Ames Technical Standard

ARC-STD-8070.1
December 18, 2018

COMPLIANCE IS MANDATORY

Subject: Space Flight System Design and Environmental Test

Responsible Office: Code D/Office of the Chief Engineer, Ext. 4-1828, M/S 241-20

DOCUMENT CHANGE LOG

Status [Baseline /Revision /Cancelled]	Document Revision	Date of Change	Description
Baseline	0	12/18/2018	Initial

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Preface

P.1 PURPOSE

This document defines engineering design and environmental test requirements and guidelines for Class C and D space flight systems. This document also incorporates sound engineering practices and lessons learned, to ensure uniformity and consistency of the design and interfaces.

This document is not intended to be all-inclusive. Project specific functional and performance requirements are above and beyond the requirements established in this document.

P.2 APPLICABILITY

This document applies to Class C and D space flight systems, payloads, and technology demonstration projects managed in-house by Ames as well as spacecraft, or spacecraft components, procured by Ames. For spacecraft, and spacecraft components, procured from well-established aerospace contractors, the best practices established by those companies are likely acceptable. The acceptability of individual contractor best practices should be addressed during the tailoring process of the requirements of this document.

ISS payloads are only required to meet ISS requirements and should use this document for design guidance and best practices.

The applicability of this document may be waived for small efforts or for strategic reasons by agreement from ACE and the performing organization management.

This standard is a living document and is periodically assessed and updated to improve its clarity and effectiveness. While the engineering principles and practices are stable, the select set of requirements may evolve based on whether they continue to warrant increased visibility by their inclusion.

In this document, all document citations are assumed to be the latest version unless otherwise noted.

P.3 AUTHORITY

NPD 1280.1, NASA Management Systems Policy
NPR 7120.5, NASA Program and Project Management Processes and Requirements
NPR 8070.6, Technical Standards
APR 1120.2, Ames Engineering Technical Authority
APD 1280.1, Ames Quality Management System

P.4 APPLICABLE DOCUMENTS AND FORMS

Projects are expected to comply with applicable document requirements cited in this standard. However, verification product submission for those requirements are not required for compliance to this document.

NPR 7150.2	NASA Software Engineering Requirements
APR 7150.2	Ames Software Engineering Requirements
APR 8730.2	Ames Electrical, Electronic, and Electromechanical (EEE) Parts Control Requirements
NASA-STD-4005	Low Earth Orbit Spacecraft Charging Design Standard
NASA-STD-5001	Structural Design and Test factors of Safety for Spaceflight Hardware
NASA-STD-5006	General Fusion Welding Requirements for Aerospace Materials Used in Flight hardware
NASA-STD-5012	Strength and Life Assessment Requirements for Liquid Fueled Space Propulsion System Engines
NASA-STD-5017	Design and Development Requirements for Mechanisms
NASA-STD-5020	Requirements for Threaded Fastening in Systems in Spaceflight Hardware
NASA-STD-6016	Standard Materials and Processes Requirements for Spacecraft
NASA-STD-7001	Payload Vibroacoustic Test Criteria
NASA-STD-8739.1	Workmanship Standard for Polymeric Application on Electronic Assemblies
NASA-STD-8739.4	Workmanship Standard for Crimping, Interconnecting Cables, Harnesses, and Wiring
NASA-STD-8739.5	Workmanship Standard for Fiber Optic

	Terminations, Cable Assemblies, and Installation
NASA-STD-8739.6	Implementation Requirements for NASA Workmanship Standards
MSFC-STD-3029	Guidelines for Selection of Metallic Materials for Stress Corrosion Cracking Resistance
MMPDS-01	Metallic Materials Properties Development and Standardization
GSFC PPL-21	Preferred Parts List
EEE-INST-002	Instructions for EEE Parts Selection, Screening, Qualification, and Derating
MIL-STD-461G	Requirements for the Control of Electromagnetic Interference Characteristics of Subsystems and Equipment
MIL-STD-462	Measurement of Electromagnetic Interference Characteristics
MIL-HDBK-17	Plastics for Aerospace Vehicles
J-STD-001	Space Applications Electronic Hardware Addendum to IPC J-STD-001 Requirements for Soldered Electrical and Electronic Assemblies
(document has no number)	Aerospace Structural Metals Handbook
ANSI/AIAA S-080	Space Systems - Metallic Pressure Vessels, Pressurized Structures, and Pressure Components
ANSI/AIAA S-081	Space Systems - Composite Overwrapped Pressure Vessels (COPVs)
ANSI/ESD S20.20	Protection of Electrical and Electronic Parts, Assemblies and Equipment for the Development of an Electrostatic Discharge Control Program
IPC A-600	Acceptability of Printed Boards
IPC-A-610	Acceptability of Electronic Assemblies
IPC D-275	Design Standard for Rigid Printed Boards and Rigid Printed Board Assemblies
IPC-2221	Generic Standard on Printed Board Design
IPC-2222	Sectional Design Standard for Rigid Organic Printed

	Boards
IPC-2223	Sectional Design Standard for Flexible Printed Boards
IPC-6011	Generic Performance Specification for Printed Boards
IPC-6012	Qualification and Performance Specification for Rigid Printed Boards
IPC-6013	Qualification and Performance Specification for Flexible Printed Boards
IPC-6018	Microwave End Product Board Inspection and Test

P.5 MEASUREMENT/VERIFICATION

Compliance with this standard will be measured during the compliance matrix preparation and approval process, project milestone reviews and the CoFR assessment.

P.6 CANCELLATION

APR 8070.2, Class D Spacecraft Design and Environmental Test, June 15, 2017.

1 INTRODUCTION

1.1 Tailoring

This document represents a philosophy to incorporate large design margins and early testing to afford a streamlined development effort that minimizes the need to perform significant design optimization, and thus, provides a low-cost approach to small spacecraft missions. Experienced teams, mission specifics, and other consideration may warrant exceptions to these requirements. In particular, technology development projects may take exceptions in both design margin and extent of testing commensurate with stakeholder expectations for the demonstration in question.

Each project ***should*** review the requirements of this document and develop a tailoring approach. The tailoring approach ***should*** define which requirements are applicable as stated, which aren't applicable, and those for which an alternate approach is being requested. Projects ***should*** document rationale for all "shall" and "should" statements that are not intended to be met by the project. It is envisioned that this process occurs very early in the project development life-cycle in order to prepare a draft compliance matrix by System Requirements

Review (SRR) and a baselined compliance matrix, approved by the Ames Chief Engineer's Office by the Preliminary Design Review (PDR). The approved tailored requirements supersede this document and will be placed under project configuration control. Additional tailoring requests can occur if required by the project. A compliance matrix template and past project tailored matrices can be obtained from ACE.

1.2 Source Documents

This document was generated using the below sources. In nearly every instance, the requirement was incorporated unchanged from its original requirement. In cases where multiple sources levied a requirement in the same subject area, the more conservative instance was incorporated into the document.

a. Goddard Space Flight Center Rules for the Design, Development, Verification, and Operation of Flight Systems (Gold Rules)
b. Jet Propulsion Laboratory Design, Verification/Validation & Ops Principles for Flight Systems (Design Principles)
c. NASA endorsed technical standards
d. Military standards
e. Goddard Space Flight Center General Environmental Verification Standard (GEVS)
f. Ames Research Center engineering design and test best practices and lessons learned

Trace matrices have been developed between this document and their source requirements. These trace matrices are documented in Appendix C.

1.3 Precedence

None.

1.4 Responsibilities

Project Manager - The project manager is responsible for understanding the requirements and properly reflecting them in resource and schedule decisions. He, or she, is also responsible for specifying and communicating these requirements to government and contractor project personnel and flow down of requirements, as appropriate.

Lead Systems Engineer - The Lead Systems Engineer (LSE) is responsible for developing the tailoring approach to the requirements of this document and is also the owner of these requirements for the project. The LSE is responsible for ensuring the implementation and verification of the requirements. Verification compliance is verfified by ACE during the CoFR process. The LSE must concur to any requested deviations to the approved tailored requirements and document these decisions with rationale in waivers submitted in Ames PRACA system for approval by ACE. The LSE should also inform the implementing organization's management of the requested deviations.

IRB/SRB - The Independent Review Board (IRB) or Standing Review Board (SRB) for the project is responsible for reviewing the technical approach, and implementation to ensure compliance to the requirements contained herein.

Ames Chief Engineer's Office – The Ames Chief Engineer's Office (ACE) is responsible for reviewing and approving the proposed tailoring to the requirements of this document and to any subsequent waivers requested.

2 REFERENCE DOCUMENTS

(document has no number)	Spacecraft Thermal Control Handbook, Volume I, Fundamental Technologies
500-PG-8700.2.2	GSFC Electronics Design and Development Guidelines
500-PG-8700.2.7	GSFC Design of Space Flight Field Programmable Gate Arrays
AFSPC MAN91-710	Range Safety User Requirements Manual Volume 3 – Launch Vehicles, Payloads, and Ground Support Systems Requirements
AIAA S-111	Qualification & Quality Requirements for Space Solar Cells
AIAA S-112	Qualification & Quality Requirements for Space Solar Panels
AIAA S-122	Electrical Power Systems for Unmanned Spacecraft
ASTM E595	Standard Test Method for Total Mass Loss and Collected Volatile Condensable Materials from Outgassing in a Vacuum Environment
SAE AS50881	Wiring Aerospace Vehicle
DOD-A-83577	General Specification for Moving Mechanical Assemblies for Space Vehicles
DOD-E-83578	General Specification for Explosive Ordnances for Space Vehicles
DOD-W-83575	General Specification for Wiring Harness, Space Vehicle, Design and Testing
DOD-STD-1578	Nickel Cadmium Battery Usage Practices for Space Vehicles
GSFC-STD-1000	Rules for the Design, Development, Verification, and Operation of Flight Systems (Gold Rules)
GSFC-STD-7000	General Environment Verification Specification (GEVS)
JPL-D17868	Design, Verification/Validation & Ops Principles for Flight Systems (Design Principles)
JPL D-26086D	Environmental Requirements Document.
MIL-A-8625	Anodic Coatings for Aluminum and Aluminum Alloys

MIL-C-5541	Chemical Conversion Coatings on Aluminum and Aluminum Alloys
MIL-F-7179	General Specification for Finishes & Coatings for Protection of Aerospace Weapons System, Structures, & Parts
MIL-M-3171	Processes for Pretreatment and Prevention of Corrosion on Magnesium Alloy
MIL-STD-463	Electromagnetic Interference and Electromagnetic Compatibility Technology Definitions and Systems of Units
MIL-STD-883	Test Method Standard Microcircuits
MIL-STD-889	Dissimilar Metals
MIL-STD-975	NASA Standard Electrical, Electronic, and Electromechanical Parts List
MIL-STD-1568	Materials & Processes for Corrosion Prevention and Control in Aerospace Weapons Systems
MS33611	Tube Bend Radii
MAPTIS	Materials and Processes Technical Information Service Database **http://mpm.msfc.nasa.gov/materialdb.html**
NASA-HDBK-1002	Fault Management Handbook
NASA-HDBK-4006	Low Earth Orbit Spacecraft Charging Design Handbook
NASA-HDBK-4008	Programmable Logic Devices (PLD) Handbook
NASA-STD-4003	Electrical Bonding for NASA Launch Vehicles, Spacecraft, Payloads and Flight Equipment
NASA-STD-5002	Load Analyses of Spacecraft and Payload
NASA-STD-7002A	Payload Test Requirements
NASA-STD-7003	Pyroshock Test Criteria
QQ-N-290	Nickel Plating
QQ-C-320	Chromium Plating

3 FLIGHT SYSTEM DESIGN

Section 3 provides both requirements and design guidance. Requirements are provided in the form of ***shall*** statements and design guidance is in the form of ***should*** statements. If a mission classification warrants a different approach to a requirement, it is explicitly noted.

All ***shall*** statements are to be verified. Verification method and associated artifact and products must be captured in the compliance matrix. All ***should*** statements ("design guidance") do not require verification. However, at each milestone review the project team ***shall*** be able to explain how the design guidance was considered with rationale documented in the compliance matrix.

3.1 Design Margins and Fault Tolerance

3.1.1 Mechanical

3.1.1.1 Mass

The Spacecraft minimum dry mass margin ***shall*** be as specified in Table 3.1.1.1-1.

Table 3.1.1.1-1
Mass Margin

SRR	PDR	CDR	SIR
30%	20%	15%	5%

Margin = Allocation - Current Best Estimate (CBE)
% Margin = (Margin / Allocation) x 100

Note: *Dry Mass Allocation is defined relative to the launch vehicle payload allocation.*
Note: *Dry Mass CBE is the best estimate taking into account everything known, but exclusive of the growth that likely will occur based on maturity.*
Note: *Mass margins less than those indicated in Table 3.1.1.1-1 may be acceptable for systems using heritage hardware. However, the amount of reduced margin will depend on the maturity of the hardware in question and the risk of other components and subsystems impacting its physical and functional requirements.*

3.1.1.2 Deployment Systems

Mission critical deployment and separation systems (e.g., solar arrays and other spacecraft appendages, spacecraft-to-launch vehicle separations, etc.) ***shall*** demonstrate a functional force or torque margin of at least 50% for the entire range of motion or show a margin of 100% by analysis.

Note: *The margin applies under worst-case conditions, including restart from any position within the range of motion including incipient latching events.*

3.1.1.3 Actuator Design

Mission critical mechanisms and actuators (e.g., electromechanical motors and solenoids, phase-change and state-change actuators, and spring-energized devices, or mechanisms driven by these devices) ***shall*** demonstrate at least 50% torque/force margin for the entire range of motion or show a margin of 100% by analysis.

Note: *The margin applies under worst-case conditions at the end-of-life, including restart from any position within the range of motion. Mechanisms and actuators* ***should*** *be kept to a minimum.*

3.1.1.4 Stroke for Linear Actuators

Linear actuators implemented for mission critical mechanisms ***shall*** demonstrate at least 100% stroke margin above the stroke requirements of the mechanism.

3.1.1.5 Mechanism Cycle Life

A life test ***shall*** be conducted, within representative operational environments, to at least 2x expected life for all repetitive motion devices. The first 1x ***should*** be completed by CDR.

Mission critical mechanisms that function in a cyclic manner, and one-time deployment mechanisms, ***shall*** demonstrate a minimum life capability according to Table 3.1.1.5-1 by test.

The mission critical mechanism ***shall*** operate within specified performance at the end of the life test, and the test unit ***shall*** be disassembled and inspected for unacceptable wear or debris generation.

Table 3.1.1.5-1
Mechanisms cycle life design margins

Mission Critical Mechanism Element Type	Minimum Cycle Life Margin Requirement
Wet-lubricated low friction elements (e.g., rolling element bearings, involute gearing, etc.)	100%

Dry-lubricated elements, and wear-life limited elements (e.g., brush motors, slip rings, worm gearing, etc.)	200%
Wet-lubricated low friction elements, operating below the temperature rating of the lubricant	200%
One-time deployment mechanisms	300%

Note: *Vendor data may be used to demonstrate compliance with this requirement.*

Note: *A guideline for the life test is 50% of the cycle life test duration be conducted under nominal temperature conditions, 25% at the Protoflight cold temperature, and 25% at the Protoflight hot temperature.*

Note: *The margin applies under worst-case conditions at the end-of-life.*

3.1.1.6 Enclosed Volume

A vented area ***shall*** be designed to accommodate ascent venting per Vented Volume/Area < 2000 inches in accordance with accepted standards such as JPL D-26086, rev. D, Environmental Requirements Document.

Note: *The accommodation of a vented area eliminates the need for structural analysis due to pressure decay loads on structures like electronics box enclosures.*

3.1.2 Electrical

3.1.2.1 Power

The Spacecraft minimum power margin under worst case conditions ***shall*** be as specified in Table 3.1.2.1-1.

