of the thermal control system shall be verified under simulated on-orbit worst case hot and worst case cold environments, and at least one other condition to be selected by the thermal engineer. Consideration should be given for testing an "off-nominal" case such as a safehold or a survival mode. Ideally the test environments will bound the worst hot and cold flight environments such that the test results directly validate the adequacy of the thermal design. An additional objective of the test is to verify and correlate the thermal model so it can be used to predict the behavior of the payload under future non-tested conditions and/or flight conditions. 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.

Thermal design margins shall be verified under worst case hot and cold, and if tested, safehold/survival, conditions. Select examples of the margins to be established are:

- Operational heater duty cycle less than 70% in worst cold case, including minimum voltage as established by the project;
- Survival heater margin, dependent on survival setpoint/temperature limit and available resources;
- Interface heat flows are within requirements;
- Selectability of multiple setpoints for two-phase flow systems, such as LHP and CPL, in worst case environments;

- Heat transport margins of 30% for two-phase flow systems, such as LHP, CPL, Constant Conductance Heat Pipes (CCHP), Variable Conductance Heat Pipes (VCHP), Diode Heat Pipes (DHP), in worst case environments, and
- Radiator heat rejection margin in worst case environments, dependent on available resources.

Note: For two-phase flow systems, it may be necessary to conduct thermal verification tests at all levels of assembly since it is often not possible to verify performance by analysis (see Table 2.6-1).

### 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 also be modified to predict the thermal performance in a test-chamber environment. That is, the models are frequently used, with appropriate changes to represent known test chamber configurations, to develop the proper environments for thermal balance test cases and to develop the proper controls for thermal vacuum test levels. Frequently it is not possible to provide a direct, one-to-one test environment to simulate the space environment (e.g., chamber walls are warmer than space, or heater plates are used in lieu of solar simulation, or a solar simulator does not exactly match the spectrum or collimation angle, etc.), so it is necessary to use the analytical model to establish the conservative hot and cold test environments.

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, evaluating the as-built thermal control system, and for improving thermal math model accuracy. The verified analytical model can then be used to predict response to untested cases as well as generating flight temperature predictions.

### 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 bound the flight worst case thermal environment. The closer the simulation is to the spectrum, intensity and the worst case environments, the less reliance on the thermal analytical model to verify the adequacy of the thermal design. Appropriate consideration shall be given to account for the effects of shadowing, blockage, and/or reflections (both diffuse and specular) in the flight and test configurations that either are needed for an accurate simulation, or are artifacts that could adversely affect the simulation. Methods of simulation and the major assumptions for a successful test are described below:

- a. Solar input - Solar inputs can be simulated by mercury-xenon, xenon, or carbon arc source, cryopanel, and/or heaters as described below. The spectrum and uniformity of the source used to simulate the sun and planet albedo must be understood. While the spectral mismatch does not significantly affect the emissivity, the effect on the absorptivity can be large and should therefore be determined and compensated for in the test and/or analysis.

Cryopanel/heater plates can also be used to simulate solar flux by setting the temperature to achieve the same heat flux as would be seen in flight. Flux controlled heaters can directly input the flight solar load onto a component.

- b. Planetary Input - 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 is simulated at the exterior surface of the payload using I2R heaters. The absorptivity and incident radiation are used to calculate the absorbed energy to be simulated.
  - (2) Cryopanel/Heater Plates - This can be an acceptable test if the payload outer skins are not to be touched. The same information is needed for the plates as for the skin heaters and the exchange factor between the plates/cryopanel and the payload must be known. In both cases, a balance equation considering absorptivity, emissivity, incident and rejected energies must be solved to establish accurate test conditions.
  - (3) Quartz Lamps - This is an acceptable method of inputting earth emissions (and solar) so long as the differences in spectrum are measured and the input is adjusted. One technique used to monitor and control lamps is to place calorimeters at the skin of the payload to measure, in situ, the incident energy from the lamps.
  - (4) Calrods - This is also an acceptable method of inputting earth emissions and solar energy to the payload. Again, a technique used to monitor and control the energy input is to place calorimeters at the skin of the payload.
- c. Interfaces – Conductive interface temperatures may be simulated with cold plates that are held at worst-case boundary conditions. Their temperature can be varied for cold flight, hot flight, and safhold conditions or parametrically varied.

Since the payload must be supported during testing there is generally a non-flight conductive heat flow path that is, in flight, a radiative interface, usually with the space environment (e.g., the launch vehicle attachment interface). As much conductive isolation as possible should be used between the test article and this non-flight conductive interface. A heater is placed on the test fixture side of such a conductive interface and two temperature sensors spanning the interface are used. The heater is controlled until the temperatures of the two sensors are the same, thereby minimizing

the heat flow through this path. Without good isolation here, it is likely that an unrealistic and hard to quantify bias will be introduced at this interface, making the test results difficult to assess. Isolation is typically achieved by using fiberglass standoffs. However, the payload may need to be suspended with low conductance cables if the system has a high sensitivity to small heat flows.

- d. Radiative Sink Temperature – The overall radiative sink temperature is typically achieved by varying the chamber shroud temperature. Three typical temperature regimes of chambers are (1) Flooded with Liquid Nitrogen, (LN2 approximately 80-90 K), (2) Controlled with Gaseous Nitrogen (GN2 approx. 170-375 K), and (3) Liquid Helium (20-30 K).

Sink temperatures for individual radiators and critical surfaces are controlled with cryopanel. Cryopanel for cryogenic systems may require special enhancements, for example, open-face honeycomb radiators to increase emittance values. Three typical temperature regimes of cryopanel are (1) GN2 (approximately 130-375 K), (2) LN2 (approximately 80-90 K), and (3) Helium (approximately 20-30 K). For temperatures in between these values heaters can be added to the cryopanel or a heater plate that is conductively coupled to the cryopanel can be used.

A single effective sink temperature is calculated using spacecraft thermal math models that encompass the effects of solar, Earth IR, Albedo and IR effects from other spacecraft surfaces (i.e. backloading), with the appropriate correction for graybody radiation. Test and flight predicts of the energy flow from critical surfaces should be compared. Predictions of both the energy flow and temperatures from the test model should be at least as severe as calculated in the flight model.

- e. Cryogenic Payloads - For cryogenic payloads, chamber walls and/or cryopanel may need to be colder than Liquid Nitrogen temperatures to adequately reject heat. Temperature variations of emissivity should be taken into account in the sink temperature determination analysis.
- f. Dewar Systems – A test dewar may be necessary to simulate the conditions that a payload would see inside a flight dewar. The cooling in a test dewar is available over the temperature range of approximately 0.3 to 80 Kelvin (with gaps). The dewar system may utilize solid cryogen, (i.e. Argon, Nitrogen, Neon or Hydrogen) or liquid cryogen (i.e. helium, nitrogen). During ground testing there is a gravity effect on cryogen that is not seen in flight. Interfaces between the top of the dewar and the payload may be warmer than what would be seen in flight.
- g. Coolers - Thermoelectric Coolers are semiconductor-based electronic components that function as a small heat pump. Heat moves through the module in proportion to the applied voltage. The devices offer active cooling and precise controllability and are used primarily for “spot cooling” (cooling of a single component).

Coolers are also used to recycle cryogen in a closed loop system. This reduces the amount of cryogen needed during a test. This is frequently done when helium is used to reduce cost.

- h. Zero-Q - Certain test-peculiar conductive paths, such as test cables attached to the thermal balance test article, are controlled so that non-flight-like heat does not flow into or out of the test article. During thermal balance the test cabling is minimized. If possible, hat couplers, stimuli, and other non-flight GSE should not be present during thermal balance testing. At a minimum, necessary test cables are wrapped with

multilayer insulation (MLI) for a sufficient distance from the test article. A more positive method of control is to place a guard heater on the test cable a short distance from the test article, place two temperature sensors spanning the interface, one on the spacecraft at the connector, the other on the test cable at the connector, and wrap the cable and heater with MLI to a sufficient distance from the test article. The heater is controlled until the temperatures of the two sensors are the same, thereby minimizing the heat flow through this path.

- i. Avoiding Contamination – Refer to section 2.6.2.2.b
- j. Hardware Orientation - Heat pipes, CPLs, LHPs and other two-phase heat transfer devices will be affected by component orientation in the 1g environment, thus limiting a 0g simulation in the test environment. Test planning should strive for orientations of flight hardware that position these devices in a gravity neutral or reflux orientation to assure their operation in the test configuration. Hardware levelness or other orientation requirements should be verified in the test chamber, prior to pump down.

#### 2.6.3.4 Internal Power

Power dissipation of individual components should be measured to an accuracy of 1% at voltage and temperature extremes during prior (component) testing. Subassembly testing should verify internal power dissipations and line losses, if possible. Prior to spacecraft level testing, the Project should provide: (1) details on what can be directly measured using current/voltage monitors, (2) how this information, in conjunction with component/subassembly test data, will be used to determine individual component dissipations during the spacecraft test, and (3) a plan to resolve discrepancies during test.

#### 2.6.3.5 Special Considerations

The test article shall be thermally coated and the mounting surface of components within the test article (as applicable) shall have the same treatment as it will have for flight.

Extraneous effects such as gaseous conduction in residual atmosphere should be kept negligible by vacuum conditions in the chamber; pressures below  $1.33 \times 10^{-3}$  Pa ( $1 \times 10^{-5}$  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.

#### 2.6.3.6 Demonstration

The number of energy balance conditions simulated during the test shall be sufficient to verify the thermal design and analytical model. To verify and correlate the thermal analytical model, a minimum of three test cases is required. It should be noted, however, that the number of variables associated with a thermal analytical model is large compared to the number of thermal balance cases that can be practically included in a test. The verification of the thermal design, whenever possible, should therefore be accomplished by using test environments that bound the worst hot and cold flight environments such that the test results directly validate the adequacy of the thermal design. The duration of the thermal balance test depends on the mission, payload design, payload operating modes, and times to reach stabilization. Stabilization is considered to have been achieved when the control sensors change less than  $0.05^{\circ}\text{C}$  per hour, for a period of not less than six hours, and exhibit a decreasing temperature slope over that period. Alternatively, another stabilization criterion which may be used is 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 and shall be determined by the Thermal Subsystem Engineer. Cryogenic payloads typically require tighter stabilization criteria and therefore have longer stabilization times; the criteria must be established by the thermal engineer.

The differences allowed between predicted and measured temperatures are determined by the cognizant Thermal Subsystem Engineer and verification of the thermal analytical model is considered accomplished if the established criteria are met. This criterion should be established prior to the environmental testing.

#### 2.6.3.7 Acceptance Requirements

The full qualification thermal balance test may be waived, but only if sufficient margin is known to exist and other tests are conducted to verify the thermal similarity to the previously qualified hardware. In addition, other metrics such as thermaloptical property measurements of flight coatings, component level tests, and review and verification of manufacturing and installation procedures for thermal hardware are shown to exist which preclude full re-verification testing.

#### 2.6.4 Temperature-Humidity Verification

##### 2.6.4.1 Temperature-Humidity Verification: Manned Spaces

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

##### 2.6.4.1.1 Applicability

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

##### 2.6.4.1.2 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 in 2.6.2.4) shall apply.

##### 2.6.4.2 Temperature-Humidity Verification: Descent and Landing

Hardware that is to undergo a specified temperature and humidity environment during a re-entry and that must survive this re-entry 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.

2.6.4.2.1 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.

2.6.4.2.2 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 checkout of the preceding test, those results may be sufficient.

The temperature and humidity profiles in Figure 2.6-2 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.

2.6.4.2.3 Acceptance Requirements

The above provisions apply for the acceptance of previously qualified hardware.

2.6.4.3 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.

2.6.4.3.1 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.

2.6.4.3.2 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 humidity 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.