Table 3.1.2.1-1
Power Margin

SRR	PDR	CDR	SIR
30%	20%	15%	10%

Margin = Allocation - Current Best Estimate (CBE)
% Margin = (Margin / Allocation) x 100

Note: *Margins greater than those above may be indicated by project-specific* *circumstances. For example, highly complex mission and/or system design, development of low TRL technology, uncertainty of heritage designs, tight performance margins, low budget reserves, and tight schedule margins might be reason to require higher than the above margins.*

Note: *Margins less than those above may be acceptable in certain cases. For example, re-use of a known system design with a like payload in a mission application previously flown; other circumstances where the unknown factors are fewer and/or mature power/energy system is by project policy not to be changed; and where ample margins in other technical and programmatic resources may be reason for lesser power/energy margins than is indicated above.*

3.1.2.2 Depth of Discharge (DOD)

For cyclic operations, both during ground testing and mission operations, that use battery energy at intervals, the maximum DOD ***shall*** be in accordance with Table 3.1.2.2-1, applicable to both NiH2 and Li-Ion battery types. The DOD is based on CBE values (measured voltage, measured current draw, manufacturer characteristic curves, project test characterization curves, estimates of temperature exposure, etc.) An engineering assessment of battery health will be required if significant excursions from the requirement are encountered.

Table 3.1.2.2-1
DOD Limits for Cyclic Operations

Number of cycles	Allowable DOD
<100	<70%
100 < # cycles < 5,000	<60%
5,000 < # cycles < 30,000	<40%
> 30,000	<20%

Note: *A battery cycle is defined as any time the battery is discharged to 50% or more of the allowable DOD and recharged to near full.*

3.1.2.3 Power Distribution Circuit Margin

At PDR, there ***shall*** be a minimum margin of 30% on spare power switches and circuit count, including wiring, connector pins, and backplane insertion slots.

3.1.2.4 Flight Electronics Hardware Margins

Flight electronics hardware margins ***shall*** be in accordance with Table 3.1.2.4-1 at key life cycle development milestones.

Table 3.1.2.4-1

Flight Electronics Hardware Margins at Key Life Cycle Milestones

Resource	**Subsystem PDR**	**Subsystem CDR**	**Delivery**
PWB Area	30%	20%	10%
Connector pin-outs	30%	10%	5%
FPGA Resource Utiliztion	40%	20%	10%

3.1.2.5 Pyrotechnic Systems

3.1.2.5.1 Pyrotechnic Circuit Margin

At PDR, there ***shall*** be 30% margin on the spare pyro firing circuits (i.e., wires and connector pins) to accommodate late identified needs with minimum cost, schedule impact.

3.1.2.5.2 Pyrotechnic Circuit Fault Protection

Pyrotechnic circuits ***shall*** be protected from inrush currents and overvoltage conditions.

3.1.3 Thermal

3.1.3.1 System Level Temperature Margin

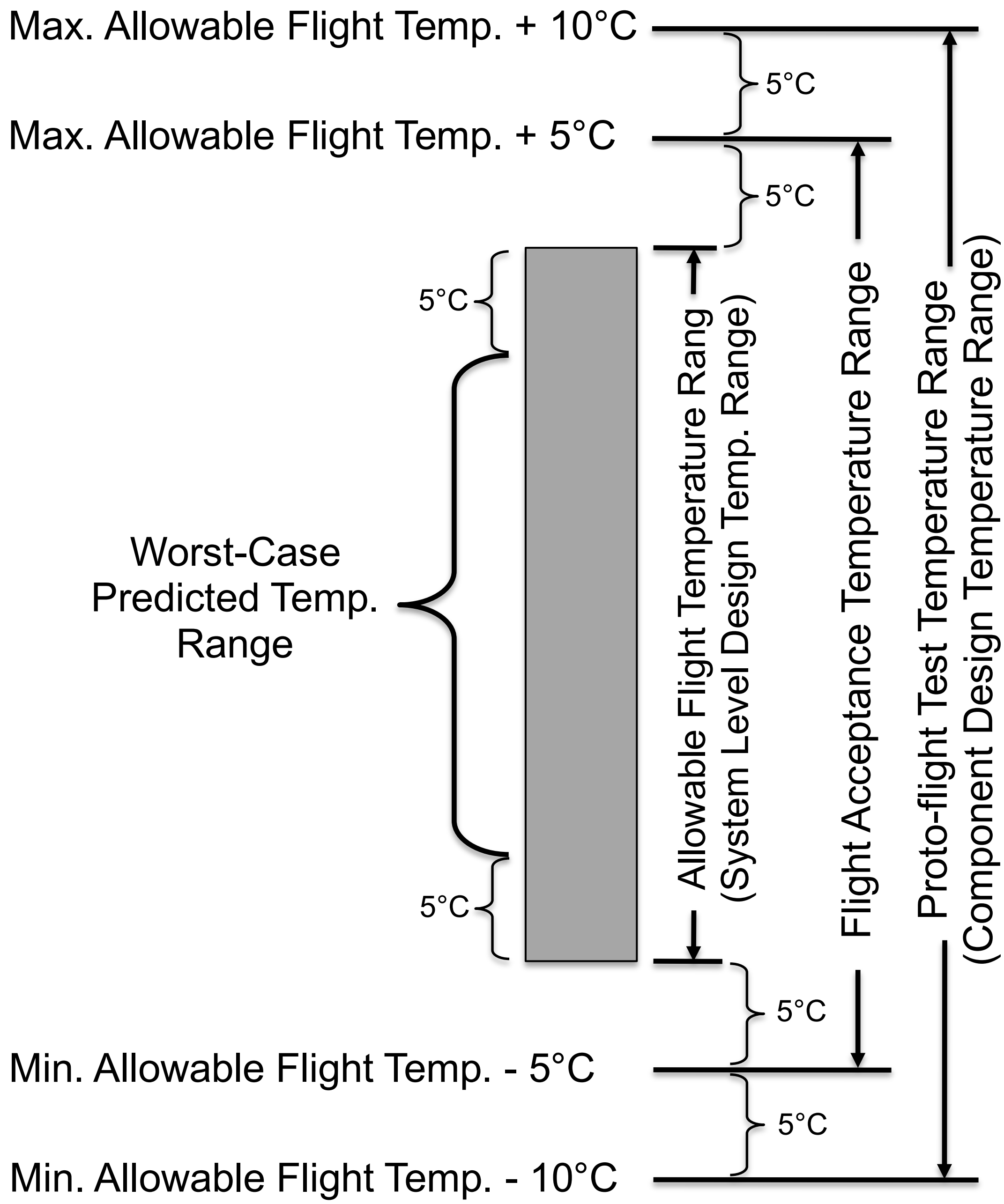
The system level thermal design ***shall*** provide a temperature margin of ≥5°C between the worst-case flight predicted temperatures (hot and cold for all mission phases) and the component allowable flight temperature.

Note: *Heater controlled components are exempted from this requirement for the cold side of the temperature range if the heater duty cycle does not exceed 70%.*

3.1.3.2 Component Level Temperature Margin

The proto-qualification test temperature range for spacecraft components ***shall*** be the allowable flight temperature range extended by ±10°C.

***Note**: For heater controlled components, the temperature margin may be reduced to 5°C on the cold side only.*



Temperature Margin
Figure 3.1.3-1

3.1.4 Propulsion

3.1.4.1 Liquid Propellant Design Criteria

Propellant fluid systems design ***shall*** conform with NASA-STD-5012.

3.1.4.2 Propellant Volume

The minimum propellant volume ***shall*** be based on the following criteria:

a. Worst case spacecraft mass properties
b. 3σ low launch vehicle performance
c. 3σ low propulsion subsystem performance (thruster performance/alignment, propellant residuals)
d. 3σ flight dynamics errors and constraints
e. Thruster failure (applies only to single-fault-tolerant systems)

3.1.4.3 Propellant Freezing

Hardware in contact with propellant or propellant vapor ***shall*** be thermally controlled to remain >10°C above the propellant freezing temperature except for cryogenic propellant applications.

3.1.4.4 Propellant Condensation

Hardware that will come in contact with propellant vapor ***shall*** be thermally controlled over the entire mission to remain >10 °C above the temperature at which propellant condensation will occur when such condensation presents a threat to the safe operation of the system.

Note: *Threats to safe system operation include condensation in pressure regulator sensing ports, lines which will be swept by high velocity pressurant, and condensation which could make a significant quantity of propellant unusable.*

3.1.4.5 Cryogenic Design Margin

The total energy load margin for passive coolers, mechanical coolers and stored cryogen systems designed to operate below -70°C ***shall*** be greater than 25%.

Note: *The cryogenic margin is to be used in conjunction with the realistic (non-anomalous) stacked worst-case thermal analysis of the cryogenic system enclosure and adjacent spacecraft hardware.*

Note: *Small additional detector heat loads in the range of a few milli-watts to tens of milli-watts can have large adverse thermal impacts on some cryogenic systems. The total load is comprised of the active and parasitic heat loads.*

3.1.4.6 Component Cycle Life

Propulsion components (e.g. chemical thrusters, catalyst beds, engine coatings, etc.) that function in a cyclic manner ***shall*** demonstrate a life capability with greater than 50% margin beyond the worst-case planned mission usage.

Note: *Based on the hardware heritage, prior mission use or qualification testing with a dedicated test unit.*

3.1.5 Attitude Determination and Control System (ADCS)

3.1.5.1 Controller Stability Margins

The Attitude Determination and Control Systems (ADCS) ***shall*** have stability margins as follows:

(1) Performance control modes: At least 6 db for rigid body stability with 30 degrees phase margin.
(2) Robust control modes: At least 10 db for rigid body stability with 60 degrees phase margin.

Note: *A robust mode might be a mission critical control mode, such as safe mode, or the nominal operating mode if a safe mode is not incorporate.*

3.1.5.2 Actuator Sizing Margins

The Attitude Control System (ACS) actuator sizing ***shall*** have at least 25% margin at the end of Phase C to allow for mass properties growth. Higher margins of up to 100% are recommended early in the design phase.

Note: *Knowledge of spacecraft mass and inertia can be very uncertain at early design stages. Actuator sizing **should** be done with the appropriate amount of margin to ensure a viable design.*

3.1.5.3 Flexible Body Systems

The magnitude of the flexible modes in the open-loop transfer function ***should*** be less than minus 12dB. Alternatively, there ***should*** be at least one order of magnitude between the controller bandwidth and the lowest resonance frequency.

Note: *Proper gain and phase margins are required to maintain stability for reasonable unforeseen changes and uncertainty in spacecraft configuration. It also allows for a reasonable level of unmodeled system properties.*

3.1.5.4 Passive Attitude Control System

A passive ACS such as permanent magnet, gravity gradient, and aero stabilized systems ***shall*** meet their performance requirements with a 30% margin. Performance requirements may include:

a. Pointing accuracy and/or stability
b. Peak or rms microgravity levels
c. Nutation angle limits

3.1.6 Telemetry & Command

3.1.6.1 Telemetry and Command Hardware Data Channels and RF Link

The telemetry and command hardware data channel and RF Link margin ***shall*** be as specified in Table 3.1.5.1-1.

Table 3.1.5.1-1
Telemetry and Command Hardware Channel and RF Link Margin

Resource	SRR	PDR	CDR	SIR
Telemetry & Command Hardware Channels	≥20%	≥15%	≥10%	0
RF Link	6 dB	6 dB	3 dB	3 dB

Margin = Allocation - Current Best Estimate (CBE)
% Margin = (Margin / Allocation) x 100

Note: *Telemetry and command hardware channels read data from hardware such as thermistors, heaters, switches, motors, etc.*

3.1.7 Flight Software and Computing System

3.1.7.1 Use of Analysis In lieu of Measurement

Before flight system computer design/procurement, analysis ***shall*** be employed to establish margins for critical performance resource parameters such as CPU speed, control cycle rates, interrupt rates and durations, communications bandwidth, and size of RAM, PROM, and EEPROM. Analysis results are documented as the Current Best Estimate (CBE).

3.1.7.2 Flight System Computing Resource Margin

The flight system computing resource design margin ***shall*** be as shown below for margin at critical development milestones to accommodate post-launch fixes, new capabilities, and to maintain adequate in-flight operating margins:

a. At computer selection 75% Margin
b. At PDR 60% Margin
c. At CDR 50% Margin
d. At launch 20% Margin

Note: *The flight system computing resource margin* ***should*** *consider constrained resources, e.g. computing capacity, memory, throughput, bus bandwidth, etc. If reused code is used, then lower margins are acceptable. The acceptability of specific margins* ***should*** *be addressed in the tailoring process.*

3.1.8 Safety

3.1.8.1 Catastrophic Hazards

For failures that may lead to a catastrophic hazard, the system ***shall*** have three independent, verifiable inhibits (dual fault tolerant).

Note: *For example, propellant leakage is a catastrophic hazard.*
Note: *Verification of independence of inhibits is necessary to preclude propagation of failure in safety inhibits than can result in critical or catastrophic threats to personnel or facility.*

3.1.8.2 Critical Hazards

For failures that may lead to a critical hazard, the system ***shall*** have two independent, verifiable inhibits (single fault tolerant).

Note: *For example, stored cryogen systems (and related GSE) are critical hazards for over-pressurization due to blockage or failure of a relief path.*

3.2 Mechanical

3.2.1 Structural and Mechanical

3.2.1.1 Loads

The spacecraft ***shall*** have positive margins for all ground, launch, and on-orbit loading conditions specified in Section 4.3.

3.2.1.2 Stiffness Requirements

3.2.1.2.1 Primary Structure

The first fundamental frequency of the primary structure ***shall*** be equal to, or greater than, the value specified by the launch vehicle provider. In the event that the spacecraft must be developed in the absence of an identified launch vehicle, then the first fundamental frequency ***shall*** be greater than 35 Hz when mounted to its interfacing structure.

Note: *The launch vehicle provider* ***should*** *be contacted early in the development process to determine the first fundamental frequency requirement. An assessment of the design against the requirement* ***should*** *be provided by PDR. If the spacecraft is flying as a secondary payload then additional structural design requirements may be levied by the primary payload.*

3.2.1.2.2 Secondary Structure & Components

The first fundamental resonant frequency of the secondary structure or a component weighing less than 23 kg ***shall*** be greater than 50 Hz when mounted to its interfacing structure.

Note: *If a coupled loads analysis of the launch vehicle and payloads, primary plus secondary, if applicable, results in first fundamental frequencies less than 50 Hz being acceptable to both the launch vehicle and payloads, then this lower value may be used.*

3.2.1.3 Factors of Safety

Structural design factors of safety ***shall*** be applied in accordance with NASA-STD-5001.

3.2.1.3.1 Joint slip factor of safety

The minimum factor of safety (yield or ultimate basis) for all spacecraft structures for non-detrimental instances of bolt slippage, bolt preload exceedance, contact, or encroachment ***shall*** be 1.00. The minimum factor of safety for detrimental instances (e.g. critical instrument alignment) ***shall*** be 1.25.

Note: *The required factor of safety is lower for joints that have room to slip in a benign way because the structure will not be strained, and the alignment isn't critical to performance. Verification: Analysis to demonstrate that friction joints will not slip.*

3.2.1.3.2 Fitting factor of safety

An additional factor of safety of 1.15 ***shall*** be applied for bolted joints, bonds, and fittings.

Note: *This additional fitting factor of safety can be reduced to 1.00 if a comprehensive development test program has been performed for the specific joint configuration, tested under representative worst-case loading and environmental conditions.*

3.2.1.3.3 Thermally-induced loading factor of safety

The spacecraft ***shall*** have positive margins of safety for yield and ultimate across the qualification/protoflight temperature range for all combined mechanical and thermal stresses.

3.2.1.4 Proof Testing Non-Metallic Structures

Primary and secondary structures fabricated from nonmetallic composites or containing bonded joints or bonded inserts ***shall*** be proof tested in accordance with NASA-STD-5001.

3.2.2 Fasteners & Pins

3.2.2.1 Torque Limits and Preload

All threaded fasteners ***shall*** be installed subject to torque limits and preload requirements of NASA-STD-5020.

3.2.2.2 Locking Requirements

All threaded fasteners for flight hardware ***shall*** utilize a locking method. The locking method ***should*** be in accordance with NASA-STD-5020.

3.2.2.3 Lubrication

All threaded fasteners ***shall*** be installed using appropriate lubrication in accordance with NASA-STD-5020.

3.2.2.4 Fastener Materials

All threaded fasteners and mating surfaces ***should*** prevent galling. Fastener material ***shall*** comply with NASA-STD-5020 and NASA-STD-6016.

3.2.2.5 Flat Head Screws

For sheet metal or similar applications, flat head screws ***should*** have 100° heads.

3.2.2.6 Use of Alignment Pins

Where pins are used for alignment of mating components, two sets of pins ***should*** be used to ensure accurate alignment. Where possible, double eccentric fasteners ***should*** be used in lieu of pins.

3.2.2.7 Use of Double Eccentric Fasteners

For transfer of shear loads between components, the use of double eccentric fasteners is preferred. Where shear pins are used to transfer loads, they ***should*** be made of hardened steel.

3.2.3 Mechanisms

The design of mechanisms including deployment, sensor, pointing, drive, despin, separation mechanisms, and other moving mechanical assemblies ***shall*** be in accordance with NASA-STD-5017.

DOD-A-83577, General Specification for Moving Mechanical Assemblies for Space Vehicles, ***should*** be used as a guide.

3.2.3.1 Spring Energized Bolt Release for Pyrotechnic Separation Nuts

Where pyrotechnically-actuated separation nuts are utilized in the flight system, a spring ***should*** be incorporated in the design to ensure positive retraction of the bolt from the separation interface. The spring stroke ***should*** meet or exceed the bolt stroke necessary for complete withdrawal of the bolt from the separation interface.

3.2.3.2 Explosive Ordnances

Explosive ordnance ***shall*** use DOD-E-83578 as a guide.

3.2.4 Materials

3.2.4.1 Environmental Effects on Material Selection

Thorough evaluation of the effect of mission flight parameters such as trajectory, orbit, planetary environment over the lifecycle of the mission ***shall*** be addressed for the impact on materials selection and design as applicable.

***Note**: Understanding the trajectory and space environmental effects (e.g., ESD, radiation, temperature, atomic oxygen, compatibility, orbital debris, etc.) on the spacecraft will eliminate costly redesign and fixes, as well as minimize the on-orbit failures due to environmental interaction with spacecraft materials.*

3.2.4.2 Metallic Materials

The composition and associated heat treatment of metallic components ***shall*** be in accordance with one of the following documents NASA-STD-6016, Standard Materials and Processes Requirements for Spacecraft, DOT/FAA/AR-MMPDS, or the Aerospace Structural Metals Handbook.

3.2.4.3 Composite Materials

The requirement for design and construction of composite components ***shall*** be in accordance with either Composite Materials Handbook (CMH) -17 or ANSI/AIAA S-081.

3.2.4.4 Galvanic Corrosion

To avoid the creation of galvanic corrosion couples, the use of dissimilar materials ***should*** be in accordance with MIL-STD-889.

3.2.4.5 Stress Corrosion Cracking

Materials susceptible to stress corrosion cracking (SCC), brittle fracture modes, liquid metal embrittlement, and hydrogen embrittlement ***should*** be avoided where possible. However, in the cases where these materials cannot be avoided they ***should*** be used in accordance with MIL-STD-1568.

3.2.5 Fluids Systems

3.2.5.1 Materials Compatibility

A materials compatibility assessment ***shall*** be performed for all hardware containing fluids by PDR for a preliminary assessment and CDR for a final assessment.

Components containing aqueous solutions ***should*** consider the use of Teflon lining or stainless steel. Nickel-plated aluminum ***shall*** not be used in place of stainless steel.

3.2.6 Mechanical Integration

3.2.6.1 Component Precision Location

When precise location of a component is required, the design ***shall*** use a stable, positive location system (not relying on friction) as the primary means of attachment.

3.2.6.2 Sensor and Antenna Blockage

When a spacecraft is in its stowed (launch) configuration, it ***shall*** not obscure visibility of any attitude sensors required for acquisition nor block any antennas required for command and telemetry.

3.3 Electrical

GSFC document, "Electronics Design and Development Guidelines", 500-PG-8700.2.2, ***should*** be used as a guide.

3.3.1 General

3.3.1.1 Minimum Operating Time

3.3.1.1.1 Prior to Spacecraft Integration

Electrical or electronic components or subsystems ***shall*** be operated for a minimum of 300 hours prior to integration with the spacecraft.

3.3.1.1.2 Integrated Spacecraft

The integrated spacecraft ***shall*** be operated for a minimum of 300 hours prior to launch with at least 50 hours in a thermal vacuum environment (10^{-5} Torr or less). The last 100 hours of operation ***shall*** be free of failures.

For Class C missions, higher operating/power-on time of ≥ 500 hours ***should*** be accumulated prior to launch with at least 100 hours in a thermal vacuum environment (10^{-5} Torr or less). At least, the last 200 hours of operation ***should*** be free of failures.

Note*: The last 24 hours is intended to cover final functional testing.*

3.3.1.2 Printed Wiring Boards

All new designed flight printed wiring/circuit boards (PWB/PCB) ***shall*** (design guidance for Class D) be coupon tested prior to population with flight components.

3.3.1.3 Electro-Static Discharge (ESD) Control

The project ***shall*** document and implement an ESD Control Program that is in compliance ESD Association Standard for the Development of an Electrostatic Discharge Control Program, ANSI/ESD S20.20, for protection of electrical and electronic parts, components, and equipment (excluding electrically initiated explosive devices).

Note: *The project **should** ensure that all ground support personnel utilize proper equipment and procedures for protection of spacecraft's electronics from damage due to electrostatic discharge (ESD).*

3.3.1.4 RF Component Immunity to Multipaction

Active high-energy RF components, such as radars, ***shall*** be designed and tested for immunity to multipaction before and after environmental testing.

Note: *Refer to NASA Public Lessons Learned Entry 770 for guidance regarding testing and evaluation for the presence of multipaction.*

3.3.1.5 Voltage/Temperature Margin Test

A Voltage/Temperature Margin Test (VTMT) ***should*** be performed on newly designed electronic parts, boards, or components, which includes combinations of worst-case environments and operational parameters prior to integration at the next level of assembly. The purpose of this test is to demonstrate that an adequate margin exists in the design to compensate for part tolerance, aging and radiation effects, as well as environment and input variation. During this test the parts, board, or component is subjected to variation of voltage at the high and low protoflight temperatures. Table 3.3.1.4-1 provides guidelines for selecting the test parameters. VTMT is not required if a Worse-Case Analysis (WCA) is performed. Existing designs must show evidence that a VTMT or a WCA is not required.

TABLE 3.3.1.4-1
VTMT TEST PARAMETERS

Assembly Type	Voltage Variation	Temperature Limits
Digital	±10% beyond the nominal supply voltage, NTE downstream circuit specifications	Qual/Protoflight Temperatures
Analog	±10% beyond the nominal supply voltage, NTE downstream circuit specifications	Qual/Protoflight Temperatures
Power Supply	±1 Volt beyond the spacecraft power bus variation	Qual/Protoflight Temperatures

3.3.1.6 Polarity Checks of Critical Components

All hardware ***shall*** be verified by test or inspection of the proper polarity, orientation, and position of all components (sensors, switches, actuators and mechanisms).

3.3.2 EEE Parts

3.3.2.1 EEE Parts Control

An Electrical, Electronic and Electromechanical (EEE) parts control (including derating, storage, and counterfeit prevention) plan (PCP) ***shall*** be developed in accordance with APR 8730.2, Ames Electrical, Electronic and Electromechanical (EEE) Parts Control Requirements, and be completed by PDR.

3.3.3 Digital Design

3.3.3.1 Synchronous Designs

Synchronous designs ***shall*** be used for digital logic to guarantee the sequence of logical decisions and the validity of data transfer. (Asynchronous design may be used if techniques are employed and demonstrated to provide guarantees for sequence verification and validation to the same confidence level as used for a synchronous design.)

3.3.3.2 ASIC/FPGA Synchronous Designs

The synchronous design of an Application Specific Integrated Circuits (ASIC) and Field Programmable Gate Arrays (FPGA) ***shall*** be verified, as a minimum by post-route timing analyses using a place and route tool and test vector simulation with timing checkers performed at the primitive level. Timing of boundary conditions (pin-outs) ***shall*** be constrained for place, route, and test vector simulation.

Note: *GSFC document "Design of Space Flight Field Programmable Gate Arrays", 500-PG-8700.2.7, and NASA Handbook 4008, "Programmable Logic Devices (PLD) Handbook",* ***should*** *be used as references.*

3.3.4 Use of Plastic Encapsulated Microcircuits (PEMs)

The use of PEMs is permitted provided each use is thoroughly evaluated for thermal, mechanical, and radiation implications of the specific application and found to meet mission requirements. PEMs should be selected for their functional advantage and availability, not for cost saving; the steps necessary to ensure reliability usually negate any initial apparent cost advantage. A PEM

shall not be substituted for a form, fit and functional equivalent, high reliability, hermetic device available within project's constraints or resources.

Screening of PEMs is essential before PEMs are inserted into most flight hardware. Burn-in at the part level is to be employed to addresses infant mortality. If burn-in at the part level is not practical, board level burn-in or board/box level Environmental Stress Screening may be substituted. The use of PEMs is time sensitive; a PEM ***should*** be no more than three years old from date of manufacture to date of application.

Boards containing PEMs ***should*** be cleaned and dried using solvents and baking methods that will not impact the reliability of parts or boards. The terminations of PEMs ***should*** be pretinned using tin–lead solder to reduce the risk of tin whisker growth or to remove gold plating. PEMs typically have pure tin-plated terminations, which are a risk for tin whisker growth and subsequent system failure due to shorting or plasma arcs. Alternatively, PEMs may be available with gold plated terminations, which are at risk for failure due to gold embrittlement. After installation and cleaning, the application of conformal coating to the devices is recommended to minimize re-absorption of moisture and to further reduce the risk of tin whisker growth.

Note: *It is important to bake out Plastic Encapsulated Microcircuits prior to storage and prior to use in order to drive out absorbed moisture from the plastic molding material.*

3.3.5 Power Systems Design

3.3.5.1 Wiring Design

The electrical wiring harnesses between components ***shall*** (design guidance for Class D) be in accordance with EEE-INST-002. SAE AS50881 ***should*** be used as a guide.

Note: *Commercial wiring* ***should*** *be acceptable as long as assembled harnesses are baked out and hipot tested for space environment.*

3.3.5.2 Power Quality

Power quality **should** be per AIAA S-122 as a guide.

3.3.5.3 Solar Array

Solar arrays ***shall*** be designed to meet the system average and peak power load required during daylight and eclipse operations. Design considerations **should** include orbit altitude, inclination, design lifetime, the type of solar cells and power output, the beginning-of-life (BOL) power production capability, and end-of-life (EOL) power production capability.

Solar arrays ***should*** be designed and qualified per AIAA S-111 and AIAA S-112 as guides.

3.3.5.4 Batteries

Batteries ***shall*** be designed to meet the system power requirements during both nominal and contingency operations.

Design considerations ***shall*** (design guidance for Class D) include the required voltage, current loading, duty cycles, activation time, storage time, mission length, primary or secondary power storage, orbital parameters, power use profile, temperature and radiation environment, and battery charge/discharge cycle limits.

For systems using solar arrays, battery capacity requirements ***shall*** (design guidance for Class D) account for periods from launch and array deployment up to nominal operational conditions, including appropriate margins for ground and/or flight anomalies.

Note: *Contact the launch provider (or vehicle platform in case of a payload) regarding design and/or certification requirements for use of Lithium-Ion batteries.*

Note: *AIAA S-122* ***should*** *be used as a guide.*

3.3.6 Power Converters and Supply

3.3.6.1 Power Converter Synchronization

When required by EMC considerations, subsystem DC-DC power converters ***shall*** be synchronized via an externally supplied sync frequency (preferred) or operated in a free-running mode subject to the following conditions:

a. If operating in a free-running mode, operating frequencies are spread.
b. Testing and/or analysis of operating frequencies/harmonics show that no interference will be generated.

3.3.6.2 Power Cycle Capability

The electronics design ***shall*** include the capability to cycle the power on and off either through on-board hardware or software, or a commanded link.

Note: *The design **should** consider the effect of power transients on the electronics. A time-delay between power resets **should** be considered to minimize the effect on electronics.*

3.3.6.3 Power Supply Transient Analysis

A power supply transient analysis ***shall*** be performed to identify all transient conditions presented to the spacecraft power bus, including turn-on/off transients, in-rush current, component state changes, etc. The supplier ***shall*** provide the transient analysis for review by the buyer upon request.

3.3.6.4 Power System Charging

The electrical power system charging design ***shall*** be in accordance with NASA-STD-4005, Low-Earth Orbit Spacecraft Charging Design Standard, when voltages beyond ±55 volts are present.

Note: *NASA-HDBK-4006 **should** be used as design guidance.*

Note: *Surge protection and power clamping should be considered in the design for the protection of power sources.*

3.3.7 Circuit Protection

Circuit protection ***shall*** be provided for the spacecraft system, subsystems, and components to adequately protect mission critical functions.

The use of fuses in mission-critical applications ***should*** be minimized.

Note: *There is no recovery from an in-flight blown fuse, thus their use in mission-critical applications must be carefully considered. The design **should** consider the use of, or lack of use of, circuit protection for mission critical hardware when tripped circuit breakers would result in the loss of space vehicle control.*

3.3.8 Grounding, Bonding, and Isolation

3.3.8.1 Local Single Point Ground

A Local Single point ground is defined such that all voltages are measured with respect to a particular point in the local ground network, not just to a general undefined “ground”. Satellite grounding is composed of multiple subsystem grounds which have unique naming, grounding needs, and characteristics.

A satellite common point/star point/master point ground is defined as a particular point on the spacecraft chassis where all local single point grounds join. There is not a one size fits all solution for utilizing a local single point ground.

Local Single Point grounds are not always recommended if the design includes mixed analog and digital ground planes. Analog and Digital Ground Planes attached to a single point ground need to be sufficient in size with appropriate mixed ground isolation circuitry as well to achieve accurate circuit references/grounds. Interface wiring can also provide ground paths between subsystems prior to joining it to a design selected local single point ground. This is to preserve intended digital logic switching and precise analog circuit function. Proper grounding sizing and configuration can prevent ground loops and ground bounce which if not addressed in a design may lead to a system malfunction.

A metallic structure utilizes a common point ground chassis location. Note: Carbon Composite Aluminum Honeycomb structure and carbon composite structures have slightly different satellite common point grounding designs that ***should*** be considered by the electrical designer.

Grounding for satellite systems ***shall*** also take into account spacecraft charging effects. Local single point ground for spacecraft charging ***should*** maintain local subsystem grounds within less than 1 ohm with respect to neighboring local ground.

Each subsystem ground tree for unmanned systems ***shall*** have a local single point ground to spacecraft chassis via the shortest practical wire length unless a different approach is required by integrated systems design. In RF or high-speed digital systems, a multi-point ground approach may be necessary to meet EMI/EMC and performance requirements. It is good practice and the design engineer ***shall*** (design guidance for Class D) provide design rationale and applicable waiver to deviate from the requirement.

Ground Isolation between Digital and Analog circuits ***should*** be considered and may drive the design to isolating grounds by way of inductive isolation that does not use a local common point ground design.

Thermal considerations ***should*** also be considered in the designing ground paths. Size of spacecraft ***shall*** also be considered.

NanoSat's of 3 to 6 U volume requirements are less affected by resistance build up due to minimal path length and close proximity of grounds. Ground path measurements ***should*** be minimized and less than 1 ohm.

A larger spacecraft (≥ 75 Kg) that has ground path lengths of meters to several meters ***shall*** meet ground path measurements in the 2.5 milliohm to 0.1 ohm range to prevent digital logic switching issues or ground loops.

Component grounds ***should*** be routed to the local single point circuit board ground/ ground plane on daughter cards of spacecraft avionics. Each circuit board ***should*** route the local ground to a common point backplane board which routes to a common point ground on the chassis if design principles allow.

When applicable, separate avionics boxes ***should*** deliver their single point common ground connections to a single spacecraft chassis common point ground also known as a star ground. Common point ground connections ***should*** have no more than 0.1 ohm resistance, verified by a 4-wire resistance measurement, between board ground path references and chassis common point ground.

3.3.8.2 Primary Circuit Return Path

Wires ***shall*** (design guidance for Class D) be used for the primary circuit return path. Structure or shields ***shall*** (design guidance for Class D) not be used.

3.3.8.3 Bonding

Bonding requirements have increased importance when the overall physical size of a system increases. Bond resistance can become additive when several subsystems are in series circuit and need to use a common ground or reference.

Bonding of electrical circuits/equipment ***should*** have a bond requirement of 2.5 milliohms or less verified by a 4-wire resistance measurement. Acceptable ranges could be from less than 2.5 milliohms up to 1 ohm depending on the application. Torque measurements on bonds associated fasteners ***shall*** be recorded. Soldered connection ***shall*** be inspected to NASA soldering workmanship standard (J-STD-001, Space Addendum). A cognizant electrical/electronics engineer can determine if a bond requirement can deviate from the spec on a case by case basis.