#### 2.6.4.3.3 Acceptance Requirements

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

#### 2.6.5 Leakage (Integrity Verification)

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

##### 2.6.5.1 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.

##### 2.6.5.2 Demonstration

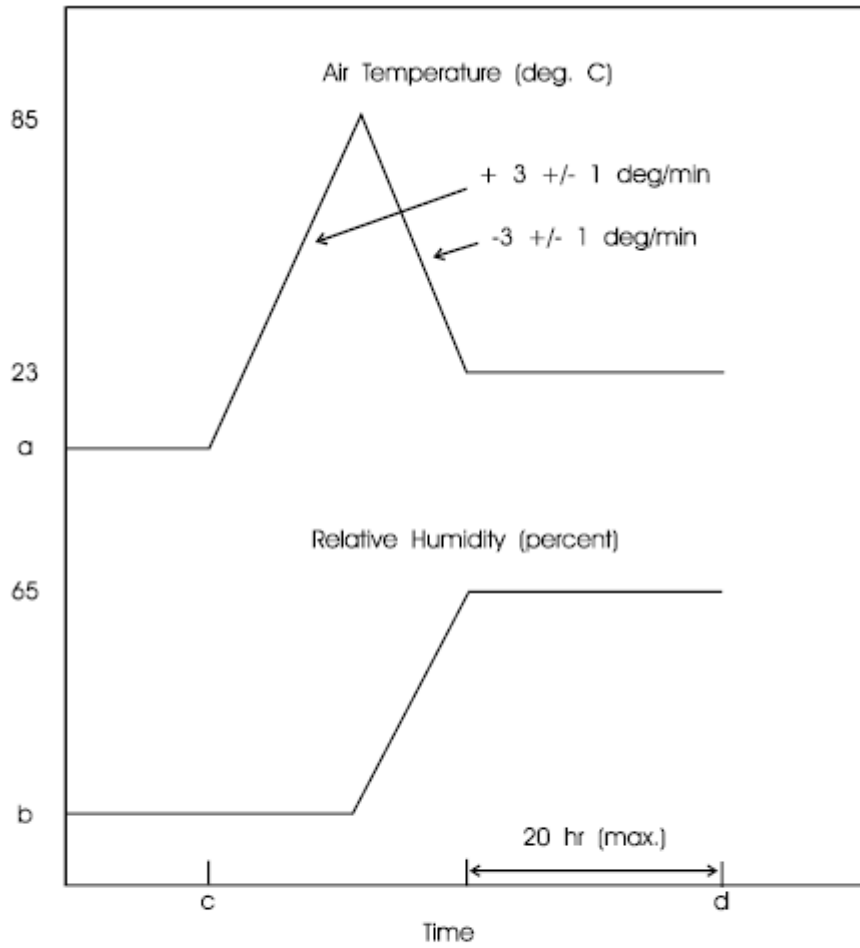
Leakage rates are checked before and after stress-inducing portions of the verification program. The final check may be conducted during the final thermal-vacuum test.

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.

##### 2.6.5.3 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 2.6-3 Temperature-Humidity Profile for Descent and Landing Demonstration

## SECTION 2.7

### CONTAMINATION CONTROL



## 2.7 CONTAMINATION AND COATINGS ENGINEERING, AND PLANETARY PROTECTION

The objective of the Contamination and Coatings Engineering Program is to decrease the likelihood that the performance of flight hardware, engineering test units, and GSE, and research endeavors, will not be unacceptably degraded by contaminants. The objective of a Planetary Protection Program is to protect solar system bodies (*i.e.*, planets, moons, comets, and asteroids) from contamination from Earth life, and to protect Earth from possible life forms that may be returned from other solar system bodies. This Section is organized as follows:

- 2.7.1 Contamination Engineering
- 2.7.2 Coatings Engineering (both optical and thermal)
- 2.7.3 Planetary Protection

### 2.7.1 Contamination

Since contamination control programs are dependent on the specific mission goals, instrument designs, planned operating scenarios, etc. it is necessary for each program to develop contamination requirements for each sensitive element based on contamination susceptibility, performance and lifetime requirements and cross-contamination potential. From the overall lifetime allowable contamination requirements, an allowable contamination budget will then be developed to allocate that amount among the various mission phases, so that the total end-of-life limit will be achieved. A governing Contamination Control Plan (CCP) which defines the complete contamination control program to be implemented for the mission will be written. The specific verification plans and requirements must be defined in the CCP. The supporting procedures that follow provide an organized approach to the attainment of the objectives so that the allowable contamination limit is not violated during each mission phase. The contamination engineering approach commences with concept and continues through end-of-life for the mission.

#### 2.7.1.1 Summary of Contamination Verification Process

- Determination of contamination sensitivity;
- Determination of a contamination requirements;
- Determination of a contamination budget;
- Development and implementation of a contamination control plan and supporting documents;
- Development of contamination verification plans;
- Performance of analytical modeling to predict contamination deposition; and comparison of prediction results to contamination requirements;
- Performance of monitoring of hardware surfaces, air cleanliness, cleanrooms, purges, etc. to verify that requirements are being met.
- Ongoing comparison between hardware cleanliness levels, and/or witness plate cleanliness levels, versus the contamination budget requirements for that phase of build-up or integration.
- If at any time there is a noncompliance between the cleanliness level of the hardware and the requirement level expected for that phase of the mission, the Contamination Engineer shall notify the Project. Together, the CC Engineer and the Project will determine the cause of the non-compliance and put together a corrective action plan to ensure requirements will be met.
- Such corrective action may include: additional cleaning, covers, purges, improving the level of the cleanroom, limiting cleanroom activities, re-assessing the requirement, etc.

Each of the above activities shall be documented at each mission phase and submitted to the project manager for concurrence and approval. The Contamination Engineer shall keep track of requirements compliance, and of monitoring/verification data during each mission phase. Should requirements be exceeded at any time, the Contamination Engineer will enact a process or approach for mitigating the requirements excursion and bringing the levels back within limits.

#### 2.7.1.2 Contamination Engineering Approach

There is a general approach to performing contamination engineering for a project, which includes identifying requirements, performing analyses, verifying requirements through analytical methods, and later through monitoring methods. Should analytical results or monitoring methods show that requirements have been exceeded, there are a variety of corrective actions, additional mitigation methods, and further verification that shall be exercised until requirements are met.

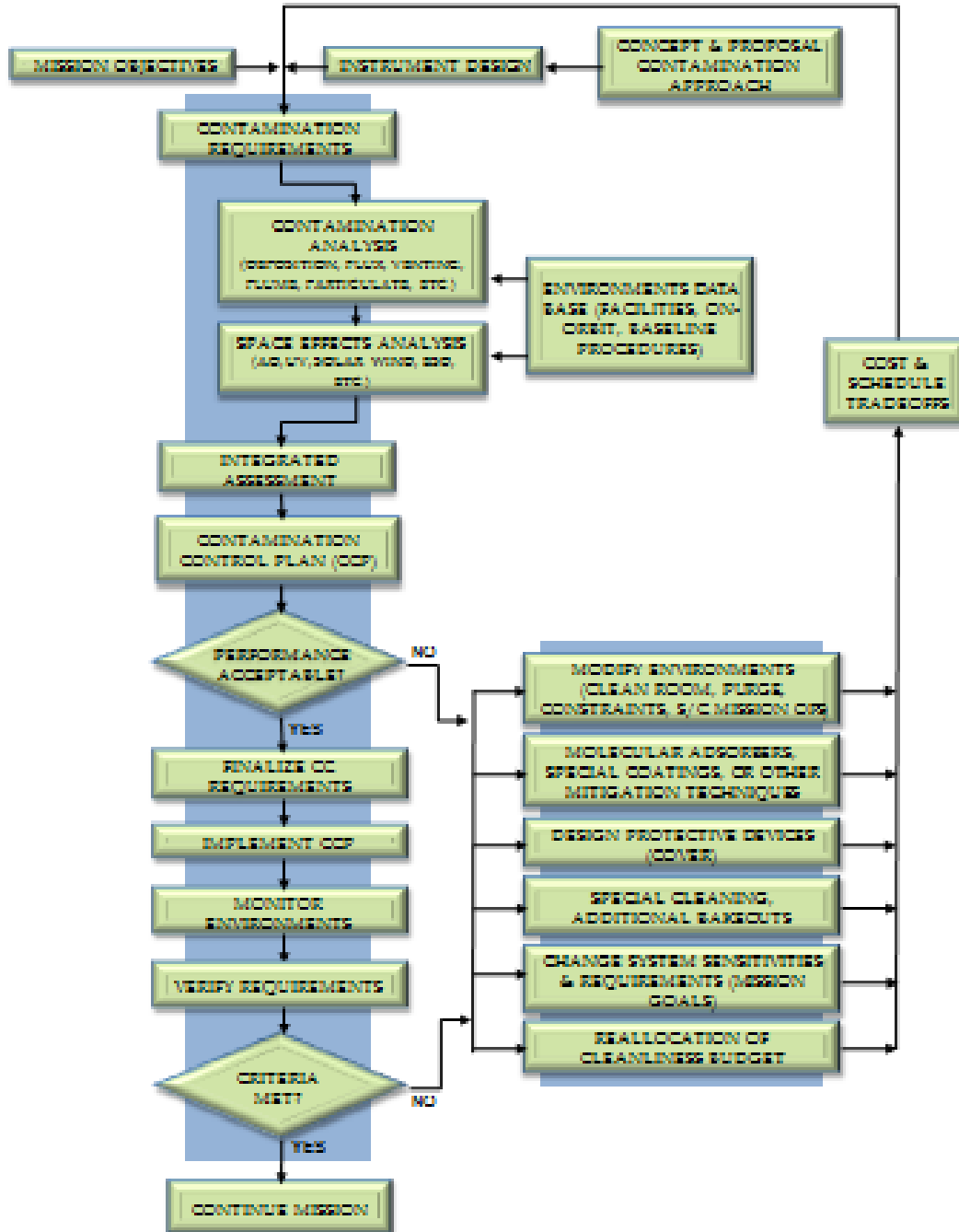
The basic approach which shall be followed for all Contamination programs is:

- a) Assemble information on the spacecraft performance goals and design requirements.
- b) Identify sensitive surfaces, components, and systems and assign quantitative contamination requirements for each element.
- c) Evaluate information, history and flight data from previous missions; apply applicable "lessons learned" to current program.
- d) Design, document, and implement a comprehensive contamination control program for the spacecraft beginning with the concept definition phase, and continuing on through fabrication, assembly, integration and test, transport, launch, and on-orbit, and post-mission or retrieval mission phases.
- e) Study spacecraft design and identify potential "problem" areas. (e.g. acceptable vs. non-acceptable vent locations).
- f) Perform trade-off studies to evaluate technical adequacy, cost and schedule impacts, of proposed contamination control measures (e.g. performing bakeouts of components vs. incorporating Molecular Adsorbers).
- g) Perform laboratory testing whenever needed, to assess contamination potentials of materials, and to evaluate contamination levels versus performance degradation.
- h) Utilize special contamination mitigation devices and techniques, when appropriate: molecular adsorbers, special coatings, on-orbit covers for sensitive apertures, on-orbit heaters, special vent placements, etc.
- i) Work with the project to include on-orbit flight contamination monitors to measure mission contamination levels, if contamination is critical to the mission.
- j) Perform analytical modeling to predict expected contamination deposition levels for sensitive surfaces.
- k) Perform cleaning, vacuum bakeouts, and implement protection devices (covers, bags, containers, etc.) to minimize resultant contamination levels.
- l) Perform adequate monitoring and verification of contamination levels on and near the spacecraft during all assembly, integration and testing, transport, storage, and launch readiness mission phases.
- m) Monitor launch and on-orbit contamination environments, and/or evaluate the performance of spacecraft systems to determine effects of contamination.
- n) Develop "lessons learned" list for implementation on follow-on projects.

The below flowchart illustrates this process.

**Figure 2.7-1 Contamination Engineering Approach**

## CONTAMINATION ENGINEERING APPROACH



### 2.7.1.3 Contamination Sensitivities

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 to other contamination sensitive hardware. 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/materials that may contaminate and cause unacceptable degradation of the test item. Contamination is any substance that can cause deleterious effects on performance of the mission either through residence of material on a surface, deposition of nonvolatile residue, particles, degradation of a surface, cross contamination, bulk material outgassing, or interaction of any ambient, vacuum, or space environment with the surface or material returning to the surface.

If the assessment indicates a likelihood that contamination will degrade performance of the flight item or other sensitive hardware associated with the mission, a contamination control program shall be instituted. The severity of contamination program 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.

### 2.7.1.4 Contamination Requirements

The amount of degradation of science performance that is allowed for critical, contamination-sensitive items shall be established, usually by the Project Scientist with support from the project contamination engineer. Likewise, the amount of permissible degradation of key performance properties (thermal, optical, mechanical, etc.) of other contamination sensitive hardware (optics, solar arrays, thermal coatings, star trackers, mechanisms, power systems, etc.) that are necessary to support the mission shall be determined by the designer of the hardware or the mission systems engineer. From these limits, the amount of contamination that can be tolerated, or the contamination requirements shall be established. The rationale for such determination and the ways in which contaminants will cause degradation is the mission level requirement and shall be described in the contamination control plan, and any project documentation or contract documentation necessary to ensure these contamination requirements are achieved.

### 2.7.1.5 Contamination Budget

Contamination budgets (a breakdown of the overall end-of-life allowable contamination levels) for allowable surface accumulation shall be developed for all elements of the flight hardware and, when applicable, critical ground support hardware. Comprehensive outgassing rates may need to be developed to meet the specified budget allotments. The budgeted levels shall reflect both the hardware's own sensitivity to contamination as well as the ability of the hardware to cross-contaminate to other hardware associated with the mission. The budget shall describe the required outgassing rates and surface cleanliness levels through all phases of ground and mission operations up to end-of-life. The budgets shall be expressed in terms of verifiable requirements – e.g, outgassing rates, partial pressures, surface cleanliness levels, visual cleanliness levels, etc. The necessary contaminant transport modeling and materials properties testing shall be conducted as required for the various environments and factors which affect contaminant generation, transport and accumulation (e.g, mass transport in vacuum, particle fallout in cleanroom,

particle redistribution during launch, atomic oxygen, radiation, uv, etc. ) to validate the budgeting efforts. 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.

Cleaning, thermal vacuum bakeout, and other mitigations may be used to bring hardware into compliance with the budget. When such mitigations cannot be performed or can only be performed a limited number of times, then protective measures may be required to maintain the contamination budgets throughout ground, launch, and post-launch operations. Such measures could include: bagging, containers, GSE/flight barriers, baffles, covers or doors, purging, etc. Contamination avoidance methods, such as cleanrooms and instrument covers, will affect the budget and a general description of their usage should be included.

#### 2.7.1.6 Contamination Control Plan (CCP)

The CCP is the most important contamination control document for any program. A detailed contamination control plan shall be prepared that describes the requirements and procedures to be followed to control contamination. It shall establish the implementation plans 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 hardware lifetime.

From the overall end-of-life allowable deposition requirements, the breakdown of specific allowable contamination levels, at progressive points in the build-up, integration, testing, launch readiness process, and on-orbit mission phases should be derived. This breakdown is often called a contamination "budget" and should be clearly presented in the CCP.

The CCP should also specify when and by what methods the various contamination requirements will be verified. It is good contamination practice to include frequent verification of contamination levels (especially during events) so that problems associated with excess contamination levels can be identified and solved as soon as possible.

The CCP should also present the overall plan for controlling contamination, from fabrication and assembly, throughout integration and testing, and continuing with launch site, launch, and on-orbit plans. Any laboratory and analytical support should also be identified in the CCP. All necessary supporting documents should be referenced in the CCP.

##### 2.7.1.6.1 Supporting Contamination Documentation

The following documentation list represents the type of supporting documents which may be required for a project:

- CCP Implementation Plan
- Hardware Cleaning Procedures
- Hardware Cleanliness Verification Procedures
  - Molecular Wash Method
  - Molecular Wipe Method
  - Particulate Tape Lift Sampling
  - Particle Counting Method
  - Optical Measurement Methods
  - Actual instrument throughput/performance measurements
- Witness Plate Measurements
- Cleanroom Personnel Training/Certification Documents

- Cleanroom Personnel Operations Requirements
- Cleanroom Operating Procedures
- Approved Materials Lists and processes
- Cleanroom Monitoring Methods
- Purging Plans
- Hardware Bagging Requirements
- Contamination Protection Methods
- Material/Hardware Outgassing Certification Plans
- Thermal Vacuum Bakeout Plans and Procedures
- Transportation/Storage Cleanliness Plans
- Launch Vehicle/Payload Interface Documents
- Launch Site Cleanliness Requirements
- Launch/On-orbit Contamination Requirements
- Testing Results Reports (Thermal Vacuum Bakeouts, Vibration Testing, Functional Testing, etc.)
- Hardware Certification Logs and Anomaly Reports
- Lessons Learned Summary Report

The basic standards and practices, and some procedural documents already exist in open literature and are controlled by various technical societies (ASTM, IES, etc.). A spacecraft project may utilize these documents and reference them in the higher level documents, rather than develop completely new documents.

#### 2.7.1.7 Contamination Engineering During Design

From the earliest stages of a program, iterative analyses must be done to determine contamination sensitivities of hardware and contamination transport (molecular outgassing, diffusion, particle redistribution, contact transfer, plume, venting, etc.) These analyses need to consider not only the hardware's own contamination sensitivity but also the sensitivity of other hardware and the allowable degradation of performance – especially science -- by contamination. Likewise, a general plan for achieving the outgassing, surface cleanliness, and other critical cleanliness levels should be formulated. It should identify elements that will significantly affect the complexity or difficulty of the contamination control program so that the appropriate cost, schedule, performance, and risk trades may be made in conjunction with the project management. The need for protective design elements (reducing lines of sight of contaminating surfaces to contamination sensitive surfaces, doors, purges, cold cups, molecular traps, cleanable surfaces, bakeout flight heaters, etc.) to assure performance or reduce overall programmatic cost should be identified as early as possible in the program.

Standard material selection requirements that are often part of contracts and specifications for hardware may need to be made more stringent to cost effectively achieve performance requirements. The contamination engineer shall determine if more stringent criteria (particle generation, water content, outgassing, "as-used" temperatures, etc.) are required as early as possible in a program to minimize impact to cost and schedule. At a minimum, established material screening criteria for vacuum stability (ASTM E 595, JSC 0700 Vol. XIV, and NASA Reference Publication 1124) should be imposed in project requirements documents. These screening criteria exist not only for contamination considerations, but also to assure that material properties remain stable after exposure to vacuum. While useful for many applications, these screening criteria are insufficient for many state-of-the-art instruments and spacecraft designs. Material selection criteria for particle generation, cleanability, specific chemical restrictions, and other unique mission specific parameters may have to be

defined and implemented in the appropriate contractual and programmatic documents. The ASTM E1559 tests material outgassing according to on-orbit source and receiver predicted temperatures.

#### 2.7.1.8 Contamination Engineering Analyses

It is recommended that a detailed analysis plan be developed for each spacecraft mission. Usually, a preliminary analysis is performed early in the program, so that results may be used to aid in making design decisions. Then, once the final design is established, a detailed analysis is performed and fine-tuned. This stage aids in verifying the spacecraft design and performance expectations, and is also used to set bakeout acceptance criteria for the hardware. A final, flight prediction analysis is typically performed near the end of spacecraft integration and test, to establish the final estimates of on-orbit contamination levels.

##### 2.7.1.8.1 Molecular Contamination Analyses

These analyses generally consist of utilizing an existing analytical tool (e.g. Molflux, CAP, ISEM, DSMC, and an entire library of plume definition/effects tools), creating a geometric model of the spacecraft (with critical surfaces well-defined), assigning materials and materials outgassing rates to each surface, assigning temperature profiles to each surface, and then exercising the code. Results are usually reported in mass/unit area, and additional iterations of the runs (with different temperatures, materials, etc.) may easily be accomplished. Once the mass/unit area values are ascertained, it is also possible to perform "effects analyses" to predict the resulting impact on performance. For example, a modeling analysis may predict that 100 Angstroms of silicone will deposit on a critical surface. Then, using different analytical tools (which are usually based on experimental test programs), it is possible to evaluate performance of the optical system with this coating of 100 Angstroms of silicone. A 100 Angstrom layer of silicone, within a UV instrument, can mean significant degradation. Additionally, if one considers specific on-orbit parameters (such as solar exposure), it is possible to further assess the impacts via analytical methods.

Materials properties become an important input factor in the molecular analyses. Outgassing rates, and reemission rates are both input into the codes. The industry has, over the years, developed an extensive materials properties data base, and where possible, the input data is obtained from these previously performed materials tests. It is often necessary to perform materials outgassing testing on specific materials, for which no other data exists.

##### 2.7.1.8.2 Particulate Contamination Analyses

The particle analyses vary in methodology, depending on the mission phase. Differing physical principals apply to the various mission phases (e.g., particle fallout values are different in 1g environments versus 0 g environments).

For the ground-based assembly, integration, test, transport, storage, and prelaunch phases, calculations are made based on exposure to the various cleanroom environments. Parameters that affect this analysis are:

- How clean is the air?
- How long is the spacecraft exposed?
- Number of personnel working on and near the spacecraft
- Other activities performed near the spacecraft (drilling, sanding, painting, etc.)

- Test chamber attributes and operations
- Number of scheduled cleanings
- Methods of protection employed

Based on these analyses, it is possible to identify potential hazards or threatening timeframes for the spacecraft critical surfaces, and preparations to protect the spacecraft may be made, well in advance of the activities. In addition, these analysis results may be used to determine the schedule of cleaning for the spacecraft. The analysis may show that contamination builds up quickly in one facility versus another, and more frequent hardware cleaning may be advisable.

For the launch period, vibration and acoustics levels act to remove and relocate particles from surfaces. In addition, the changing gravity levels, and venting of the spacecraft and launch vehicle payload volume become important influencing parameters. There are models and codes to evaluate these events, and determine the resulting particle redistribution during launch.

During the on-orbit mission phases, particles become dislodged from surfaces due to spacecraft operations (launch vibro-acoustic modes, solar array openings, aperture cover ejections, attitude and altitude changes) and other phenomena such as micrometeoroid impacts. These particles tend to be ejected into a type of "orbit" around the spacecraft and could potentially interfere with instrument and sensor viewing. There are codes that predict these trajectories (based on particle size, shape, relative velocities of spacecraft and particle, and mass), and then predict the potential for re-encounter with the spacecraft, and subsequent deposition.

In all cases, once the particle deposition predictions are determined, it is then possible to perform effects analyses. For example, after predicting the particle deposition (number and sizes of particles) on an optical element, analyses may be performed to assess the scattering associated with the particles on a mirror, or the transmission loss due to particles obscuring a lens. With these analysis results, spacecraft performance may be predicted, and corrective actions may be taken. At the very least, by knowing the level of particle contamination and the predicted effects, it is possible to more effectively evaluate the flight data (making data corrections for the spurious particulate effects).

#### 2.7.1.8.3 Other Analyses

There are many other analyses performed for spacecraft programs that are related to contamination. These include:

- Atomic Oxygen Prediction Analyses
- Predicts atomic oxygen fluence
- Can predict materials erosion rates
  - Combined Effects Analyses (e.g. UV plus Atomic Oxygen)
  - Contamination Effects Analyses
- Thermal properties changes
- Optical properties changes
- Lifetime and performance impacts
  - Materials Properties Analyses
- Specific analyses on materials outgassing rates and cumulative amounts
- Particulate generation analyses for specific materials
- Materials aging studies



#### 2.7.1.9 Contamination Engineering During Integration and Test

The goal of the integration and testing phase is to perform the necessary activities without compromising the cleanliness integrity of the spacecraft hardware. It should be noted, here, that there are generally some testing activities which must be accomplished in non-cleanroom environments (e.g. vibration/acoustics testing facilities are typically not cleanrooms). Significant pre-planning, the use of protection devices (covers, bags, etc.), and frequent sampling of cleanliness levels, must be implemented for these testing periods. Contamination control during the integration phase becomes crucial to the success of the overall mission. Many programs fall short during this phase, without realizing that this is where the strictest of attention to details such as contamination control become all-important.

During spacecraft integration, all systems, subsystems, boxes, solar arrays, etc. are merged together and now constitute the entire spacecraft program. It is especially important to identify the most critical elements (usually the optical and thermal control elements) and to plan integration activities, while keeping in mind the strict contamination requirements of these crucial systems.

The spacecraft is generally maintained only in cleanroom environments and bagging and protection of the entire spacecraft is recommended during all downtimes.

A major part of the development of the contamination control plan is to develop a cost-effective integration and test plan. There are often many possible means of maintaining a particular cleanliness level. It is important to work with the program to determine which solution(s) best meet the project's overall cost, risk, schedule, and, most importantly, mission success criteria. Critical elements that contribute to a program's overall cost and success include: type of facility, gowning requirements, Ground Support Equipment design and cleanliness, cleaning processes, thermal vacuum plans and tests, bagging provisions, shipping and handling provisions, contamination verification and testing, etc. Contamination requirements need to be established in the contamination control program that addresses these elements and any others that can prevent contamination or reduce the risk of contamination.

A thorough cleanliness monitoring and verification program must be enacted during integration and testing activities, and checked regularly. Each major integration event and testing activity must be separately monitored to characterize the levels of contamination which may have occurred during that unique period. This serves to prove that the integration and/or testing event was a "clean" event, or aids in forensics should there be an anomaly during one of these activities. All of the testing requirements are delineated in the CCP and supporting documents.