Communications Equipment and Antenna ***shall*** have a bond requirement to chassis of 2.5 milliohms or less verified by 4 wire resistance measurement.

High Impedance sensor circuits ***shall*** use a bonding requirement of 2.5 milliohms or less verified by a 4-wire resistance measurement.

Note: *NASA-STD-4003, Electrical Bonding for NASA Launch Vehicles, Spacecraft, Payloads and Flight Equipment,* ***should*** *be used as a guide.*

3.3.8.4 Electrical Isolation

All component interface circuits ***shall*** be electrically isolated to not less than 1 mega-ohm DC between power lead and chassis ground and between power lead and signal ground.

For systems less than 100 volts, the isolation ***shall*** be not less than 0.5 mega-ohms.

3.3.9 Connectors

3.3.9.1 Inadvertent Mating

Keyed or unique connectors ***should*** be used to prevent inadvertent connector mating.

If keyed connectors are not used then the design ***shall*** preclude inadvertent mating of the wrong connectors.

3.3.9.2 Powered Connections

Unit and harness connector gender ***shall*** be selected such that powered pins are always pretected. Typically this translates to selection of sockets for connectors delivering powered outputs, but some connector series (e.g., MIL-DTL-83513 Micro-D) are polarized in an opposite fashion.

When a connector contains both inputs and outputs, protection priority should be give to the applicable harness. Test and integration procedures ***shall*** make provisions to prevent mating or de-mating of connectors at signal or powered interfaces unless power has been turned off.

3.3.9.3 Blind Mate Connection

Provisions ***shall*** be provided to allow verification of proper connection (e.g. alignment and depth of engagement) of blind mate connectors. However, the use of blind mate connectors ***should*** be avoided due to difficulties in verifying proper connection.

3.3.9.4 Connector Pin Population

Connectors ***shall*** be fully populated with pins for structural rigidity (to avoid bending of pins during mating and de-mating). Populated but unused pins ***shall*** be electrically terminated.

3.3.10 Radiation Tolerance

Note: *Verification by analysis is acceptable for the radiation tolerance requirements of Section 3.3.10.*

3.3.10.1 Single Event Effects (SEE)

SEE is any measureable effect in a circuit caused by a single incident particle. It can be either non-destructive such as single event upset (SEU) and single event transients (SETs) or destructive such as single event latchup (SEL) and single event gate rupture (SEGR).

The spacecraft electronic devices ***shall*** be chosen such that the subsystem operates within performance specification during and after exposure to the high-energy radiation environments defined for the mission with an RDF of 1. The high-energy radiation environments are unique for each mission and ***should*** be determined using NASA qualified models.

The Radiation Design Factor (RDF) is defined as:

$$\text{RDF} = \frac{\text{Radiation-resisting capability of a part or component in a given application}}{\text{Radiation environment present at the location of the part or component}}$$

The component, subsystem, and spacecraft system level requirements regarding performance with respect to SEE during operation are as follows:

a. Temporary loss of function or loss of data ***shall*** be permitted provided that the loss does not compromise subsystem/system health, full performance can be recovered rapidly as defined based on mission needs and constraints, and there is no time in the mission that the loss is mission critical.
b. Normal operation and function ***shall*** be restored via internal correction methods without external intervention in the event of an SEU.
c. Fault indication ***shall*** be provided in the telemetry stream for anomaly traceability involving SEE's.

3.3.10.2 Total Ionizing Dose (TID)

Total dose effects in electronic and photonic parts are cumulative, long term degration due to ionizing or non-ionizing radiation. For TID, the main concern is the effects in insulating regions of metal-oxide semiconductors (MOS) and bipolar devices. Ionizing radiation can also cause leakage currents in MOS devices.

3.3.10.2.1 General Shielding

The spacecraft's electronic components ***shall*** be chosen such that the component operates within performance specification during and after the Total Ionizing Dose (TID) exposure at an RDF of 2 times the TID level present at the location of the device.

Note*: The TID is unique for each mission and needs to be determined using NASA qualified models.*

3.3.10.2.2 Spot Shielding

Where spot shielding is to be applied, an RDF of 3 ***shall*** be required.

Note: *The greater RDF with use of spot shielding is to account for uncertainties in part capabilities, space environment, and transport modeling. General shielding* ***should*** *be considered in the evaluation of shielding needs for an RDF of 3. Spot shielding alone does not need to account for an RDF of 3.*

3.3.11 Pyrotechnic Functions

3.3.11.1 Enabling of Pyrotechnic Functions

The spacecraft design ***shall*** provide an enable function for each pyro event. Pyro functions can be enabled in groups when mission success is not dependent on the firing order or sequence of the pyro functions within each enabled group.

Note*: Separate and independently commanded 'enable' and 'fire' functions, implemented in series fashion, provide protection against single failures, e.g. in the 'fire' circuitry, inadvertently energizing a pyrotechnic. When the order in which the pyros are fired makes no difference to mission success, then they may be simultaneously enabled as a group- because failure of a 'fire' circuit has no adverse impact. When the order of pyro events is essential for mission success, then these events must be enabled separately from each other- to be tolerant of potential single failures in the 'fire' circuits. Single failure of a 'fire' or 'enable' function to activate the pyrotechnic is overcome by using redundant A- and B-side pyro circuits.*

Note*: The timing by when pyro functions are enabled should take into consideration the possibility that with a failed 'fire' circuit, a pyro event could occur when the 'enable' is given.*

Note*: Unintended activation of pyrotechnics is prevented by a series of four independent functions: Inhibit, Arm, Enable, and Fire. The Inhibit function is used in ground operations including at the launch pad. The Inhibit is typically removed by launch vehicle separation or other action at launch. The Arm function is* *activated at the launch pad. The Enable function is activated shortly before actual activation of the pyro. The Enable often affects groups of pyro functions. The Fire function is the last required action that sends electrical current to activate a specific function or NSI.*

3.4 Thermal

3.4.1 General Design Approach

3.4.1.1 Design Tailored to Specific Application

Thermal control design ***should*** be tailored to the specific applications of the mission, with consideration for both equipment reliability and temperature/ performance interactions. Nominal and worst-case temperature ranges for mission critical hardware ***shall*** be determined and the impact to the thermal design from credible failure modes assessed.

3.4.1.2 Passive vs. Active Thermal Control Systems

When possible, passive thermal control systems ***should*** be utilized. Active thermal control systems ***should*** be used only when necessary.

3.4.1.3 Motor and Actuator Self-Heating

Motor or actuator operating flight acceptance, protoflight, and qualification temperature ranges ***shall*** be defined by the interface or environmental temperature extremes, prior to energizing a motor or actuator. The maximum allowable temperature limit of the motor or actuator due to self-heating ***shall*** be defined separately.

Note: *This requirement is to ensure consistency between the definition of operating and non-operating Allowable Flight Temperature limits, while accommodating the characteristic of motor or actuator self-heating during ground testing.*

3.4.1.4 Thermal Coatings Properties

All thermal analysis ***shall*** employ thermal coatings properties validated to be accurate for materials and mission flight parameters over the lifecycle of the mission. The beginning of life (BOL) and the end of life (EOL) properties ***should*** be utilized where applicable in a manner that adds conservatism to the analysis.

3.4.2 Use of Heaters

The heater design ***should*** accommodate the survival and operational temperature ranges of the hardware.

3.4.2.1 Power Density Limit for Film Heaters

Kapton film heaters, when bonded to a metallic or composite substrate over 100% of the heater's active area, ***should*** be limited to a maximum power density of 3.0 watts per square centimeter

Kapton film heaters ***should*** be limited to a maximum power density of 0.5 watts per square centimeter when bonded over voids or other discontinuities of the heat-dissipating substrate; this situation ***should*** be avoided if possible.

3.4.2.2 Maximum Duty Cycle of Heaters

The maximum duty cycle ***shall*** be 70% under worst-case cold environmental conditions when heaters are implemented in a thermal control design.

Note: *The purpose of this requirement is to ensure that positive heater control authority exists during the design phase of the flight system thermal control. Heater duty cycling can fall outside of the prescribed design range during the mission, but* ***should*** *generally meet the above maximum duty cycling criteria during ground testing to validate the thermal control design.*

Note: *This principle applies to systems above -70 degrees C. Active controllers include mechanical thermostats and PID or pulse wave modulated controllers. To conserve power consumption throughout the mission due to typically large spacecraft bus voltage ranges, the maximum 70% duty cycle applies to the condition of a minimum nominal voltage, and not a minimum failed voltage, e.g. failed battery cell or solar cell string.*

3.4.3 Thermal Environments

3.4.3.1 Thermal Environment for Earth Orbiting Missions

3.4.3.1.1 Direct Solar Radiation

The direct solar radiation values to be used for the design of earth orbiting spacecraft ***shall*** be as defined in Table 3.4.3.1.1-1.

Table 3.4.3.1.1-1
Solar Radiation

Case	Direct Solar Flux (W/m^2)
Cold	1322
Median	1367

Hot	1414

3.4.3.1.2 Earth Albedo

Albedo is highly variable across the globe and is dependent on orbit inclination, orbit beta angle, the distribution of reflective properties of the surface and the amount and type of cloud cover. Earth Albedo values captured in Spacecraft Thermal Control Handbook and other NASA near Earth thermal environment guidelines are based on statistical analysis of the data collected from the Earth Radiation Budget Experiment (ERBE) hosted on two NOAA satellites. For recommended values for Earth IR and Albedo, refer to “Spacecraft Thermal Control Handbook, Volume I, Fundamental Technologies”, Environments of Earth Orbit section.

3.4.3.1.3 Earth Infrared Radiation

The earth infrared radiation value (global annual average) to be used for the design of earth orbiting spacecraft ***shall*** be 234 ±7 W/m^2.

3.4.3.2 Thermal Environment for Interplanetary Missions

For interplanetary missions, spacecraft’s thermal environment varies based on its distance from the Sun during interplanetary cruise, during planetary flybys, and during planetary orbit, based on its altitude and latitude from the planet.
Solar flux as a function of distance from the sun in AU can be calculated from the following equation:

$$\frac{1367.5}{AU^2}\frac{W}{m^2}$$

For Lunar and planetary IR and Albedo recommended values, refer to “Spacecraft Thermal Control Handbook, Volume I, Fundamental Technologies”, Environments of Interplanetary Missions section.

3.5 Propulsion

3.5.1 Design and Analysis Requirements

3.5.1.1 Fuses for Propulsion System

Flight fuses for wetted propulsion system components ***shall*** be selected such that overheating of propellant will not occur at the maximum current limit rating of the flight fuse.

3.5.1.2 Plume Impingement Analysis

Thruster or external venting plume impingement ***shall*** be analyzed and demonstrated to meet mission requirements.

3.5.2 Sizing

3.5.2.1 Propellant Tanks

Propellant tank volume ***shall*** be sized to accommodate the propellant required to perform the mission per paragraph 3.1.4.1, plus the propellant required for spacecraft disposal and ullage.

Note: *The design* ***should*** *meet the special safety requirements for hydrazine tanks as documented in AFSPC MAN91-710, Range Safety User Requirements Manual Volume 3 – Launch Vehicles, Payloads, and Ground Support Systems Requirements. The specific requirements that must be met* ***should*** *be addressed with the Range Safety Office for the launch site.*

3.5.2.2 Propellant Quantity

Propellant load estimates ***shall*** be based on specification minimum value Isp for engine/thruster and allocated spacecraft system mass.

3.5.3 Pressurized Components

3.5.3.1 General Design and Test Requirements

Propulsion elements and other pressurized components ***shall*** meet the design and test requirements of ANSI/AIAA S-080 “Space Systems – Metallic Pressure Vessels, Pressurized Structures and Pressure Components” and ANSI/AIAA S-081, “Space Systems – Composite Overwrapped Pressure Vessels (COPVs)”, except as noted in this section (refer to the latest published revision year of each document).

***Note:** Currently, the minimum yield factors of safety required for the design of metallic pressure vessels are equal to the minimum Proof Test Factor. While the design and production of metallic pressure vessels is a mature technology, programmatic risk may exist if the pressure vessel is a new design, or if the vendor is inexperienced. It is strongly encouraged to incorporate higher than the minimum design factors of safety for non-heritage metallic pressure vessel design to ensure that all units pass the Proof Test per ANSI/AIAA S-80.*

A leak detection test ***shall*** be performed for components such as seals, pressure vessels, leak-proof valves, and hermetically sealed units, or which have hermetically sealed parts attached. For purposes of this test, a hermetically sealed unit is defined as a sealed unit which contains a gaseous atmosphere, as opposed to a potted unit. Leak test ***should*** be performed at the end of environmental testing as shown in Figure 4.2.2-1. Units ***should*** be pressurized with a gas containing more than 10 percent helium. The maximum allowable leak rate ***should*** be 1×10^{-6} standard cubic centimeters per second of helium, when tested in a chamber whose maximum pressure is 3×10^{-2} Torr.

3.5.3.2 Tubing

Tubing ***should*** be stainless steel or titanium, where practicable. Tubing bend radii ***should*** be in accordance with MS33611, Tube Bend Radii.

3.5.3.3 Joints

Tubing joints ***shall*** be thermal welded and NDE verified.

3.5.3.4 Access for Cleaning and Testing

Tubing design ***shall*** incorporate provisions for cleaning and to allow proof testing.

3.5.3.5 Separable Fittings

Separable fittings ***shall*** have redundant sealing surfaces, such as double “O” rings, be “parallel loaded” type, and include a locking provision. “Parallel loaded” means that the fitting contains a compressed element that exerts outward pressure on the other elements of the fitting such that both seals are maintained even if relaxation occurs. Separable fittings ***should*** be accessible for leak tests and for torque checks. Separable fittings ***should*** not be designed or assembled with lubricants or fluids that could cause contamination or could mask leakage of a poor assembly.

3.5.3.6 Pressure Surge Prevention (Liquid Systems)

The propulsion system design and operations ***shall*** preclude damage due to pressure surges ("water hammer").

3.5.4 Propulsion Safety

3.5.4.1 Fiber-Reinforced Composite Over-Wrapped Pressure Vessels

The minimum design ultimate factor of safety for fiber-reinforced composite over-wrapped pressure vessels (COPV) ***shall*** be 1.75.

Note: *The value of 1.75 comes from multiplying the burst test SF of 1.50 by the ratio of the JPL design/test philosophy, i.e. 1.5 times design SF of 1.4, divided by the test SF of 1.2.*

Note: *The yield factor of safety is not applicable. Long-term stress rupture failure criteria may indicate higher factors of safety.*

Note: *Safe-life fracture analysis and testing **should** be per ANSI/AIAA S-081.*

3.5.4.2 Use of Passive Isolation in Bi-Propellant Systems

Bi-propellant propulsion systems ***shall*** incorporate a passive means of ensuring that liquid fuel and oxidizer are prevented from mixing in the pressurization system or tanks.

3.5.4.3 Use of Gas Regulators

Gas regulators (single or series redundant) ***shall*** not be used to provide isolation of pressurant from the propellant tank. Isolation devices such as latch valves or pyrotechnically actuated valves ***shall*** be incorporated for long periods of quiescent operation.

Note: *Experience shows that gas regulators can leak, sometimes at rates far in excess of the device specification.*

3.5.4.4 Propulsion Ignition

Propulsion system design ***shall*** preclude ignition of propellants in the feed system.

3.5.4.5 Residual Test Fluids

Propulsion system design and the assembly and test plans ***shall*** preclude entrapment of test fluids that are reactive with wetted material or propellant.

3.5.4.6 Propulsion System Safety Electrical Disconnect

An electrical disconnect "plug" or set of restrictive commands ***shall*** be provided to preclude inadvertent operation of components.

3.6 Attitude Determination and Control System (ADCS) Design

Note: *The effects of spacecraft-generated and external magnetic fields need to be considered in the design of the attitude control system including impacts on attitude control system energy management in counteracting magnetic torques.*

3.6.1 Control Authority

ADCS design ***shall*** provide a ratio of disturbance torque to control torque of less than 40%.

Note: *Proper control authority is required to maintain stability for reasonable unforeseen changes and uncertainty in spacecraft configuration and/or disturbance environment. It also allows for a reasonable level of unmodeled system properties.*

3.6.2 Sampled Control System Timing

Sampling rates used in control systems ***shall*** be chosen larger than the control bandwidth, and with considerations for system delays.