#### 2.7.1.10 Thermal Vacuum Bakeouts

The importance of the thermal vacuum bakeout and certification phases for space hardware has become a dominant factor in Contamination Engineering planning. Experience has shown that taking the time, at this point in the program development, to fully bakeout and certify each component, assembly, subsystem, etc., has significant pay-offs later during the on-orbit mission phase, reference Hubble Space Telescope's Wide field Planetary Camera II ultraviolet performance. Clearly, by reducing the spacecraft outgassing levels in a controlled vacuum chamber during this ground phase, the amount of material left to outgas during on-orbit periods is diminished. For those spacecraft and instruments which are sensitive to

even small amounts of molecular contamination, it is important to plan for and cost-out a thorough vacuum bakeout and certification program.

The general philosophy behind vacuum bakeouts and certification of space hardware is as follows:

- Commence with bakeouts at the parts level, if possible. Bakeout hardware at the highest temperature possible.
- Set quantitative acceptance criteria for each bakeout. These criteria should be based on the total allowable outgassing level for the spacecraft. Usually computer modeling analyses aid in the determination of the allowable outgassing level for hardware.
- Do not accept hardware which does not meet the acceptance criteria or adjust the outgassing cleanliness budget accordingly.
- As parts become assembled, often, the maximum temperature limits become lower. Bakeouts at lower temperatures are less effective, and take considerably longer to complete.
- Develop a detailed Thermal Vacuum Bakeout and Certification Plan for your program. Include the overall plan, schedule, need for instrumentation, and QCM acceptance criteria, as well as a list of the responsible individuals to contact during the test.

#### 2.7.1.11 Spacecraft Transportation and Storage Phases

During transportation and storage, it is important to preserve the state of cleanliness of the space hardware. Generally this means sufficiently protecting the hardware via covers, containers, bagging, etc. and often means employing a high purity purging of flight hardware. Develop a purge plan early in the program to accommodate purge system procurement.

#### 2.7.1.12 Launch Site Contamination Control

The purpose of a Launch Site Contamination Control Plan (LSCCP) is to identify all contamination requirements, all interface requirements, all necessary equipment and supplies needed at the launch site, all necessary personnel requirements, an overall schedule, and a detailed plan of activities for the launch site. Contamination Engineers shall work with the launch site teams, spacecraft and instrument teams, and support launch site activities up through launch.

#### 2.7.1.13 Launch Vehicle and Companion Payload Considerations

The following information must be obtained and integrated into the overall contamination control program: Obtain and review system information regarding the launch vehicle and companion payloads.

- Evaluate the mission scenario and timeline.
- Identify potential contamination sources; attempt to qualitatively prioritize the magnitude of threat posed by the launch vehicle and companion payload sources:
  - Fairing materials
  - Fairing surface cleanliness levels
  - Primary engine firings and plumes
  - Secondary engine firings and plumes

- Launch vehicle venting
- Launch pad purging and T-0 purges, including materials and certifications of gas at point of delivery.
- Companion payload materials and venting
- Companion payload ejection and orbit insertion (springs, pyrotechnic devices, engine firings, etc.)
- Effects of Helium on spacecraft system performance as applicable.
- Perform modeling and plume analyses (typically during CDR timeframe) to quantify contamination threats.
- Develop resolution methods, if required, such as:
  - Implement a change-out or substitution of specific materials
  - Improve cleanliness levels of launch vehicle and/or companion payload (such as performing additional vacuum bakeout of components, plan for more rigorous cleaning of surrounding launch vehicle or companion payload surfaces, etc.)
  - Design protective shields or barriers for your spacecraft or
  - Change ejection and orbit insertion methods

#### 2.7.1.14 Launch and Orbit Insertion Mission Phases

There are several parameters associated with the launch and orbit insertion mission phases which may be discussed and adjusted to fit individual spacecraft needs. For example, it may be possible to work with the launch vehicle team to adjust the retro maneuvers to minimize plume impingement on the released spacecraft. Usually, a detailed analysis and justification for these types of adjustments is required, before approval is given.

#### 2.7.1.15 On-Orbit Through End-of-Life Mission Phases

There are a number of actions which may be taken during the on-orbit through end-of-life mission phases to minimize contamination levels and even to “clean-up” certain surfaces while on-orbit. Careful planning and in some cases, special equipment may be necessary to carry out contamination control measures at this stage of a mission.

For spacecraft which are sensitive to molecular contaminants, all on-orbit sources must be minimized. This may include inhibiting engine firings, redirecting vents and other high outgassing sources.

For instruments which require clear FOV operations, it may be necessary to inhibit certain on-orbit operations during viewing times. Inhibits of vents, mechanisms, solar array movement, engine firings, and any other particle “jostling” activities may be required.

In the case of spacecraft which are sensitive to photopolymerized molecular contaminants, it is recommended that solar exposure be limited. This may mean designing on-orbit maneuvers such that sensitive elements are not exposed to solar illumination, thus minimizing the risk of photopolymerization of contaminants.

The planning for the operations of any aperture doors or covers must be evaluated well in advance of the actual flight. It may be necessary, however, based on actual mission circumstances, to utilize the aperture doors and covers in order to prevent further contamination of critical surfaces.

In the case of lower altitude spacecraft, with surfaces which are vulnerable to the effects of atomic oxygen degradation (erosion), it is recommended that sensitive surfaces never be oriented into the RAM direction. Or if this is impractical, it may be possible to design barriers to “shadow” or protect the vulnerable surfaces from the atomic oxygen environment.

Ironically, the atomic oxygen environment may also serve a beneficial purpose for other surfaces. For example, if a fairly stable surface become contaminated with a molecular residue, it may be possible to deliberately orient the surface in the RAM direction so that it is exposed to atomic oxygen impingement, which usually “erodes” away the contaminant layer, leaving the substrate surface once again clean.

It may also be possible not only to deliberately locate vents in areas of the spacecraft which pose little threat the sensitive elements, but also to design vent barriers or deflectors to deflect venting products (usually considered contaminants) away from sensitive spacecraft elements. Molecular Adsorber pucks may be utilized in vent locations to “capture” molecules in high outgassing volumes.

For sensitive missions, it is always advisable to design and fly an accompanying contamination monitor with the spacecraft so that direct measurements of accumulated contamination may be confirmed.

Another aspect of this mission phase which is often neglected is the ability of engineers to evaluate mission performance data (actual optical measurements, temperature data, etc.) and to derive what the effects of contamination might be. For example, if the temperature of a thermal control surface is rising faster than anticipated, it may be because of layer of contamination has deposited on it, and has changed the absorptance properties of the surface, which in turn is causing the surface to heat up faster. Knowing this information, it may be possible to take corrective action, such as exposing the surface to atomic oxygen (to erode the contaminant layer) or to modify the thermal system (via computer commands) to rely more heavily on another, uncontaminated thermal component for spacecraft temperature control.

If the spacecraft is demonstrating unexpected responses or seemingly incorrect data, is has often been possible to quickly perform modeling analyses, or perform an experimental investigation to simulate the actual environment (temperatures, solar activity, etc.) being experienced by the spacecraft. Then, is has been possible to verify what is happening, through these ground-based investigations, and help to prevent further degradation, or devise scenarios for correcting or compensating for the on-orbit anomaly.

In recent years, much attention has been paid to the possibility of performing on-orbit cleaning of contaminated surfaces and systems. The most obvious method for achieving this is designing heater systems under sensitive elements (mirrors, lenses, detectors, etc.) which can be turned on to heat-up and “outgas” contaminants from surfaces. This method has already be implemented on a number of spacecraft in the past, including the HST.

#### 2.7.1.16 Spacecraft Post-Mission and Follow-on Program Phases

It is often during the post-mission analysis of on-orbit performance that the aerospace community learns the most valuable lessons from the mission. Many program improvements, and “lessons learned” have resulted from taking a clear and complete look at what happened during the life of the spacecraft.

## 2.7.2 Coatings Engineering

Coatings Engineering for a spacecraft mission, instrument, or component involves 3 main-steps. An additional step may be required where a coating need to be qualified:

- **Assessment of Mission Needs:** Selection of coatings for various applications to meet *thermal and optical* performance requirements for individual surfaces or systems is the first step. Determination of Beginning of Life (BOL) and End of Life (EOL) requirements is then assessed. Thermal radiative property predictions and preliminary testing may be required to determine that the selected coatings are valid. Documentation for methods of applications, curing, handling, and special instruction must be developed.
- **Research & Development or New Coatings Development:** If necessary, due to project performance needs, new or tailored coatings may be required and research and development may be required to develop flight worthy coatings. Absorptance tailoring of Coatings, emittance tailoring of coatings, new application techniques, composite coatings, etc. may require development to support the mission.
- **Coatings Application:** Once agreed upon, the application of the Thermal Control Paints, Thin Film Coatings, Dielectric Coatings, Conductive Coatings, Lacquers, Tapes, Molecular Adsorber Coatings, etc. will be performed. Full validation of adhesion, thickness, smoothness, bonding, coverage, and integrity of the application shall be performed after application, until coating is acceptable and meetings requirements.
- **Coatings Flight Qualification and Measurements:** **If a flight coatings needs to be qualified for flight, the coating engineer will design a test program accommodating any special testing required by the project specific requirements, such as: mission parameters, spacecraft configuration, desired thermal/optical coating's properties, coating application, need for a conductive coating, space environmental effects and/or contamination issues.** As dictated by the coatings engineer and the project, a set of post application characterization measurements will be performed to verify the characteristics of the coating. These may include: Optical Property Characterization, Thermal Radiative Property Characterization, Hemispherical Emittance Characterization, Bi-Directional Reflectance Distribution Function (BRDF), Light Scattering/Surface Specularity, Electrostatic Discharge Testing, UV Degradation Testing, Thermal Cycle Testing, Solar Wind Testing, and Outgassing Testing.

To design a thermal control system that address the mission's requirements, the thermal radiative properties and durability of the material must be obtained through thermal optical/radiative property measurement and space environmental testing.

Thermal Optical/Radiative Properties (Reflectance =  $\rho$ , Transmittance =  $\tau$ , Absorptance =  $\alpha$ , and Emittance =  $\epsilon$ ) are used to evaluate a material's ability to maintain temperatures. The reflectance of a material's surface is measured over the Infrared, Visible and Ultraviolet regions of the electromagnetic spectrum to calculate the solar absorptance and over the infrared region to calculate emittance. Thermal coatings are tested for good coating adherence to the substrate through Coating Adhesion testing and/or thermal cycle testing.

### 2.7.2.1 Key Verification Milestones for Coatings Selections and Testing

- System Requirements Review (SRR)
  - Conceptual Design presented - including which coatings are planned for key thermal and optical surfaces.
- Preliminary Design Review (PDR)
  - Major trade studies complete
  - A complete design, meets all requirements within system resources (power, mass, volume, cost, schedule)
  - All coatings for key thermal control surfaces and optical surfaces should now be assigned and preliminary test data available at the review
- Critical Design Review (CDR)
  - Detailed design complete, ready to fabricate, and assemble/integrate
  - Results of development tests, activities
  - Test concepts presented
  - Complete coatings selection, design, application methodology, environmental test data, etc. will be presented at CDR
- Pre-Environmental Design Review (PER)
  - Details since CDR presented: design changes, problems, detailed analysis complete
  - Results of 1<sup>st</sup> complete system functional
  - Details of planned environmental tests: EMI, Mechanical (vibe, acoustics), Thermal Vac & Balance (TB)
  - All coatings will have been applied, tested, and validated.
- Pre-Ship Review (PSR)
  - Results of testing, including problems & resolution; TB correlation, final flight predictions.
  - Plans for ship, launch site ops
  - Any touch-up of coatings will occur as needed
- Launch Readiness Review (LRR)
  - Report on launch site preps, all paperwork complete, staff & plans for L&EO
  - Pre-launch touch-up will occur as needed

### 2.7.3 Planetary Protection

Planetary protection is essential for several important reasons: to preserve our ability to study other worlds as they exist in their natural states; to avoid contamination that would obscure our ability to find life elsewhere.

Planetary Protection during spacecraft design, fabrication, assembly, integration and testing, follows a similar approach to that of contamination control, with the difference that the goal is to minimize, prevent, clean, measure, and verify the microbial contamination levels on spacecraft surfaces.

Planetary Protection encompasses a set of requirements pertaining to spacecraft hardware and missions involving (1) the control of terrestrial microbial contamination associated with robotic space vehicles intended to land, orbit, flyby, or otherwise encounter extraterrestrial solar system bodies, and (2) the control of contamination of the Earth and the Moon by extraterrestrial material collected and returned by robotic missions. Detailed compliance and verification requirements for Planetary Protection programs for NASA missions are found in NPR 8020.12C (or latest version), "Planetary Protection Provisions for Robotic

Extraterrestrial Missions”, and NPD 8020.7G, “Biological Contamination Control for Outbound and Inbound Planetary Spacecraft” (Revalidated 11/25/08).

2.7.3.1 Classification of Missions

Specific planetary protection requirements for each planned mission will be determined by the NASA Planetary Protection Office (PPO), in accordance with the governing documents, and consistent with the policy and guidelines of the Committee on Space Research (COSPAR), recommendations of the Space Studies Board of the National Research Council (NRC), and advice from the NASA Advisory Council. Requests for categorization of missions and associated mission requirements shall be submitted to the PPO during the mission design phase (before the completion of the draft Planetary Protection Plan) by the mission Project Manager. Such correspondence shall be accompanied by a mission description and shall include a request and justification for a specific mission categorization. The PPO will respond, in writing, with the appropriate categorization, conveying such explanatory information or supplemental conditions as may be appropriate. Subsequent approval of a mission's Planetary Protection Plan will constitute formal categorization of the mission.

2.7.3.2 Planetary Protection Categorization of Missions

Each planetary mission will fall into one or more categories based on the planetary protection priorities of each extraterrestrial solar system body and the mission plan. Planetary protection priorities and corresponding mission categories are given in the following table. Each category has increasingly more severe requirements as the Mission Category Level increases. Documentation, microbial cleanliness, sampling, monitoring and verification requirements become more complex. The NASA Headquarters Planetary Protection Office (PPO) is responsible for reviewing each planetary mission, including planned operations, possible unplanned events involving planets, asteroids, comets, etc., and then assigning a Planetary Protection Category and Mission Type to each program. The PPO shall follow the progress and adherence to Planetary Protection requirements and verification plans throughout the life of each mission.

<b>Planetary Targets Priority</b>	<b>Mission Type</b>	<b>Mission Category</b>
Not of direct interest for understanding the process of chemical evolution, or where exploration will not be jeopardized by terrestrial contamination. No protection of such planets is warranted and no requirements are imposed.	Any	I
Of significant interest relative to the process of chemical evolution but only a remote chance that contamination by spacecraft could jeopardize future exploration.	Any	II
Of significant interest relative to the process of chemical evolution and/or the origin of life or for which scientific opinion provides a significant chance of contamination which would jeopardize a future biological experiment or exploration program(s).	Flyby, Orbiter	III
Of significant interest relative to the process of chemical evolution and/or the origin of life or for which scientific opinion provides a significant chance of contamination which	Lander, Probe	IV

would jeopardize biological experiments or exploration programs(s).		
Any Solar System Mission	All Earth-Return	V

Notes:

- 1) For missions that target or encounter multiple planets, more than one category may be specified for planets targeted or encountered.
- 2) For missions utilizing gravity assist means of a flyby of another planet, requirements will usually be those for the target requiring the higher degree of protection.

### 2.7.3.3 Planetary Protection Plan Development (Categories II-V)

Each Planetary Mission (Categories II-V) is required to produce and follow a Planetary Protection Plan. The Planetary Protection Plan shall be the primary planning document describing how a planetary flight project will meet its planetary protection requirements. The Planetary Protection Plan shall indicate planned conformance to those requirements and shall include, as a minimum, the items given in the following outline (see below). It is recognized that each project will prepare various other documents that may adequately cover some of the topics in the outline (e.g., the Project Plan may thoroughly cover the subject of Planetary Protection Management). In such instances, it is suggested that the Planetary Protection Plan include only the major aspects of the topic and that free reference be made to the basic project documents that provide specificity.

The Planetary Protection Plan shall include, but is not limited to, the items given in the following outline:

- ▶ General
- ▶ Introduction
- ▶ NASA Planetary Protection Constraints
- ▶ Designation of Mission Category
- ▶ Planetary Protection Specifications
- ▶ Planetary Protection Management and Organization
- ▶ Organization Description
- ▶ Responsibilities and Relationships
- ▶ System Interface Management
- ▶ Contractor Management
- ▶ Data Management
- ▶ Documentation
- ▶ Identification of References and Applicable Documents
- ▶ Facilities
- ▶ Identification and Description of Controlled Facilities
- ▶ Activities Performed
- ▶ Hardware Affected
- ▶ Schedules
- ▶ Identification of Milestones
- ▶ Preliminary Schedules

In addition, the following subsidiary plans shall be prepared when required for the particular category assigned:

1. Contamination Analysis Plan
2. Microbiological Assay Plan
3. Microbial Reduction Plan
4. Earth Safety Analysis Plan



#### 2.7.3.4 Measurements and Verification

Specific constraints imposed on spacecraft involved in solar system exploration will depend on the nature of the mission and the identity of the target body or bodies. These constraints will take into account current scientific knowledge about the target bodies through recommendations from both internal and external advisory groups, but most notably from the Space Studies Board of the National Academy of Sciences. The most likely constraints on missions of concern will be a requirement to reduce the biological contamination of the spacecraft, coupled with constraints on the spacecraft operating procedures, an inventory of organic constituents of the spacecraft and organic samples and restrictions on the handling and methods by which extraterrestrial samples are returned to Earth. In the majority of missions, there will also be a requirement to document spacecraft flyby operations, spacecraft impact potential and the location of landings or impact points of spacecraft on planetary surfaces or other bodies. Specific requirements (reviews, documentation, and levels of cleanliness) are detailed in implementing procedures and guidelines, primarily NPR 8020.12, "Planetary Protection Provisions for Robotic Extraterrestrial Missions," and will be used to measure adherence to this directive.

#### 2.7.3.5 Documentation Requirements for Planetary Protection Requirements and Verification

Since Planetary Protection includes a similar approach as the contamination control approach, there are a number of documentation requirements, analyses reports, and monitoring reports needed to satisfy Planetary Protection verification requirements. Below is a summary of documentation requirements.

- Category I missions:
  - Certification of mission as Category I relieves a project of all further planetary protection requirements.
- Category II missions:
  - A Planetary Protection Plan outlining intended or potential impact targets.
  - Brief Pre- and Post-Launch Planetary Protection Reports detailing impact avoidance strategies.
  - End-of-Mission Report providing the final actual disposition of launched hardware and impact location.
- Category III missions:
  - A Planetary Protection Plan that details the planned approach to compliance with planetary protection requirements, including subsidiary plans.
  - A Pre-Launch Planetary Protection Report which documents that all requirements have been met (note that an inventory of bulk constituent organics, if the probability of impact is significant, must be included in the Pre-Launch Planetary Protection Report).
  - A Post-Launch Planetary Protection Report that updates the Pre-Launch Planetary Protection Report.
  - An End-of-Mission Report which provides a complete report of compliance, the final actual disposition of launched hardware, and, in the case of accidental impact, the probable location of impact and its region of uncertainty.
- Category IV missions:
  - A Planetary Protection Plan that details the planned approach to compliance with the implementation requirements (e.g., mission description, probability estimates, microbial burden estimates, contamination analysis plan, assay plan, microbial reduction plan).

- A Pre-Launch Planetary Protection Report that documents the degree to which all requirements have been met and that must include the values of the microbial burden at launch and the organics inventory.
- A Post-Launch Planetary Protection Report that updates the Pre-Launch Planetary Protection Report.
- An End-of-Mission Report that provides a complete report of compliance and the final disposition of all launched hardware.
- An inventory of bulk constituent organics that includes:
  - Parts lists, material lists, and other program documentation containing data relevant to organic material identification that are prepared by a flight project to specify and control the materials that are included in a vehicle destined for planetary landing.
- The locations of landings and impact points (determined and defined as accurately as mission constraints permit) of major components of space vehicles on the planet surface,
- Estimates of the condition of each landed spacecraft to assist in calculating the spread of organic materials.
- Category V missions. Missions categorized as "Unrestricted Earth return" have outbound phase requirements, only (*see above requirements*). Missions categorized as "Restricted Earth return" require:
  - A Planetary Protection Plan, including outbound phase requirements, if any, and an Earth Safety Analysis Plan.
  - A Pre-Launch Planetary Protection Report, including outbound phase requirements, if any, that must document the degree to which all Earth-return requirements to be attained prior to launch have been met.
  - A Post-Launch Planetary Protection Report, including outbound phase requirements, if any, to update the Pre-Launch Planetary Protection Report with respect to Earth-return requirements.
  - After sample collection, a report analogous to the outbound phase launch report: i.e., an Earth Pre-Launch Report.
  - An Earth Pre-Entry Report demonstrating readiness to enter the Earth's atmosphere in compliance with planetary protection requirements.
  - An End-of-Mission Report to address compliance with requirements for the protection of the Earth's biosphere and detailing the transfer of the samples to an appropriate containment facility.
  - A Sample Pre-Release Report to provide verification of sample analysis procedures subsequent to the End-of-Mission and demonstrating that any planned sample release will not harm the Earth's biosphere.

## 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.

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 responsibility of the mission operations manager and shall involve all participating elements and operating personnel (from project and support elements).

Information describing the NASA 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). Information on non-NASA networks can be found in project requirements, contracts and agreements.

## 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.

NASA-STD-7001, Payload Vibroacoustic Test Criteria recommends the use of the following acoustic Fill Factor:

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

- 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),
  - $H_{\text{gap}}$  is the average distance between the payload and the fairing, or cargo bay, wall, and
  - $\text{Vol}_{\text{ratio}}$  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 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.

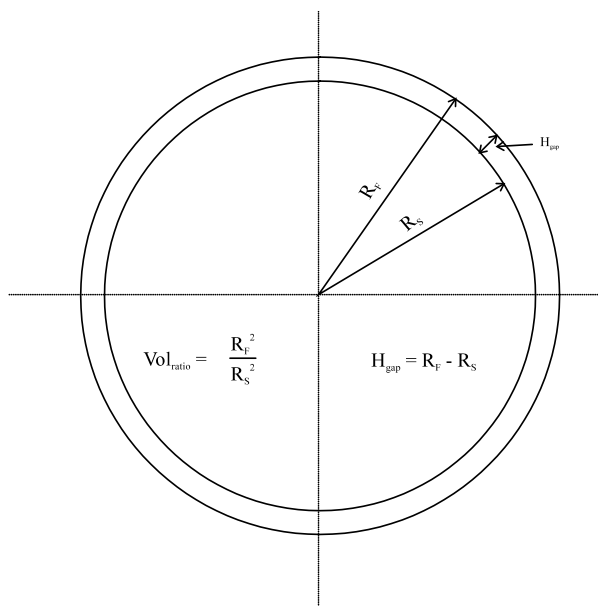


Figure A-1 - Cylindrical Payload in Fairing Acoustic Fill-Factor

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	2.00/44.4
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

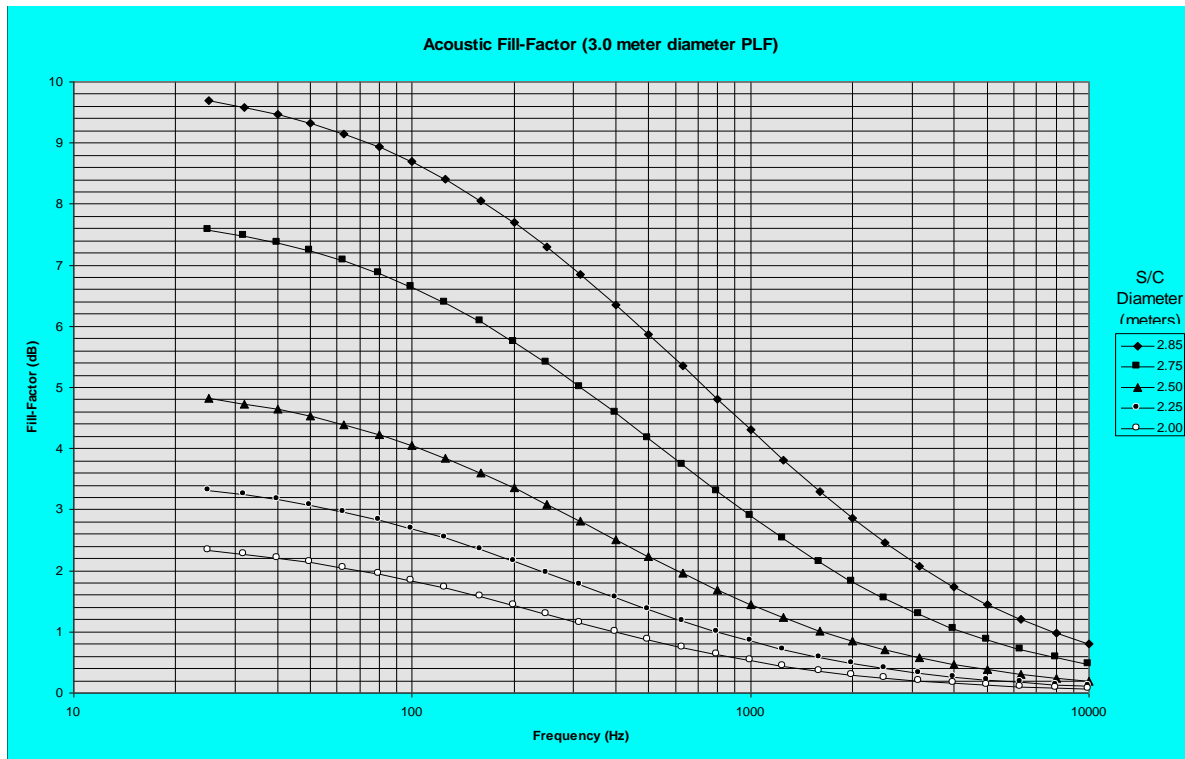


Figure A-2 - Acoustic Fill-Factor for various size Payloads in a 3 meter Diameter Payload Fairing

Additional methods for determining vibroacoustic loads can be found in Dynamic Environmental Criteria NASA Technical Handbook, NASA-HDBK-7005.