A general rule of thumb for sampling frequencies expressed as follows:

Sampling Frequency > {[10 + 20*(number of full sample delays)]*control bandwidth}

Note: *This ensures an appropriate level of robustness and stability margins.*

3.7 Telemetry and Command

3.7.1 Mission Critical Telemetry and Commands

The spacecraft ***shall*** be designed to maintain continuous telemetry coverage during all mission-critical events. Mission-critical events include separation from the launch vehicle; power-up of major components or subsystems; deployment of mechanisms and/or mission-critical appendages; and all planned propulsive maneuvers required to establish mission orbit and/or achieve safe attitude.

If for some reason the mission-critical event has to occur out-of-view or during a period when no downlink is feasible, then the spacecraft data ***shall*** be stored onboard and downlinked at the first available communication opportunity.

3.7.2 Spacecraft State Management

The spacecraft ***shall*** be designed to operate in a finite set of defined states..

The preliminary definition of spacecraft states ***shall*** be defined by SRR and the final state definition ***shall*** be by PDR.

3.7.2.1 Explicit Commanding of States

Commands that are intended to place the spacecraft in a specific known state, ***shall*** explicitly specify the target state.

Note: *This requirement precludes the use of toggle commands, which have a target state implicitly defined by the current state.*

Note: *Elements to consider when establishing state include inertial, temporal, device capability or configuration, file allocation tables, and boot code in RAM.*

3.7.2.2 Command Logging

The flight software generated data products available for downlink ***shall*** include a log of all received, executed and rejected commands that indicates:

a. time of receipt
b. time and nature of disposition (execution or rejection)

3.7.2.3 Critical Command Locking

Critical commands that may adversely affect system operation or safety ***shall*** be controlled so that they cannot be inadvertently executed without ground mission operation acknowledgement.

***Note:** Multiple independent actions help to reduce the possibility that hazardous or mission critical actions will occur in error or be started prematurely.*

3.7.2.4 Power-On Reset (POR) State

At flight system power turn on or recovery from a power under-voltage condition, each subsystem ***shall*** autonomously configure to an unambiguous, safe, system compatible state.

3.7.2.5 Power-On Reset (POR) State Visibility

Any POR occurrence ***shall*** be unambiguously identifiable via telemetry.

***Note:** The design **should** also consider what state the spacecraft **should** be set to after the power on reset.*

3.7.2.6 Visibility of Spacecraft State

The telemetry subsystem end-to-end design ***shall*** permit ground mission operations team to determine the state of the spacecraft, particularly to determine if the spacecraft executed a fault-protection response.

***Note:** The flight system **should** track both intended commanded states as well as actual system state and assess/report discrepancies for system health assessment and fault management.*

3.7.2.7 Visibility of Health Status

The telemetry subsystem ***shall*** be designed to provide telemetry data so that the spacecraft health can be assessed under normal, stressed, and faulted operations. This includes health, anomaly determination, and visibility into spacecraft state and any mission unique functions.

3.7.2.8 Visibility for Anomaly Determination and Reconstruction

The telemetry subsystem design ***shall*** provide telemetry data and sampling frequency for engineering data, including any special diagnostics, to enable the mission operations team to perform anomaly determination, investigation/reconstruction, particularly for mission critical activities.

3.7.2.9 Visibility of Mission-Unique Functions

Special consideration ***should*** be given to providing increased telemetry instrumentation for mission-unique or other sensitive functions.

3.8 Software and Information Systems

3.8.1 NASA Software Engineering Requirements

The project ***shall*** comply with the requirements of APR 7150.2 as determined by the classification of the software described in those requirements. Specifically, software design documentation contents ***shall*** comply with the documentation requirements of APR 7150.2.

3.8.2 Flight Software Design

The requirements of this section are intended for newly developed code. The acceptability of commercial code, or reused code, will be evaluated on a project specific and application specific basis.

3.8.2.1 Software Modifiability

The flight software design ***shall*** (design guidance for Class D) provide the capability to upload new software and replace old modules during the mission.

3.8.2.2 Protection from Unintended Software Modification

Flight software that is modifiable during flight ***shall*** be protected from unintended modifications including those caused by operations errors, single event effects, and hardware problems.

Note: *Protection is typically provided by intentionally enabling a write operation before modifying the software; at all other times, write operations are disabled to protect the software from unintended modifications. Unintended modifications can be introduced through configuration management, design, and operation flaws as well as physics.*

3.8.2.3 Compatibility with COTS Tools

Where commercial hardware, operating systems, or other tools are used to support testing, flight software design ***should*** accommodate such platforms with minor change, such that these tests are relevant to software V&V.

Note: *This is to support unti testing and early integration testing, and to lessen dependency on high fidelity hardware-in-the-loop testbeds.*

3.8.2.4 Start-Up Response

Flight software ***shall*** be designed to initialize itself and any associated hardware, including any back-up hardware or software system, to a safe and known state upon startup.

3.8.2.5 Software Design Robustness

3.8.2.5.1 Command Validation and Acknowledgement

a. Flight software ***shall*** be designed to verify uplinked commands, data, or loads.
b. Flight software ***shall*** reject, and log incorrectly formatted commands, data, or loads and provide notification that they were incorrectly formatted.
c. Flight software ***shall*** be designed to send acknowledgement of command receipt to the source with indication of acceptance or rejection of command. For rejected commands, the acknowledgement message ***shall*** include a reason for rejection in the transmitted message.

Note: *For example, flight computer designs have included Error Detection and Correction (EDAC) logic on EEPROMs, and the load process has been designed to detect and respond to failure if the EDAC detects an uncorrectable bit error. Software designs have included check sum logic and periodic verification of memory to detect command, data, or load, and memory faults.*

Note: *For example, a command handler* ***should*** *check whether a received command is appropriate for the current system mode, and a software module* ***should*** *check whether a command is appropriate for its local state.*

3.8.2.5.2 Detection and Response to Radiation Events

Flight software ***shall*** be designed to detect and respond to memory faults allocated to the software, such as stuck bits or single event upsets (SEU) if architecture is not protected in hardware (e.g., Triple Modular Redundancy (TMR)).

3.8.2.5.3 Predictable Behavior When Stressed

Software algorithms and their implementation ***should*** be designed to behave predictably when stressed beyond their performance limitations. Some examples include:

a. Being sensitive to identified uncertainties.

b. Precluding an undesired response to mathematical singularities or limitations.

c. Responding predictably to possible events that exceed capabilities.

3.8.2.5.4 Response to Resource Over-Subscription

The software design ***should*** accommodate unintended situations where resource usage is oversubscribed. The action to be taken in such situations ***should*** be specified as part of the requirements on the design.

Note: *Examples of these situations include buffers overflowing, exceeding a rate group time boundary, and excessive inputs or interrupts. There are several common methods for tolerating these situations, most of which relate to reducing demand from non-essential items, especially if they are the source of over subscription:*

a) Generate warning messages when appropriate.
b) Instruct external systems to reduce their demands.
c) Lock out interrupts.
d) Change operational behavior to handle the load. For example, the software may use faster but less accurate algorithms to keep up with the load.
e) Reduce the functionality of the software, or even halt or suspend a process or shutdown a computer.

3.8.2.5.5 Response to Missing Inputs

Software ***should*** be designed to tolerate and continue functioning in situations where inputs are temporarily missing.

Note: *An example of this situation is resorting to dead reckoning for navigation as long as navigation measurements are not available from hardware.*

Input/output completion time-outs are often used to detect a failed input/output transaction and restore continuity.

When writing to output, it is often good practice to read it back to verify that the write completed successfully.

3.8.2.5.6 Response to Failed I/O

Software ***shall*** be specified and designed to maintain the aliveness and continuity of the software when failed input/output transactions or other processing or interactions fail to complete.

Note: *Watchdog timers are commonly used for when there is a failure to complete a certain process or action. Upon completion of a defined processing path, the software resets a watchdog timer. If the processing gets lost, or fails to make progress, the timer times-out. The timer directs the software to a known point where the processing is restored.*

3.8.2.5.7 Response to Nominal and Off-Nominal Inputs

Software ***shall*** accommodate both nominal inputs (within specifications) and off-nominal inputs, from which recovery may be required.

3.8.2.6 Protection Against Incorrect Memory Use

Software ***shall*** be designed to prevent incorrect use of memory with the following considerations:

a. Execution in data areas, unused areas, and areas not intended for execution.
b. The updating of code/software be limited to a single target memory device under user ground control and monitoring at a time. If dual memory units are incorporated in the design, under no circumstances are the prime and redundant memories to be modified concurrently, or before the operational performance of the change is properly assured in a single unit.

3.8.2.6.1 Data Set Consistency

Software ***shall*** be designed to ensure that data sets and parameter lists are consistent with respect to time when passed among processes such as software subsystems, rate groups, and others. The software design ***should*** not allow execution to be interrupted in a manner that permits it to use both old and new components of a vector.

3.8.2.6.2 Self-Test Capability and Fault Diagnostic

The software design ***shall*** include capabilities to test operation and permit timely fault diagnostics with the following considerations.

a. If not removed, the test capabilities do not cause flight hardware damage or interfere with proper operation of the flight software if inadvertently executed in flight.
b. If removed, rerun the regression test baseline and perform V&V testing after removal.

Note: *Examples of test capabilities include, testing the underlying computing hardware, production of diagnostic traces, hooks for debuggers, introspection* *capabilities, instrumentation of performance or resource use, simulations for closed loop control, and special modes that support scripted I/O test.*

Note: *In order to get the most benefit from it, the test/diagnostic code* ***should*** *be designed by PDR, and incorporated into the software by CDR. This requirement is primarily applicable to newly developed code. The applicability to commercial or reused code will need to be evaluated on a project specific and application specific basis.*

3.8.2.6.3 Measurement of Constrained Resources

Software ***shall*** be designed to provide timely visibility into the use of computing resources during testing and operations.

Note: *Examples of resources to measure are: real time tasks, background tasks, throughput, memory, bus utilization, stack size and headroom, cycle slip statistics, fragmentation, memory leaks, and allocation latency. This makes it possible to validate margins and makes the flight software resource usage testable.*

3.8.3 On-Board Data Management

3.8.3.1 Protection of Critical Data

The spacecraft data system ***shall*** be designed to protect critical data from loss in the event of selected anomalies (e.g., transient power outage). The design process ***should*** include an analysis showing that protected critical data is transmitted to Earth after an anomaly as soon as practical.

3.8.3.2 Redundant Handling of Critical Data

The spacecraft data system ***shall*** (design guidance for Class D) be designed to allow simultaneous real-time transmission and on-board storage of mission critical data (e.g., fly-by science, orbit insertion, etc.).

3.8.3.3 Compatibility with Tracking Outages

The spacecraft data system ***shall*** be designed to enable storage of time critical science data and spacecraft health and status engineering telemetry data during long non-track periods and accommodate flight operational uncertainties caused by weather effects or ground tracking station problems. The system ***shall*** (design guidance for Class D) have sufficient storage to cover the missed tracking period as well as subsequent tracked periods, as defined by the Project, while the stored telemetry data is being recovered.

3.8.3.4 Multiple Restart Flight Software Initialization

The software ***shall*** be designed to detect off-nominal restarts and to successively reinitialize with less and less dependency on preserved state (e.g. inertial, temporal, device capability or configuration, file allocation tables, boot code in RAM, etc.) from before the most recent reset, until a fully known and tested initial configuration is obtained, and until stable operation has been restored.

Note: *Reset is commonly used as a means of autonomous recovery from serious software problems caused by errors or single event upsets. Reset is not effective unless the problematic software state is cleared during re-initialization. Ultimately, all software states must be presumed suspect and expendable, if prior re-initializations have failed to resolve a problem. A complete accounting of preserved state is essential, if effective measures are to be taken against it.*

3.9 Fault Management

3.9.1 General

NASA Fault Management Handbook, NASA-HDBK-1002, ***should*** be used as a guide for defining, developing, analyzing, evaluating, testing, and operating the Fault Management (FM) element of flight systems.

3.9.1.1 Management of Credible Single Faults

Fault management software ***shall*** have the ability to detect and respond, in a timely, deterministic way, to single-fault scenarios deemed credible by system-level hazard analysis. For example, the system should be able to be in a known, deterministic state through an under-voltage situation and should be able to recover when voltage conditions improve.
Failure scenarios where fault management is not practical ***should*** be handled through “design for minimum risk” (e.g. failures in propulsion lines, tanks). The failures should not propagate beyond the fault containment region.

3.9.1.2 Fault Management Response During Time-Critical Mission Activities

The fault management response ***shall*** be designed to autonomously re-establish the needed spacecraft functionality to permit safe, reliable and timely completion of the mission critical activity.

3.9.1.3 Fault Management Response During Non-Time-Critical Mission Activities

The fault management response ***shall*** be designed to, at a minimum, autonomously configure the spacecraft to a safe, quiescent, ground command-able state, transmitting periodically, at least an RF carrier downlink signal during non-mission-critical periods following a fault condition. The safe state ***shall*** be power-positive and preserve spacecraft resources. The safe state ***should*** be as simple as practical and employ minimum hardware to maintain a safe attitude.

Note: *A safe state is a state in which the spacecraft thermal condition and inertial orientation are stable, the spacecraft is commandable and is transmitting a downlink signal, and requires no immediate commanding to ensure spacecraft health and safety that preserves vital spacecraft resources.*

3.9.1.4 In-Flight Ability to Command and Parameter Visibility

The fault management system ***shall*** not allow a lockout of ground command capability, e.g. do not turn off the onboard communication receiver.
The fault management system ***shall*** be designed as in-flight-command-able to permit changing the state of enable/disable parameters and other pertinent parameters as needed for system reconfiguration, such as threshold and persistence values. This will enable "tuning" the fault management system based on actual in-flight system performance.
The observed parameters should reflect the spacecraft's current state. The parameters ***shall*** (design guidance for Class D) be telemetered and made available for timely flight team use. This will support ground-based diagnosis, or post-mortem analysis, of spacecraft state and actions.

3.9.1.5 Fault Indication Filtering

The design ***should*** select the enable/disable, trigger, and persistence values for fault indication filtering to ensure safety but not be "hair triggered" to cause inadvertent Fault Response entry/execution (e.g., false alarm).

Note: *Trigger is the specified threshold min/max alarm values. Persistence is when the condition for the threshold values (min/max) must hold for a specified amount of time before an action is executed. For example, if a sensor is noisy, the specified threshold may be crossed or triggered even though the component could be healthy.*

3.10 Ground Support Equipment

3.10.1 General

3.10.1.1 Acceptability for Interface to Flight Hardware

GSE that will interface with flight hardware ***shall*** be manufactured and assembled in accordance with the workmanship standards specified in Section 3.11.4, Workmanship, of this document.

3.10.1.2 GSE Testing

GSE that will interface with flight hardware ***shall*** be functionally tested prior to integration with flight hardware to verify compliance with the interface specifications defined for the GSE.

3.10.1.3 GSE Use at Launch Site

All testing of operations of flight systems at the launch site or in the field ***should*** only use GSE and test configurations, that functionally represent the launch or flight configuration, that have been previously used with the flight hardware.

Note: *In many cases it is too expensive to develop GSE to be used at the development location that is also compatible with the launch site. Often in these cases launch site provided GSE is used which also dictates new test configurations.*

3.10.1.4 Electrical Fault Propagation

Electrical GSE ***shall*** prevent failures in the GSE from propagating to the flight hardware.

3.10.1.5 GSE Software

Software used in GSE ***shall*** comply with the requirements of APR 7150.2. There ***should*** be no commands that are unique or different from flight software commands for GSE that perform a flight hardware function.

3.10.1.6 Fluid System GSE Fault Tolerance

Fluid systems GSE used to pressurize flight systems ***shall*** be single-fault tolerant against over-pressurizing the flight system.

3.10.1.7 Connector Savers

Connector savers ***shall*** (design guidance for Class D) be used on all flight connectors to prevent using up the flight connector cycle life.

3.11 Parts, Materials, and Processes

3.11.1 General

Parts and materials ***should*** be chosen from experience on previous flown space hardware, whenever possible.

Selection of parts and materials ***should*** emphasize proven performance, minimum contamination potential and suitability for one-year storage plus the projected on-orbit lifetime for the mission.

Materials ***should*** be procured from qualified suppliers and ***should*** be inspected to ensure that they meet the applicable specifications in this document. MAPTIS online database (http://maptis.nasa.gov/) ***should*** be used as a guide for selection of materials.

For hardware and material that is in contact with sensitive payloads such as biological or chemical, material and bio-compatability ***should*** be assessed or tested.

A gap analysis ***should*** be performed to ensure that previously flown hardware is appropriate for use in its new flight conditions and environment.

3.11.2 Parts, Materials, and Processes List

The project ***shall*** prepare and maintain an "As-Built" parts list for all components and subsystems which includes all parts and materials, by generic part number and manufacturer, used in space flight hardware.

3.11.3 Limited Shelf-Life Materials

3.11.3.1 Limited Life Item Tracking

Life start and expiration dates for all limited-life components and materials ***shall*** be recorded and tracked.

3.11.3.2 Limited Life Item Marking

Where practical, components and materials ***shall*** be marked with expiration dates.

Note: *It may not be practical to mark some materials such as fluids or some components due to size limitations, in these cases, tracking paperwork and logs can be used to record material content, expiration dates, etc.*

3.11.4 Workmanship

The design and workmanship standards listed in the NASA workmanship and IPC standards listed below, or their equivalent, ***shall*** be used unless noted otherwise below.

Note: *COTS parts, which the project has received approval for use, are exempted from this requirement. However, all COTS parts reworked at Ames* ***shall*** *meet the workmanship requirements identified below.*

a. Conformal Coating and Staking: NASA-STD-8739.1, “Workmanship Standard for Staking and Conformal Coating of Printed Wiring Boards and Electronic Assemblies”

b. Soldering – Flight, Surface Mount Technology: J-STD-001, Space Applications Electronic Hardware Addendum to Requirements for Soldered Electrical and Electronic Assemblies, “Surface Mount Technology”

c. Soldering – Flight, Manual (hand): J-STD-001, Space Applications Electronic Hardware Addendum to Requirements for Soldered Electrical and Electronic Assemblies, “Soldered Electrical Connections”

   (1) All materials at a solder joint ***should*** be selected to avoid the formation of potentially destructive intermetallic compounds

d. Soldering – Ground Systems: Association Connecting Electronics Industries (IPC)/Electronics Industry Alliance (EIA) J-STD-001, Space Applications Electronic Hardware Addendum to Requirements for Soldered Electrical and Electronic Assemblies

e. Fusion Welding – NASA-STD-5006: General Fusion Welding Requirements for Aerospace Materials Used in Flight Hardware

f. Electronic Assemblies – Ground Systems: IPC-A-610, “Acceptability of Electronic Assemblies”

g. Crimping, Wiring, and Harnessing: NASA-STD-8739.4, “Crimping, Interconnecting Cables, Harnesses, and Wiring”

h. Fiber Optics: NASA-STD-8739.5, “Fiber Optic Terminations, Cable Assemblies, and Installation”

<u>Printed Wiring Board (PWB) Design</u>

a. IPC-2221, “Generic Standard on Printed Board Design”

b. IPC-2222, “Sectional Design Standard for Rigid Organic Printed Boards”

c. IPC-2223, “Sectional Design Standard for Flexible Printed Boards”

d. IPC D-275 “Design Standard for Rigid Printed Boards and Rigid Printed Board Assemblies”

<u>PWB Manufacture</u>

a. IPC A-600, “Acceptability of Printed Boards”

b. IPC-6011, “Generic Performance Specification for Printed Boards”
c. IPC-6012, “Qualification and Performance Specification for Rigid Printed Boards”
d. IPC-6013 “Qualification and Performance Specification for Flexible Printed Boards”
e. IPC-6018 “Microwave End Product Board Inspection and Test.”

3.11.5 Parts & Material Traceability

All parts and materials ***shall*** be identified and traceable to a specific manufacturer lot number or lot date code. Certification of Conformance from the manufacturer ***shall*** be obtained for all procured parts. Lot date code limitations for Electrical, Electronic and Electromechanical (EEE) parts ***shall*** (design guidance for Class D) be at least five years.

Note: *COTS parts, which the project has received approval for use, are exempted from this requirement.*

3.11.6 Outgassing

Refer to Section 4.6., Contamination Control, for requirements and guidance regarding outgassing and the effects of contamination on the spacecraft and instruments.

3.11.7 Mechanical Parts Selection

Mechanical parts ***should*** be chosen based on similar applications on prior space flight programs whenever possible. Major factors such as outgassing, flammability, contamination, aging, stability, corrosion, applicability for solder joints, and electrical properties of materials ***should*** be considered.

New design mechanical parts ***shall*** (design guidance for Class D) be analyzed and/or tested to ensure adequate performance.

Thorough evaluation of the environmental effects of the mission paths/orbits and life time ***shall*** (design guidance for Class D) be assessed for the impact on materials selection and design.

3.11.8 Finishes

The finishes used ***should*** be such that completed devices are resistant to corrosion. The design goal ***should*** be that there would be no destructive corrosion of the completed devices when exposed to moderately humid or mildly corrosive environments that could inadvertently occur while unprotected during manufacture or handling, such as possible industrial environments or sea coast fog that could be expected prior to launch. Destructive corrosion ***should*** be construed as being any type of corrosion which interferes with meeting the specified performance of the device or its associated parts. Protective methods and materials for cleaning, surface treatment, and applications of finishes and protective coating ***should*** be in accordance with MIL-F-7179. Cadmium, tin, and zinc coatings ***should*** not be used. Chromium plating ***should*** be in accordance with QQ-C-320. Nickel plating ***should*** be in accordance with QQ-N-290. Corrosion protection of magnesium ***should*** be in accordance with MIL-M-3171. Coatings for aluminum and aluminum alloys ***should*** be in accordance with MIL-C-5541 or MIL-A-8625.

3.11.9 Prohibited Materials

The following materials ***shall*** not be used:

a. Pure cadmium, magnesium, zinc or selenium, except internal to hermetically sealed devices
b. Unalloyed tin
c. Corrosive solder fluxes, unless detailed cleaning procedures are specified, along with appropriate verification methods to insure removal of residual contaminants
d. Mercury and compounds of mercury
e. Materials that exhibit or are known to exhibit natural radioactivity such as uranium, potassium, radium, thorium, and/or any alloys thereof
f. Materials that exhibit or are known to exhibit health hazards such as beryllium, toluene, lithium and/or any alloys thereof except in the application of lithium-ion or lithium-polymer batteries

Note: *COTS parts may require a waiver to this requirement since COTS parts typically do not come with a list of materials.*

Note: *NASA-STD-6016, Standard Materials and Processes Requirements for Spacecraft,* ***should*** *be used as a guide.*

3.11.10 Dissimilar Materials

The use of dissimilar metals ***should*** be in accordance with MIL-STD-889.

The use of dissimilar metals in the intended environment ***should*** consider the following at a minimum:

a. Galvanic corrosion
b. Liquid metal embrittlement

c. Galling
d. Wear properties

4 INTEGRATION AND ENVIRONMENTAL TEST

Section 4 provides both requirements and design guidance. Requirements are provided in the form of ***shall*** statements, design and test guidance is in the form of ***should*** statements. If a mission classification warrants a different approach to a requirement, it is explicitly noted.

All ***shall*** statements are to be verified while ***should*** statements do not require verification. However, at the appropriate milestone reviews the project team ***shall*** be able to explain how the integration and test guidance was considered.

Although test levels are provided for qualification and acceptance tests, a qualification test article is not a requirement for Class D spacecraft. It is expected that most, if not all, Class D spacecraft will be Protoflight development programs.

Some tests may be eliminated for components and subsystems that are not sensitive to the environments identified in Section 4. These tests ***should*** be identified and vetted early in the project life-cycle in order for all project activities to be appropriately scoped.

4.1 Integration and Test Plan

The project ***shall*** develop a project specific Integration and Test (I&T) Plan by the Critical Design Review (CDR) which addresses integration from the component level to assembled spacecraft ready for delivery to the launch site that includes the following:

a. "Test Like You Fly (TLYF) – Fly Like You Test" approach
b. required test facilities
c. required test equipment including GSE, EGSE and software tools
d. sequence, duration, and schedule for testing in facilities outside the direct control of the project including any departure from the recommended sequence shown in Figure 4.2.2.1 below.
e. verification activities at each level of integration
f. definition of all ground environments including storage, transportation, and prelaunch operations and approach to ensuring hardware is not subjected to environments beyond those specified in Section 4 of this document
g. tests cover a range of operational conditions that envelope worst case environments for all operating parameters anticipated for the mission
h. pass/fail criteria at all levels of testing at component, sub-system, and system level, including testing the Fault Management system's failure scenarios and response.

Note: *NASA-STD-7002, Payload Test Requirements* ***should*** *be used as a guide in developing the Integration and Test Plan.*

The design ***should*** incorporate test and telemetry points to allow verification of functional performance. The design ***should*** accommodate easy installation and replacement of major components during factory assembly and of explosive ordnance devices, batteries, and other site replaceable items at the launch site when mated to the launch vehicle. Access ***should*** be provided to those test plugs, harness break-in points, external umbilical connections, safe and arm devices, explosive ordnance devices, pressurant and propellant fill and drain valves, and other devices as might be required for prelaunch maintenance, alignment, and servicing. Alignment references for critically aligned components ***should*** be visible directly or through windows or access doors.

All test equipment and tools ***shall*** be calibrated.

4.1.1 System Level Functional & Performance Verification

4.1.1.1 Release Mechanisms for Flight Deployables

A release mechanism test for the flight deployable components ***shall*** be performed. A first motion test of the mechanism ***shall*** be performed at the system level.

4.1.1.2 End-to-End Data System Testing

System end-to-end testing ***shall*** be performed using actual hardware or simulation, from input to instrument(s), through the spacecraft, transmitted to receiving antennas, and through the ground system - reconciled against what is physically achievable before launch, and consistent with associated mission risk.

Ground test of the fully integrated flight software system ***shall*** include demonstration of error free operations-like scenarios over an extended time period. The minimum duration uninterrupted flight software system-level test (on the highest fidelity flight software testbed) ***shall*** be 100 hours.

4.1.1.3 Transmitter RF Power Output Testing

All beacon and radio transmitters ***shall*** be tested for RF power output either by direct measurement using a power spectrum analyzer or by physical separation of transmitter and receiver (25 miles or more).

4.1.1.4 ADCS Sensors and Actuators

All attitude determination and control sensors and actuators ***shall*** undergo end-to-end (i.e., from sensor stimulus to actuator response) phasing/polarity testing after spacecraft integration in the final flight configuration (hardware and software). End-to-end testing ***shall*** be performed for all sensor-to-actuator combinations.

Note: *Inadequate verification of signal phasing or polarity can result in unexpected on-orbit performance and possible loss of mission. Component-level and end-to-end phasing tests and flight software mitigations can ensure correct operation.*

4.1.1.5 Mechanical Clearances

Inspection of mechanical clearances and margins (e.g. potential reduced clearances after insulation blanket expansion) ***shall*** be performed on the final as-built hardware.

4.1.1.6 Software Regression Testing

New versions of flight software delivered during integration and test ***shall*** undergo regression testing on the flight vehicle prior to use in system level verification.

Note: *Regression testing demonstrates that there are no obvious, unintended changes to the software, and verifies the functionality of new or changed capabilities in this delivery.*

Note: *The regression test is not a substitute for thorough pre-delivery verification of the flight software on testbeds.*

4.1.2 General I&T Requirements

4.1.2.1 Capping of Test Points and Plugs

All test points and plugs ***shall*** be capped or protected from discharge for flight.

Note: *Capping open connectors provides protection from electrostatic discharge resulting from space charging.*

4.1.2.2 Non-Condensing Environment

The I&T plan ***shall*** identify controls to preclude condensation formation on the spacecraft.

4.1.3 Qualification of Heritage Hardware

All use of heritage flight hardware ***shall*** be qualified for use in its new application which takes into consideration necessary design modifications, changes in the expected environments, and differences in operational use. Qualification for use by similarity is acceptable if the above conditions are met. An acceptance test program ***shall*** be implemented for these components and subsystems.

4.2 Environmental Verification

4.2.1 Level of Assembly Test Requirements

This document 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. If this is not the case or if it is not practical to test at each of these levels the project must develop a verification program that is appropriate for the mission's level of risk taking into account that testing may include levels of assembly in-between the three levels given in Table 4.2.1-1 and the overall verification program continues through on-orbit operations.

The environmental tests identified in Table 4.2.1-1 ***shall*** include the highest level of assembly being delivered. Testing at lower levels of assembly may be eliminated providing associated schedule risk assessment and mitigation are approved by stakeholders.

Table 4.2.1-1
Environmental Test Requirements[8]

	Unit/ Component	Subsystem/ Instrument	Payload/ Spacecraft[9]
Structural			
Vibration/ Acoustic	X		X
Shock	X		X
Modal Survey/Sine Sweep		X[1]	X[1]
Thermal			
Thermal Vacuum/Cycling[2]	X[3]		X[4]
Thermal Balance			X
Pressure	X		
Leakage	X[5]	X[5]	X[5]
EMI/EMC	X		X (self-comp)
Life[6]	X		X[7]

1 The requirements for these tests are based on launch vehicle specific requirements and need to be discussed with the launch provider.

2 Temperature cycling at ambient pressure may be substituted for thermal vacuum temperature cycling if the component can be shown to be unaffected by a vacuum environment. For ambient pressure testing the number of cycles ***should*** be increased by 50% to account for possible analytical uncertainties and to increase the probability of detecting any workmanship defects.

3 8 cycles with a minimum soak time of 4 hours for each hot and cold test point. For thermal cycling the maximum rate of change of temperature ***shall*** not exceed acceptable limits (based on HW characteristics or orbital predictions).

4 4 cycles with a minimum soak time of 8 hours for each hot and cold test point. For thermal cycling the maximum rate of change of temperature ***shall*** not exceed acceptable limits (based on HW characteristics or orbital predictions).

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

6 Life testing is required at a minimum for electronics burn-in testing. Other spacecraft unique life tests may be required.

7 Additional burn-hours may be required at the spacecraft level if the hour requirements of paragraph 3.3.1.1 have not been satisfied.

8 Goddard document General Environmental Verification Standard, GSFC-STD-7000, ***should*** be used as a guide for all environmental testing.

9 Highest level of assembly being delivered.

4.2.2 Test Sequence

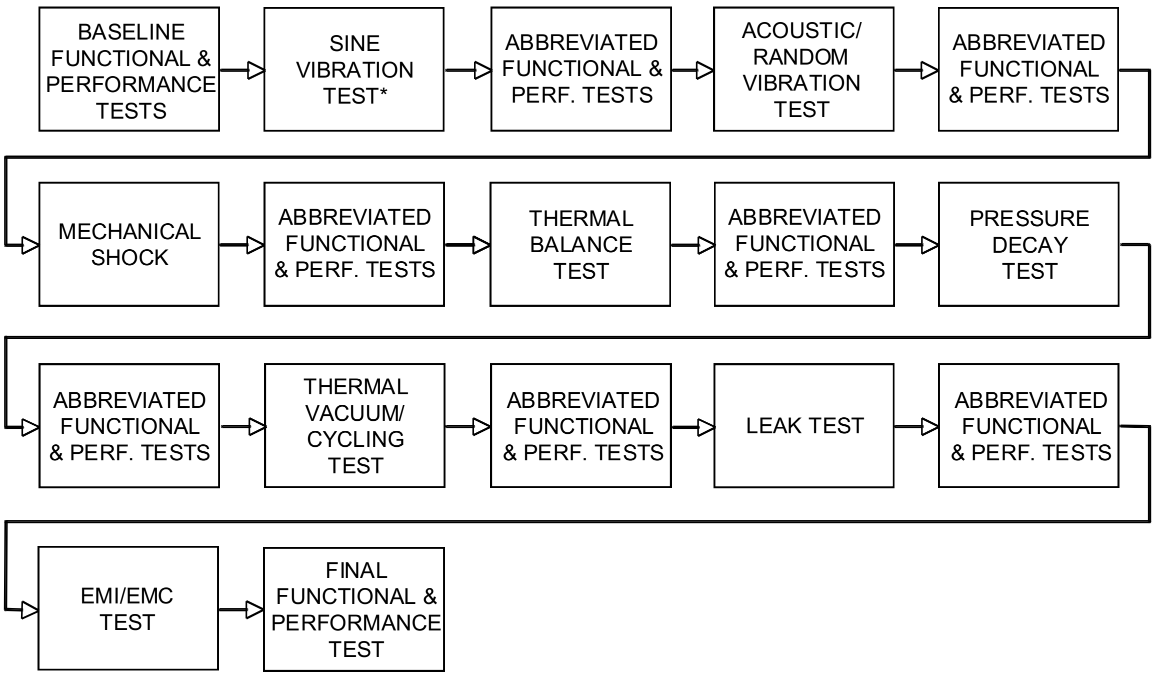
The suggested test sequence is shown in Figure 4.2.2-1. This is sequence is based on the knowledge that mechanical stress tests ***should*** occur before thermal stress tests so that workmanship and material defects can more easily be uncovered. Projects can modify this sequence based on spacecraft and component specific designs, however, the project must show that the change in sequence will not compromise the ability to uncover defects. For some spacecraft, it may be acceptable and cost effective to conduct sine vibration, random vibration, and mechanical shock in series without abbreviated functional tests in between. All departures from the recommended sequence ***shall*** be documented in the I&T Plan.