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., which 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.



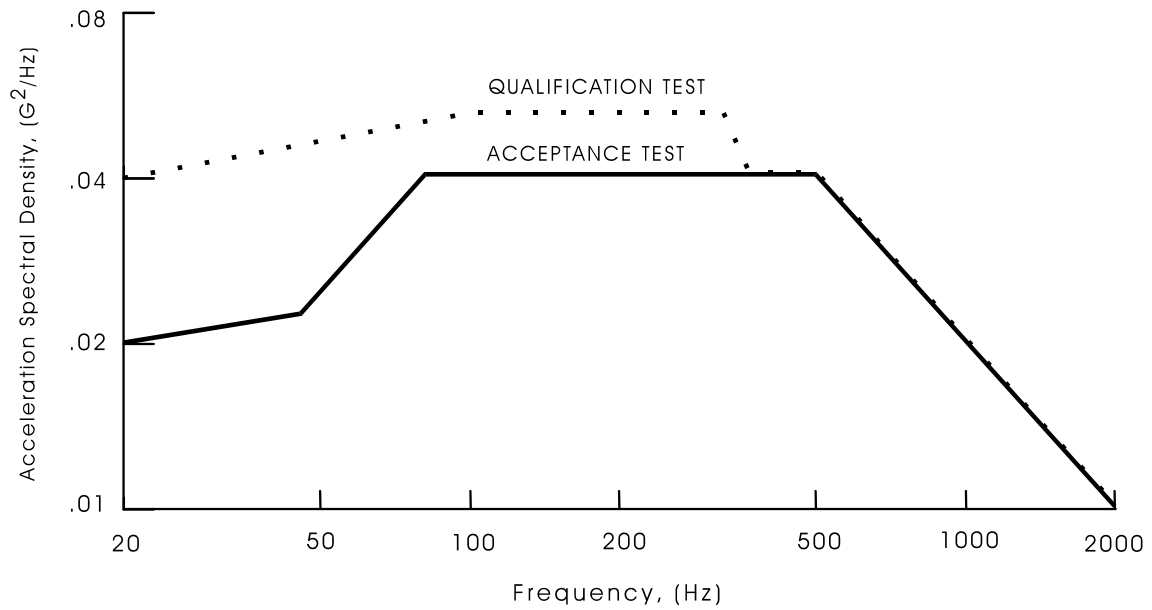
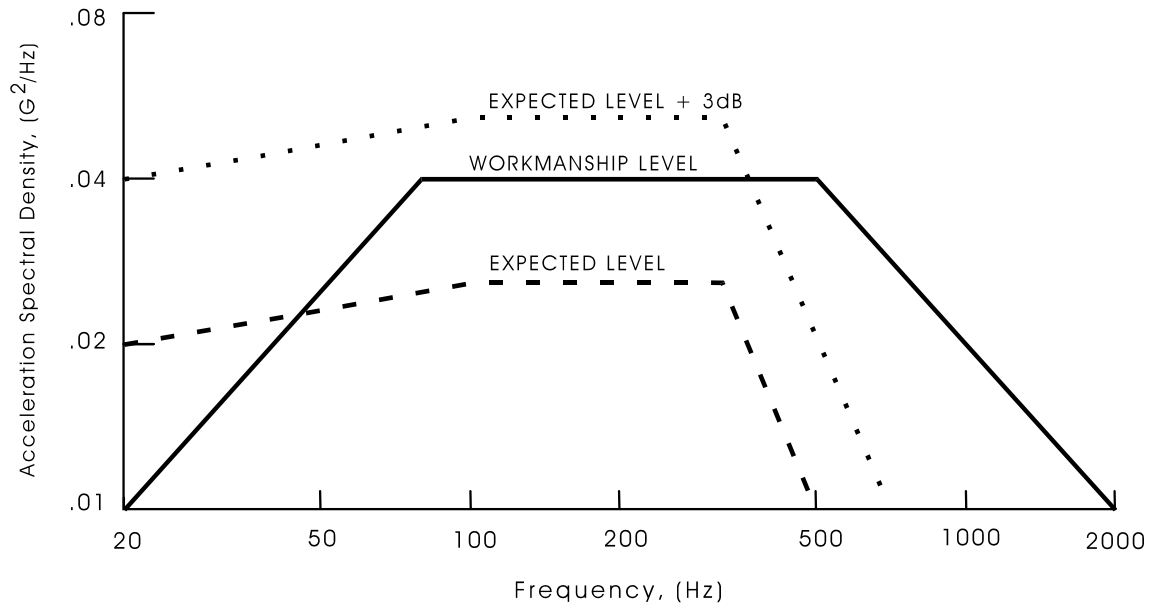


Figure A-3 Determination of Qualification and Acceptance Random Verification Test Levels

## Mechanical Shock

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 study was performed for GSFC and a report was generated in 1970 entitled Aerospace Systems Pyrotechnic Shock Data, NASA Contractor Report-116437, -116450, -116401, -116402, -116403, -116406, and -116019, Vol. I-VII. (Additional information and references can be found in Pyroshock Test Criteria NASA Technical Standard NASA-STD-7003). 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.