Perform dynamic tests prior to performing thermal-vacuum tests on flight hardware. Sinusoidal vibration, random vibration, pyroshock, and acoustics, as required. For most launch vehicles, the sine vibration environment is benign and no sinusoidal vibration test is required. In the case of an Atlas rocket a sinusoidal vibration test is required. The order among these dynamics tests may be interchanged. Experience has shown that until the thermal-vacuum tests are performed, many failures induced during dynamics tests are not detected because of the short duration of the dynamics tests. In addition, the thermal-vacuum test on flight hardware at both the assembly level and the system level provides a good screen for intermittent as well as incipient hardware failures.

During the normal flight sequence, the launch environment is followed by vacuum and potential temperature extremes. In this flight sequence, the flight hardware is therefore exposed to acoustics and vibration followed by vacuum and temperature variations. Consequently, by performing dynamics tests prior to thermal-vacuum tests, the actual flight sequence is simulated. Also, if the flight sequence produces synergistic effects, the synergism will be simulated. In addition, preserving the sequence of the service environments in the environmental test program is a widely accepted practice. As a result, the effect of reversing the test sequence on spacecraft failure rates has not been quantified. However, evidence exists that many acoustic induced failures have not been detected until the spacecraft is exposed to the thermal-vacuum environment. These failures may not be detected during acoustics tests because of the short one-minute duration or a non-operating power condition. Typically, the identified failures that could be related to or caused by the dynamic acoustic environment were bad solder joints, intermittents, bad bearings, broken wires, poor welds, leaks, foreign materials, etc.

Inspections ***shall*** be performed prior to the start of qualification, protoflight and/or acceptance tests to ensure that the unit is acceptable to start testing. Items to be inspected include connectors, fasteners and mounting surfaces. Connector savers ***shall*** (guidance for Class D) be used during testing of flight hardware and the connector that mates with flight hardware ***shall*** (guidance for Class D) be a flight connector.

Baseline and final functional tests ***shall*** be conducted and compared to each other to look for trends or degradation that might affect the life of the mission.



*** If required by the launch provider**

Recommended Test Sequence
Figure 4.2.2-1

4.2.3 Test Tolerances

The testing tolerances ***shall*** meet or exceed those shown in Table 4.2.2.1-1 below.

Table 4.2.2.1-1
Testing Tolerances

Environment	Test Parameter	Tolerance
Acoustics		
	Overall Level	≤1 dB
	1/3 Octave Band Frequency Tolerance (Hz)	
	F ≤ 40	+3 dB, -6 dB
	40 ≤ F ≤ 3150	±3 dB
	F ≥ 3150	+3 dB, -6 dB

Environment	Test Parameter	Tolerance
Antenna Pattern Determination		
		±2 dB
Electromagnetic Interference		
	Voltage Magnitude	± 5% of peak value
	Current Magnitude	± 5% of peak value
	RF Amplitudes	±2 dB
	Frequency	±2%
	Distance	± 5% of specified distance
Humidity		
		±5% RH
Loads		
	Steady-State (Acceleration)	±5%
	Static	±5%
Magnetic Properties		
	Mapping Distance Measurement:	±1 cm
	Displacement of assembly center of gravity (cg) from rotation axis	±5 cm
	Vertical displacement of single probe centerline from cg of assembly	±5 cm
	Mapping turntable angular displacement:	±3 degrees
	Magnetic field strength	±1 nT
	Repeatability of magnetic measurements (short term)	± 5% or ± 2 nT whichever is greater
	Demagnetizing and Magnetizing Field Level: from rotation axis	± 5% of nominal
Mass Properties		
	Weight	±0.2%
	Center of Gravity	±0.15 cm
	Moments of Inertia	±1.5%
Mechanical Shock		
	Response Spectrum	+25%, -10%
	Time History	± 10%
Pressure		
	Greater than 1.3 X 104 Pa	±5%

Environment	Test Parameter	Tolerance
	(Greater than 100 mm Hg)	
	1.3 X l04 to 1.3 X l02 Pa (l00 mm Hg to 1 mm Hg):	±10%
	1.3 X l02 to 1.3 X 101 Pa (1 mm Hg to 1 micron):	±25%
	Less than 1.3 X 101 Pa (less than 1 micron):	±80%
Temperature		
		±2°C
Vibration		
Sinusoidal	Amplitude	±10%
	Frequency	±2%
Random	RMS level	±10%
	Acceleration Spectral Density	±3 dB

4.2.4 Test Article Types

Class D spacecraft will almost exclusively be Protoflight development programs. However, Class D projects are not precluded from developing a Qualification Unit and therefore the Sections 4.3 through 4.5 provide Qualification and Acceptance test criteria in addition to Protoflight test criteria.

4.2.4.1 Qualification Testing

Qualification tests are tests conducted on a dedicated test article not intended for flight to demonstrate that the design, manufacturing process, and acceptance program produce mission items that meet specification requirements. In addition, the qualification tests validate the planned acceptance program including test techniques, procedures, equipment, instrumentation, and software.

4.2.4.2 Protoflight Qualification Testing

Protoflight qualification, or otherwise known as protoflight testing, is required for all new designs in order to demonstrate that adequate design margins exist in the final product to assure that the specification and operational requirements are met. The objective of protoflight testing is to judiciously increase test levels above the acceptance test levels and durations to uncover deficiencies in the design and methods of manufacture without stressing the hardware beyond the limits of the design and operational capabilities. Protoflight levels are generally 3 dB above maximum measured flight loads for dynamic testing and ±5°C beyond acceptance test temperature levels. Protoflight hardware is intended to be used as flight hardware for the associated mission spacecraft or demonstration.

4.2.4.3 Acceptance Testing

Acceptance tests are tests that are conducted to demonstrate acceptability of an item for flight. Acceptance tests measure performance parameters and reveal inadequacies in manufacturing process such as workmanship or material. Acceptance tests demonstrate acceptable performance over the specified range of mission requirements.

After a component has been tested to protoflight levels, each subsequent flight component only requires acceptance testing. Acceptance testing is not required for a component that has already been subjected to protoflight qualification tests.

4.3 Structural and Mechanical

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 documented requirements such as factors of safety, interface compatibility, structural reliability, and workmanship.

Table 4.3-1 specifies the structural and mechanical verification activities required at the component and spacecraft level. 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 including launch, orbit insertion, and preparation for orbital operations. For analyses, the factors of safety are as specified in paragraph 3.2.1.3 of this document (consistent with NASA-STD 5001). NASA-STD-5002, Load Analyses of Spacecraft and Payload, ***should*** be used as a guide in developing the analysis process and methodology.

Table 4.3-1
Structural & Mechanical Verification Requirements

Requirement	Unit/ Component	Subsystem/ Instrument	Spacecraft/ Payload
Structural Loads			
Modal Survey/Sine Sweep		T[5]	
Design Qualification		(T,A)/A[1]	A
Structural Reliability			
Primary & Secondary		A, T[2]	
Vibroacoustics			
Acoustic Vibration	T[3]		T[4]
Random Vibration	T		T[4]
Sine Vibration	T[5]		T[5]
Shock	T[6]		T

Pressure	A	A, T[5]	
Mechanical Function		A, T	A, T
Mass Properties		A, T[5]	A/T

A Analysis required

T Test required

A/T Analysis and/or Test is required

A,T Analysis & Test is required

1 Analysis & Test OR analysis only if "Qualification by Analysis Only" factors of safety are used.
2 Combination of fracture analysis and proof tests on selected elements.
3 Required for components with high surface area/mass ratios (par. 4.3.2, 4.3.3) in lieu of a random vibration test.
4 Either an acoustic or random vibration test is required at the spacecraft level.
5 Test must be performed unless assessment justifies deletion.
6 Test required for all self-induced shocks, but may be performed at the spacecraft level for externally induced shocks if the component is shock tolerant.

Coupled Loads Analysis

A Coupled Loads Analysis (CLA) will be performed by the launch provider of the integrated launch vehicle (launch vehicle plus spacecraft). For the launch vehicle verification activity to occur, the project will likely be required to provide a test-verified finite element model of the spacecraft that simulates the mass and stiffness of the spacecraft. The model ***shall*** be of sufficient detail to make possible an analysis that defines the spacecraft's modal frequencies and displacements below a specified frequency that is dependent on the fidelity of the launch vehicle finite element model. Verification of the model is dependent on the launch vehicle provider and program requirements.

The frequency below which a modal test is required is dependent on the specific launch vehicle. The determination will be made on a case-by-case basis and specified in the design and test requirements. Modal tests are generally performed at the subsystem level of assembly (i.e., primary and secondary structure), but may be required at other levels of assembly depending on launch vehicle specific requirements.

4.3.1 Test Factors and Durations

The test factors and durations ***shall*** be according to Table 4.3.1-1 below.

Table 4.3.1-1

Test Factors & Durations

Test Type	Qualification	Protoflight	Acceptance
Structural Loads Level Duration Centrifuge/Static Load	 1.25 x Limit Load 30 seconds	 1.25 x Limit Load 30 seconds	 1.0 x Limit Load 30 seconds
Random Vibration Level Duration	 Limit Level + 3 dB 2 minutes/axis	 Limit Level + 3 dB 1 minute/axis	 Limit Level 1 minute/axis
Acoustics Level Duration	 Limit Level + 3 dB 2 minutes	 Limit Level + 3 dB 1 minute	 Limit Level 1 minute
Sine Vibration Level Sweep Rate	1.25 x Limit Level 2 oct/minute	1.25 x Limit Level 4 oct/minute	 Limit Level 4 oct/minute
Mechanical Shock Actual Device Simulated	 2 actuations 1.4 x Limit Level 2/axis	 2 actuations 1.4 x Limit Level 1/axis	 1 actuation Limit Level 1/axis

4.3.2 Random Vibration

Random vibration inputs ***shall*** be applied at the base of the adapter or fixture in each of three orthogonal directions (X, Y and Z) with all mechanical and electrical items in their launch configuration and monitored for failures and intermittent operations if the component or subsystem is powered during launch. Power spectral density plots of both input and response random vibration data ***shall*** be made and included in deliverable acceptance test reports.

In some cases, it is more appropriate to conduct an acoustic vibration test rather than a random vibration test of a component.

An acoustic vibration test ***shall*** be conducted on components that have a large surface area to mass ratio (such as solar arrays and large antennas) since the component will be excited by the acoustic vibrations more than by the structural-born vibrations. Components that have an area to mass ratio of 0.21 m2/kg (145 in2/lb) or greater ***should*** be considered for acoustic vibration testing.

Note: *NASA-STD-7001* ***should*** *be used as a guide in developing the random vibration test approach.*

4.3.2.1 Component Level

Components ***shall*** be subjected to random vibration along each of three mutually perpendicular axes for one minute each. Components ***shall*** operate as required during, if appropriate, and after application of the random vibration environment.

When possible, the component random vibration spectrum ***shall*** be based on levels measured at the component mounting locations during 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 spacecraft. 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 4.3.2.1-1 may be used.

Table 4.3.2.1-1
Components Generalized Random Vibration Test Levels
22.7-kg (50-lb) or less

Frequency (Hz)	ASD Level (g^2/Hz)	
	Qualification/Protoflight	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}

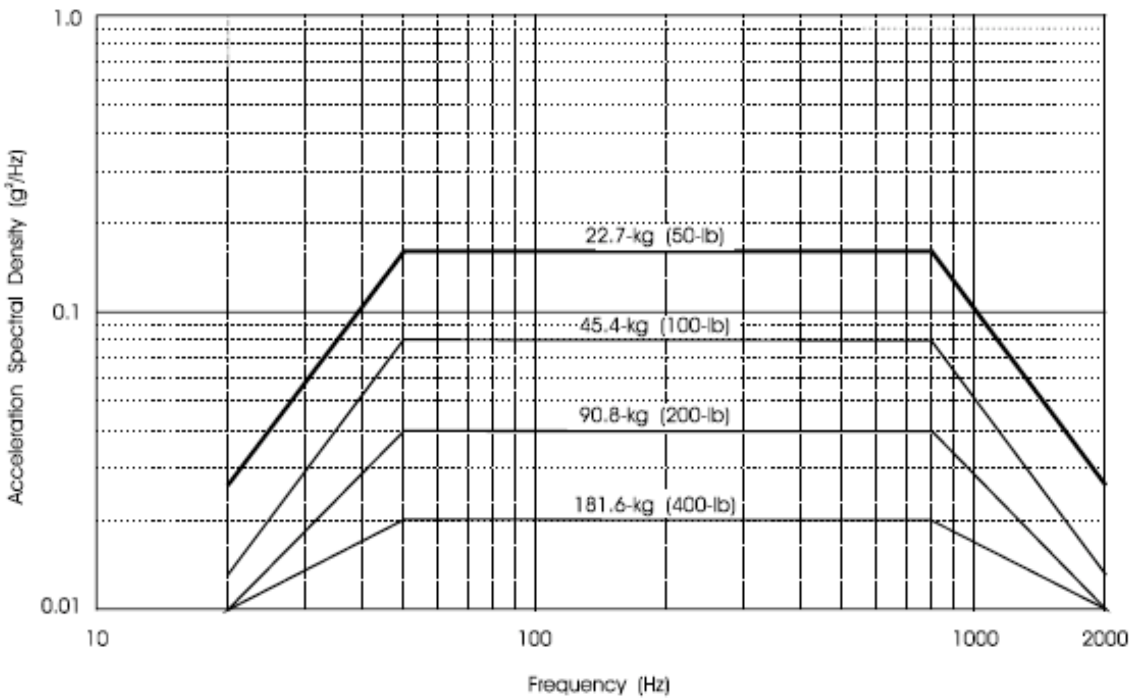
The acceleration spectral density (ASD) level may be reduced for components weighing more than 22.7 kg (50 lb) according to:

	Weight in kg	
dB reduction	= 10 log (W/22.7)	
ASD (50-800 Hz)	= 0.16 (22.7/W)	for protoflight & qualification
ASD (50-800 Hz)	= 0.08 (22.7/W)	for acceptance

See Figure 4.3.2.1-1

The slopes ***shall*** be maintained at ±6 dB/oct for components weighing up to 59 kg (130 lb). Above 59 kg the slopes ***shall*** be adjusted to maintain an ASD level of 0.01 g2/Hz at 20-2,000 Hz.

For components weighing over 182 kg (400 lb) the test level ***shall*** be maintained at the level for 182 kg (400 lb).



Components Generalized Random Vibration Test Levels
Figure 4.3.2.1-1

4.3.2.2 Spacecraft Level

At the spacecraft level the area to mass ratio is typically too low to be excited by the random vibration environment. At the spacecraft level the area to mass ratio is typically high enough that the structure is excited by acoustic vibration.

An evaluation ***shall*** be made to determine the appropriateness of either a random vibration or acoustic vibration test based, at a minimum, on the criteria discussed in paragraph 4.3.2. If a random vibration test is appropriate then the project ***should*** obtain the spacecraft level random vibration environment from the launch vehicle provider.

Note: *GSFC-STD-7000 (GEVS), Section 2.4.2,* ***should*** *be used as a guide.*

4.3.3 Acoustic Vibration

Note: *NASA-STD-7001* ***should*** *be used as a guide in developing the acoustic vibration test approach.*

4.3.3.1 Component Level

Components that have a large surface area to mass ratio (such as solar arrays and large antennas) ***shall*** be subject to acoustic testing to determine survival during the launch/ascent phase acoustic environment. Components that have an area to mass ratio of 0.21 m2/kg (145 in2/lb) or greater ***should*** be considered for acoustic vibration testing.

Components ***shall*** be in the launch/ascent mechanical and electrical configuration and be subjected to the acoustic environment shown in Table 4.3.3.1-1 unless the launch vehicle specific acoustic spectrum is available.