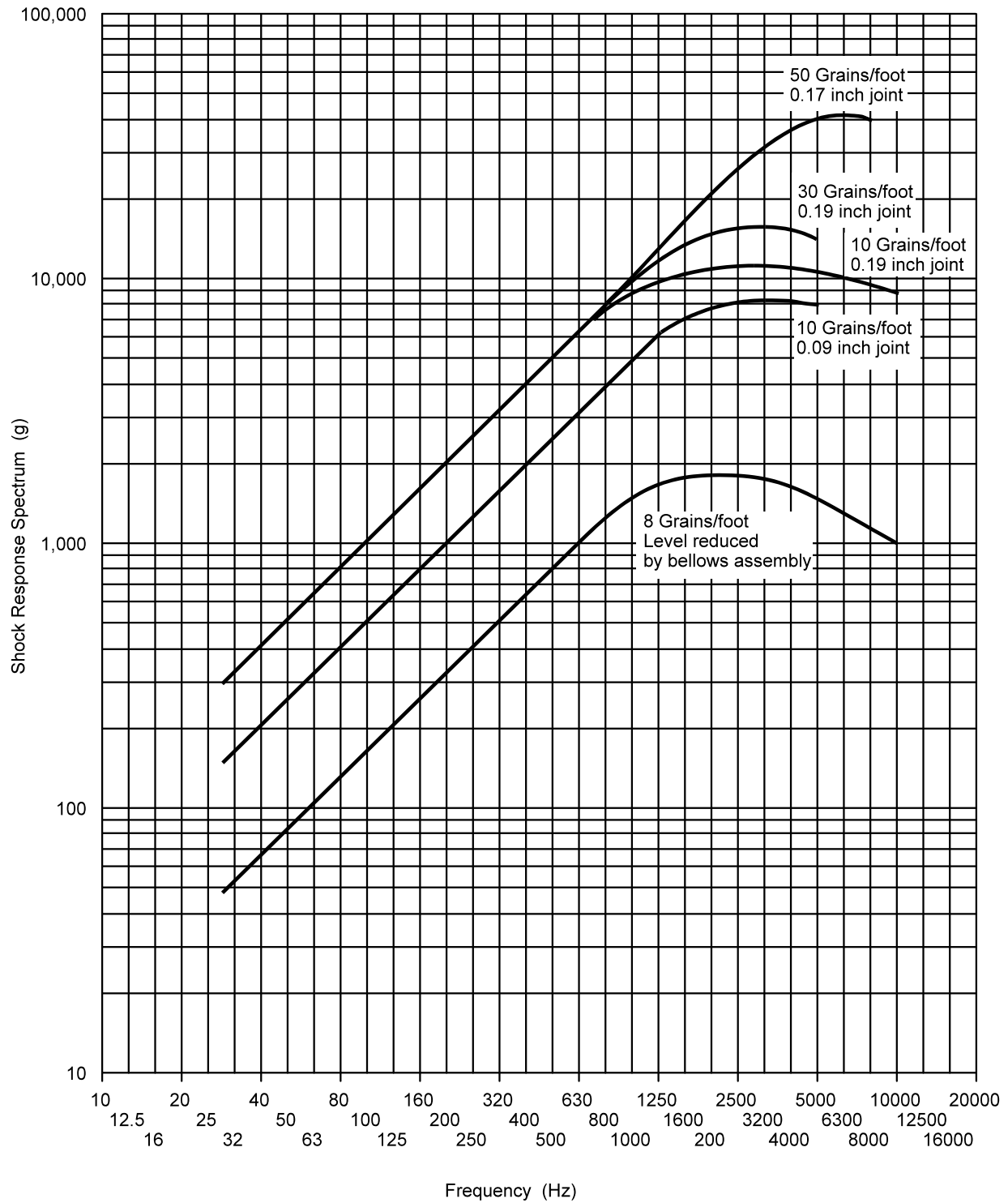


Figure A-4 - Shock Environment Produced by Linear Pyrotechnic Devices

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

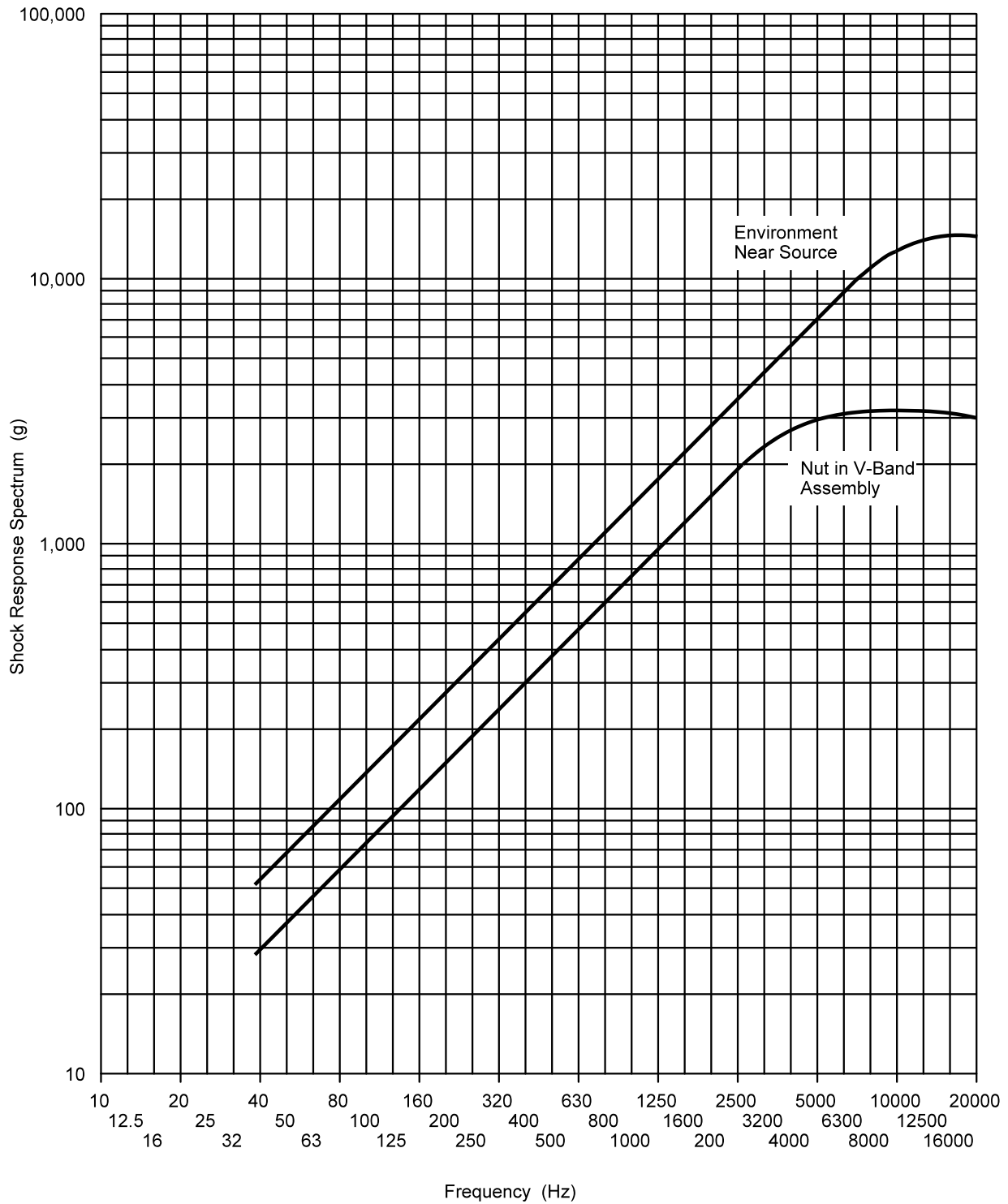


Figure A-5 - Shock Environment Produced by Separation Nuts and Explosive Bolts

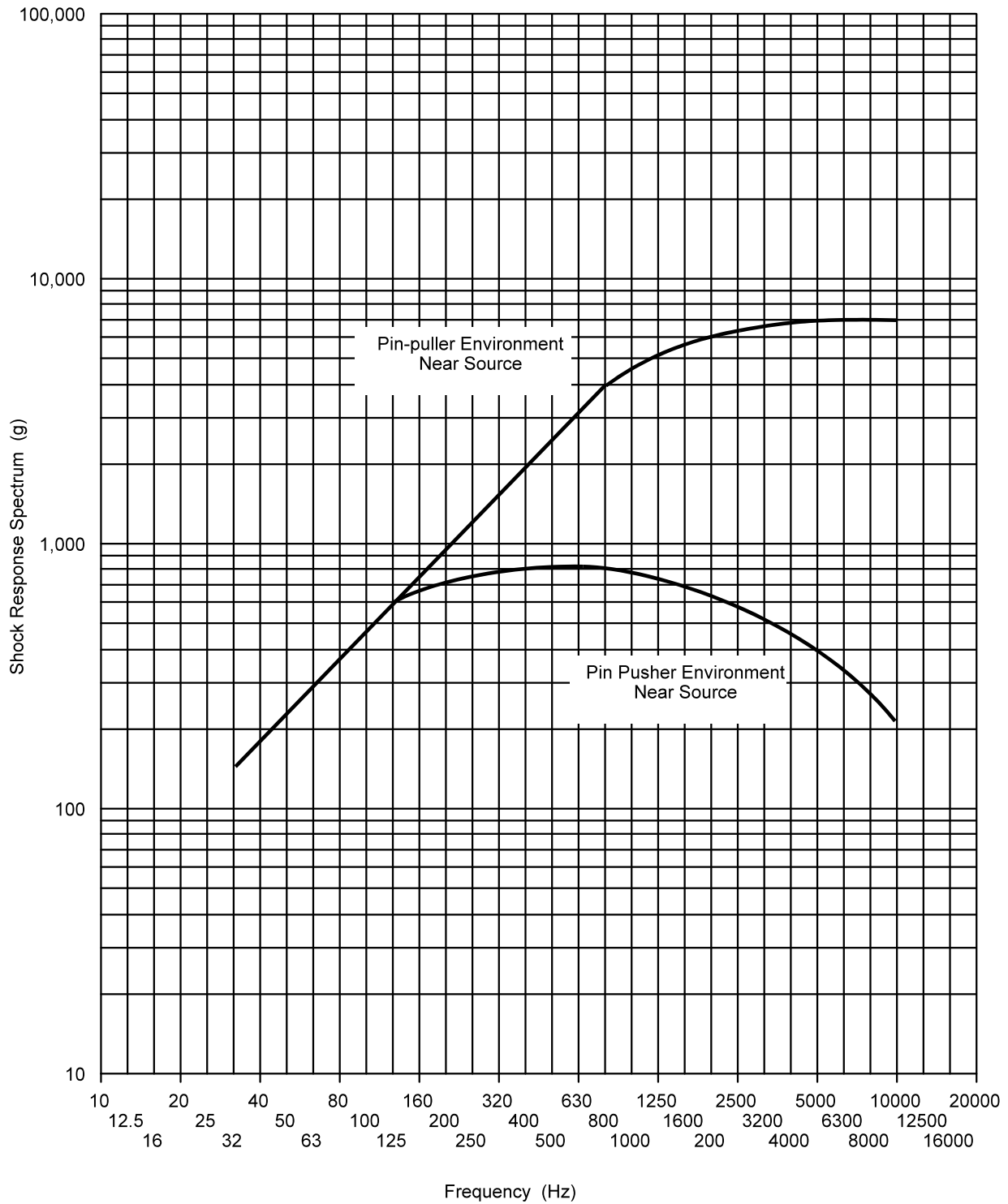


Figure A-6 - Shock Environment Produced by Pin-Pullers and Pin-Pushers

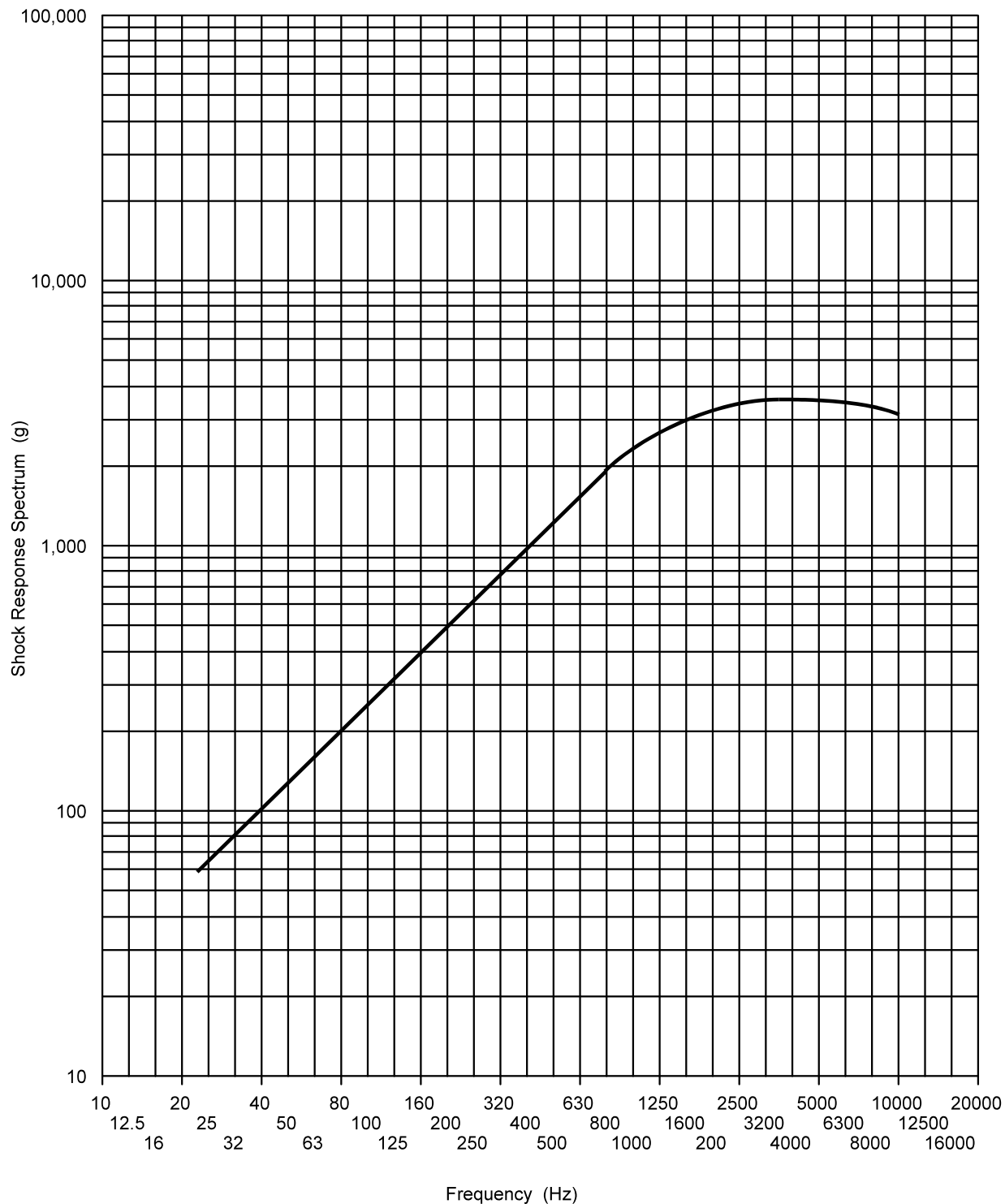


Figure A-7 - Shock Environment Produced by Bolt-Cutters, Pin-Cutters, and Cable-Cutters

Check the GSFC Technical Standards Program website at <http://standards.gsfc.nasa.gov> or contact the Executive Secretary for the GSFC Technical Standards Program to verify that this is the correct version prior to use.

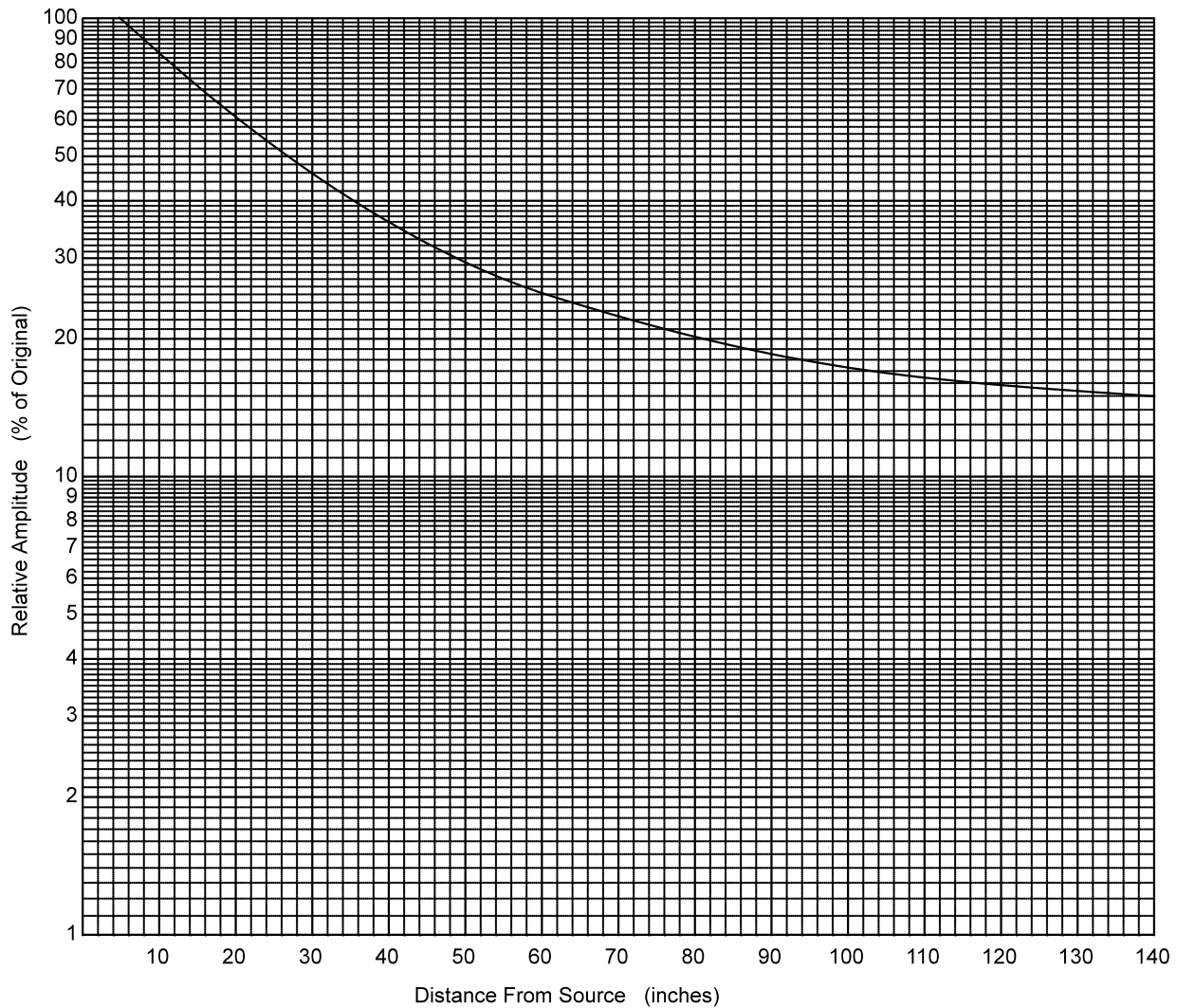
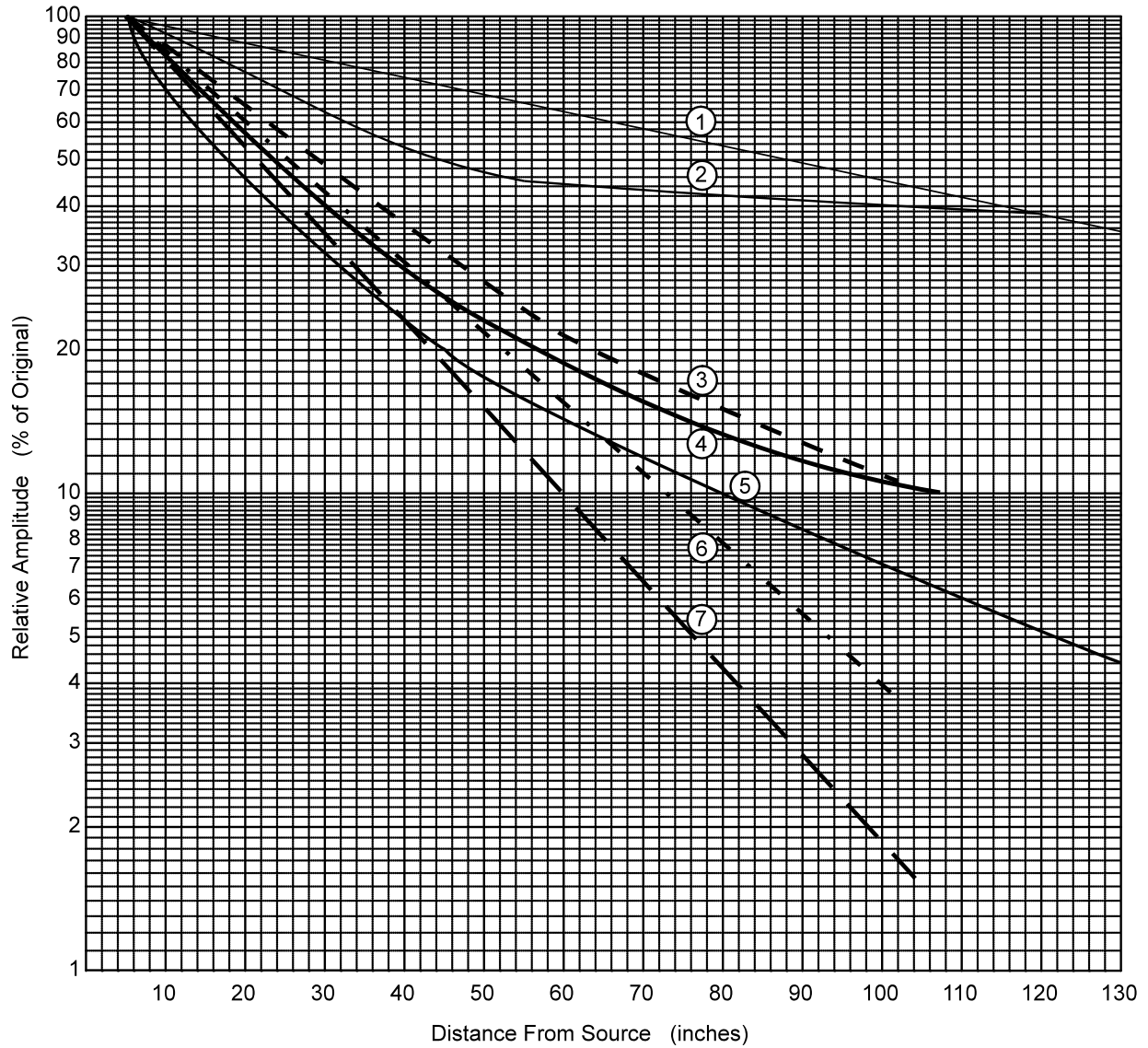


Figure A-8 - Attenuation of Constant Velocity Line





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

Figure A-9 - Peak Pyrotechnic Shock Response vs Distance

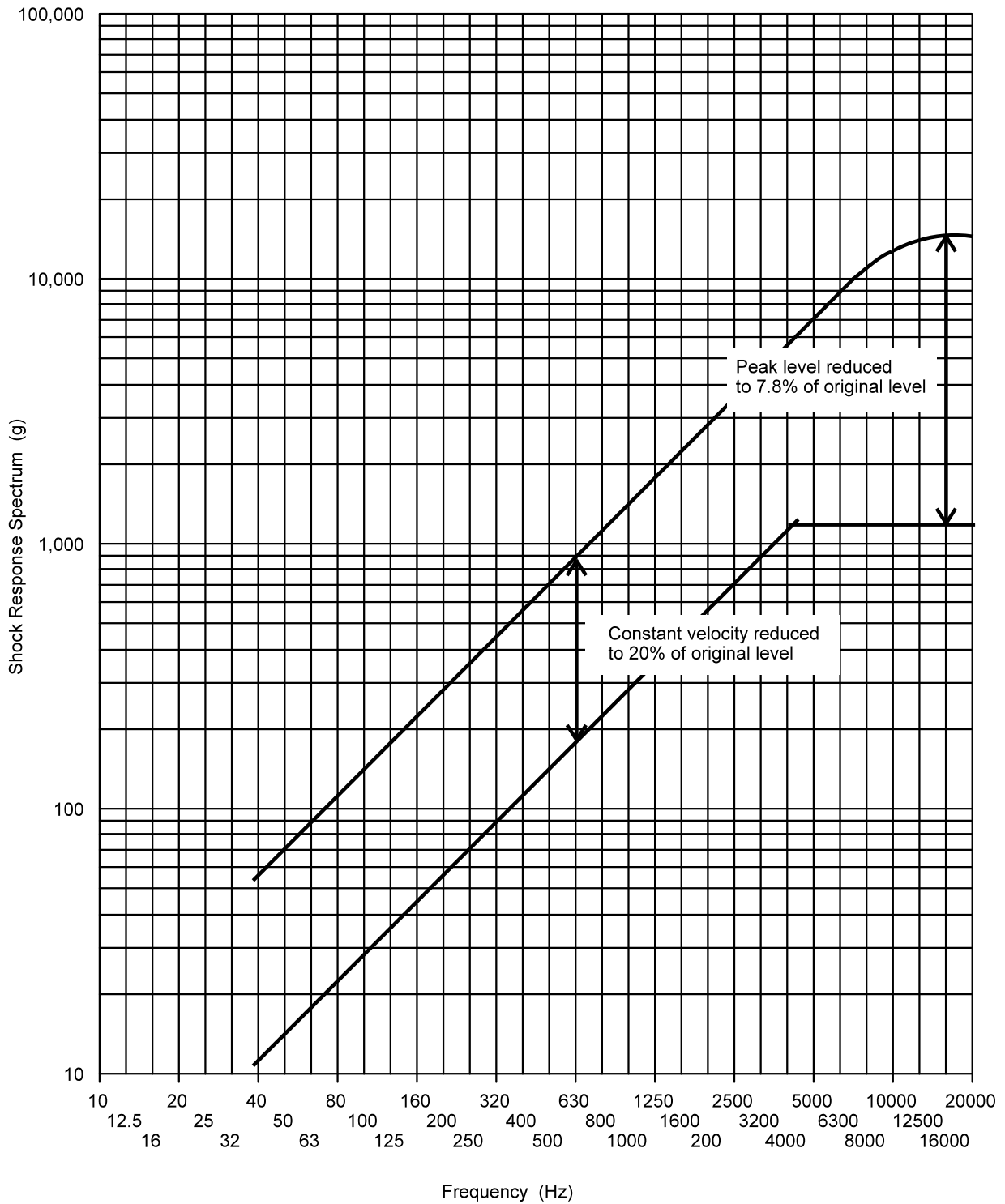


Figure A-10 - Shock Attenuation Example

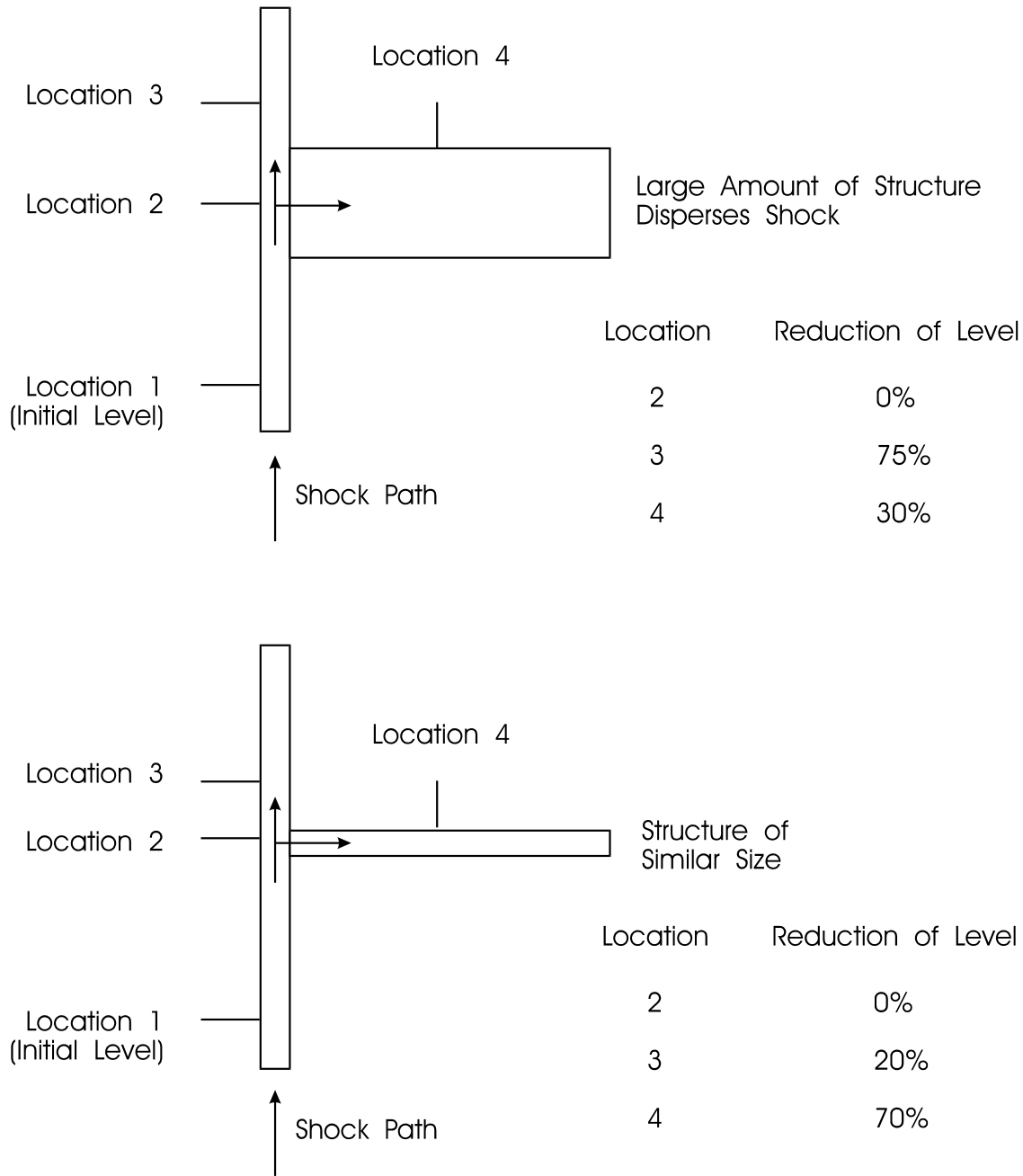


Figure A-11 - Reduction of Pyrotechnic Shock Response due to Intervening Structure