Components ***shall*** be suspended or otherwise positioned within the acoustic chamber such that no major surfaces are parallel to the chamber walls, floor or ceiling, with a minimum of 0.6 m (2 ft) of clearance from any chamber surface.

Table 4.3.3.1-1
Acoustic Spectrum Levels

	Sound Pressure Level (dB ref $20x10^{-6}$ Pa)	
1/3 Octave Band Center Frequency (Hz)	Acceptance Duration = 1 minute (dB)	Qualification/Protoflight Duration = 1 minute (dB)
31.5	123.0	126.0
40	125.0	128.0
50	127.0	130.0
63	128.5	131.5
80	129.0	132.0
100	129.0	132.0
125	129.0	132.0
160	129.0	132.0
200	129.5	132.5
250	129.5	132.5
315	129.5	132.5
400	129.0	132.0
500	128.0	131.0
630	127.0	130.0
800	126.0	129.0
1000	125.0	128.0
1250	124.0	127.0
1600	122.5	125.5
2000	121.0	124.0
2500	119.5	122.5
3150	117.5	120.5
4000	115.5	118.5
5000	113.5	116.5
6300	111.5	114.5

8000	109.5	112.5
10000	107.5	110.5
Overall SPL	**140.4**	**143.4**

If the launch vehicle specific acoustic environment is available then testing ***shall*** be in accordance with paragraph 4.3.1.

4.3.3.2 Spacecraft Level

The project ***shall*** determine whether the area to mass ratio of the spacecraft is large enough to be excited by acoustic vibration testing based, at a minimum, on the criteria discussed in paragraph 4.3.3.1. In general, spacecraft are tested in an acoustic vibration environment and not a random vibration environment. The spacecraft ***shall*** be subjected to the acoustic environment defined by the launch vehicle provider in accordance with paragraph 4.3.1 and shown to perform within specification, both functionally and structurally.

4.3.4 Sine Vibration Test

Sine-sweep vibration tests are performed to qualify protoflight hardware for the low-frequency transient or sustained sine environments when they are present in flight for some launch vehicles, and to provide a workmanship test for all spacecraft hardware which is exposed to such environments and normally does not respond significantly to the vibroacoustic environment at applicable frequencies, such as wiring harnesses and stowed appendages. For most launch vehicles, the sine vibration environment is benign and no sinusoidal vibration test is required.

4.3.4.1 Component Level

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 spacecraft-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 applicable frequency range in each test axis.

Component levels depend on the type of structure to which the component 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 components, and mission-specific test levels ***shall*** be developed. Refer to GSFC General Environment Verification Specification, GSFC-STD-7000, paragraph 2.4.3.2b, for guidance in developing mission specific test levels.

A low-level sine sweep ***should*** 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.

4.3.4.2 Spacecraft Level

Test levels ***shall*** be developed on a mission-specific basis and are only necessary if required by the launch provider.

Sinusoidal vibration test levels required to simulate the flight environment 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 spacecraft based on the coupled loads analysis. Refer to GSFC General Environment Verification Specification, GSFC-STD-7000, paragraph 2.4.3.1b, for guidance in developing mission specific test levels.

Prior to the availability of coupled loads analysis results, preliminary sine test levels may be estimated by using the launch vehicle "user's guide" sine vibration levels with notching levels based on net loads equivalent to the user's guide c.g. load factor loads. Alternatively, spacecraft interface dynamic response data from flight measurements or coupled loads analysis for similar spacecraft may be used for the base drive input in conjunction with a suitable uncertainty factor.

4.3.5 Shock

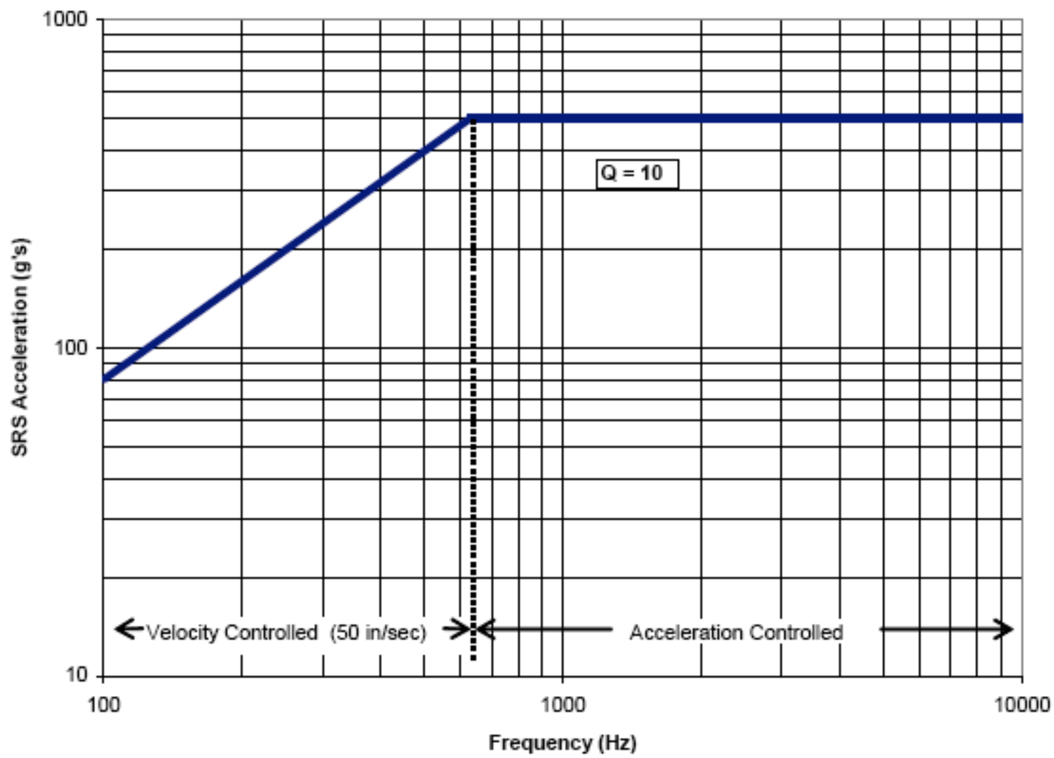
Both self-induced and externally induced shocks ***shall*** be considered in defining the mechanical shock environment.

Note: *NASA-STD-7003, Pyroshock Test Criteria,* ***should*** *be used as a guide in developing the pyroshock test approach.*

4.3.5.1 Component Level

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 component level.

The decision to perform component shock testing is typically based on an assessment of the shock susceptibility of the component and the expected shock levels. 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. For standard electronics, the potential for damage due to shock can be quantified based on Figure 4.3.5.1-1. If the flight shock environment as shown on a shock response spectrum plot is enveloped by the curve shown in Figure 4.3.5.1-1, 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. The curve provided in Figure 4.3.5.1-1 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.



Shock Response Spectrum for Assessing Component Test Requirements
Figure 4.3.5.1-1

If it is determined there is a need to test, then the test ***shall*** demonstrate that the component will perform within specification after being exposed to a representative separation environment. The shock environment for each component ***shall*** be determined by analysis or test of the spacecraft when subjected to the launch vehicle shock environment. The shock spectrum shape ***shall*** be applied in each of three orthogonal axes. At least 50% of spectrum amplitudes ***should*** exceed the nominal test specification. Components that are normally powered-on during spacecraft separation ***shall*** be shock tested in the powered-on state.

Any spacecraft self-induced shock events ***shall*** also be incorporated into the tests program.

4.3.5.2 Spacecraft Level

A separation system test ***shall*** be performed on the spacecraft in order to demonstrate that the spacecraft will perform within specification after being exposed to the separation environment. Functional equivalent flight-like pyrotechnic shock devices ***should*** be used for this test. A test with two pyro firings ***should*** be implemented. The shock environment may be simulated by test instead of using the actual flight pyrotechnic devices that is consistent with the level and duration specified in Table 4.3.1-1 above.

Any spacecraft self-induced shock events ***shall*** also be incorporated into the test program.

4.3.6 Pressure Decay/Venting

The need for a pressure profile test ***shall*** be assessed for all components and at the spacecraft level. A 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.

A vented area of ≥2 times the required area to accommodate the launch profile venting rate ***shall*** be provided.

The flight pressure profile ***shall*** be determined by the analytically predicted pressure-time history inside the payload fairing for the nominal launch trajectory for the mission (including reentry if appropriate).

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).

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

4.4 Electrical

4.4.1 Electrical Functional & Performance Tests

The following paragraphs describe the required electrical functional and performance tests that verify the spacecraft'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 Verification Plan.

4.4.1.1 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.

4.4.1.2 Comprehensive Performance Tests

A comprehensive performance test (CPT) ***shall*** be conducted on each hardware element after each stage of assembly: component, subsystem and system. 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 system level, the comprehensive performance test ***shall*** demonstrate that, with the application of known stimuli, the spacecraft 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.

4.4.1.3 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.

4.4.2 Electromagnetic Interference and Compatibility (EMI/EMC)

The general requirements for electromagnetic compatibility are as follows:

a. The spacecraft ***shall*** not generate electromagnetic interference that could adversely affect its own subsystems and components, other payloads, or the safety and operation of the launch vehicle and launch site.
b. The 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.

<u>Requirements Summary</u>
The levels ***should*** be tailored to meet mission specific requirements, such as, the enveloping of launch vehicle and launch site environments, or the inclusion of very sensitive detectors or instruments in the spacecraft. The launch vehicle provider ***should*** be consulted early in development to understand the EMI/EMC interface requirements. The radiated and conducted emissions requirements will vary from launch vehicle to launch vehicle and launch site to launch site. The requirements defined in this section are for use in the absence of launch vehicle and launch site specific requirements and ***should*** be used only until launch vehicle and launch site specific requirements are obtained.

For spacecraft that are unpowered during the launch phase it is likely that EMI testing will not be required except for self-compatibility testing to ensure the spacecraft is not effected by its own radiated and conducted emissions. Requirements for testing for self-compatibility are unique to each spacecraft and are unrelated to the launch vehicle and launch site.

Note: *The effects of electromagnetic and magnetic fields on instrument and attitude control system designs need to be considered. This could not only affect the design of components but also the placement of them on the spacecraft in order to minimize their effects.*

<u>The Range of Requirements</u>
Table 4.4.2-1 is a matrix of EMC tests that apply to a wide range of hardware. Tests are prescribed at the component, subsystem, and system levels of

assembly. Not all tests apply to all levels of assembly or to all types of spacecraft. The project must select the requirements that fit the characteristics of the mission and hardware, e.g. a transmitter would require a different group of EMC tests than a receiver. The EMC test program is meant to uncover workmanship defects and unit-to-unit variations in electromagnetic characteristics, as well as design flaws. The qualification, proto-qualification, and flight acceptance EMC programs are the same.

Table 4.4.2-1
EMI/EMC Requirements Per Level of Assembly

Type	Test	MIL-STD-461G Designator	Unit/Component	Subsystem/Instrument	Spacecraft/Payload
CE	DC power leads	CE101 CE102	R	R	
CE	Antenna terminals	CE106	R		
CS	Power line, 30Hz to 150kHz	CS101	R	R	
CS	Power Line, 150kHz to 200MHz	CS114	R	R	
CS	Intermodulation products	CS103	R		
CS	Signal rejection	CS104	R		
RE	Magnetic fields	RE101	R	R	R
RE	E-fields	RE102	R	R	R
RE	Transmitters	N/A	Note 1	Note 1	Note 1
RE	Spurious (transmitter antenna)	RE103		R	
RS	AC Magnetic field	RS101	R	R	R
RS	E-field	RS103	R	R	R

Note 1: *Must meet any unique requirements of launch vehicle and launch site for transmitters that are on during launch until separation from launch vehicle*

Testing at Lower Levels of Assembly

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

Basis of the Tests

The tests are based on the requirements of MIL-STD-461G, with GSFC-STD-7000 (GEVS) used as a supplemental guide. The tests and their limits are to be considered minimum requirements; however, they may be revised as appropriate for a particular spacecraft or mission. Additional EMC requirements may be placed on the spacecraft by the launch vehicle or launch site or in consideration of the mission launch radiation environment. Those requirements ***shall*** be established during coordination between the spacecraft project and the launch vehicle project/program office. More stringent requirements may be needed for spacecraft with very sensitive electric field or magnetic field measurement systems.

Safety and Controls

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

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

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

Special Considerations:

4.4.2.1 Conducted Susceptibility

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

b. For CS103 and CS104 tests on receivers operating in the frequency range of 30 Hz to 18 GHz, the operational frequency range of equipment subject to this test ***should*** be increased to 18 GHz and the highest frequency used in the test procedure ***should*** be increased to 40 GHz.

4.4.2.2 Radiated Emissions

a. Additional tests or test conditions ***should*** be considered by the project if it appears that this may be necessary, for example, if the spacecraft receives at frequencies other than S-band (1.77 - 2.3 GHz).

4.4.2.3 Radiated Susceptibility

Equipment that must operate at launch ***shall*** be tested, at minimum, to the following levels:
(1) 20 V/m from 2 MHz to 18 GHz

Equipment that will not be powered on during launch ***shall*** be tested, at minimum, to the following levels:
(2) 2 V/m from 2 MHz to 18 GHz

4.5 Thermal

An appropriate set of tests and analyses ***shall*** be selected to demonstrate the following spacecraft capabilities.

a. The spacecraft ***shall*** perform satisfactorily within the vacuum and thermal mission limits (including launch and return as applicable).
b. The thermal design and the thermal control system ***shall*** maintain the affected hardware within the established mission thermal limits during planned mission phases, including survival/safe-hold, if applicable.
c. The hardware ***shall*** withstand, as necessary, the temperature conditions of transportation, storage, launch, and flight.
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.

The verification approach ***shall*** be as defined in Table 4.5-1 below.

Thermal Verification Requirements
Table 4.5-1

Requirement	Unit/Component	Subsystem/Instrument	Spacecraft/Payload
Thermal Balance	A	A	T & A
Thermal Vacuum	T[1]		T
Leakage	T[2]	T[2]	T[2]

1 Temperature cycling at ambient pressure may be substituted for thermal vacuum testing if an analysis can show the component to be unaffected by a vacuum environment. This analysis must show that temperature levels and gradients are as severe in air as in a vacuum.

2 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.

4.5.1 Thermal Balance

Also known as Thermal Vacuum and Power Management (TVPM) represents “Test Like You Fly” philosophy to validate power and thermal modeling of spacecraft systems and components for the mission profile. It is a test to evaluate both power and thermal characteristics and predicted thermal and power margins. This test is ideal to run on a final flight design that encompasses final hardware, thermal, and astrodynamics design characteristics. TVPM should not be mistaken for a workmanship test such as Thermal Vacuum or Environmental Stress Screening (ESS) although in some rare cases it may be possible to combine the two tests.

TVPM ***shall*** be performed in a Thermal Vacuum Chamber under conditions $\leq 10^{-4}$ Torr.

***Note**: It is important that system is powered off while pumping down the chamber to achieve vacuum to avoid spacecraft arcing or multipaction within the spacecraft radio systems.*

The test is meant to simulate orbit by orbit thermal and power balance conditions.

TVPM verifies that the actual hardware dynamics of heat loading and shedding as well as the corresponding power draw and battery capacity depth of discharge are as predicted by modeling. Solar panel power generation is simulated by a power supply that follows sunlight and eclipse timing. The system ***should*** be run in the flight experiment phase protocol condition predicted for space operations. This includes turning radios on and off while simulating ground station TX/RX contacts timing.

***Note**: take appropriate precautions such as antenna attenuators/hats if necessary for test personnel safety and circuit amplifier protection.*

Actual measurements versus model predictions may indicate errors in build configuration or software simulations. This is a valuable finding in the TVPM test process and if found, ***should*** be iterated on until the simulation and hardware response correlate. If no error is found in the modeling and power balance with margin was not achieved, then a hardware design correction is indicated. This is achieved by way of changes to thermal path or adjusting surface coatings and lastly adjusting power generation sizing (i.e., solar panel).

TVPM modeling ***should*** take into account conditions from time of Spacecraft deployment, which includes tip off and stabilization conditions, through mission end of life.

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 ***shall*** be given for testing an “off-nominal” case such as a safe hold 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 spacecraft 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.

4.5.2 Thermal Vacuum/Cycling

The purpose of thermal vacuum testing is to detect defects in materials, processes, and workmanship by subjecting the unit under test to thermal cycling in a vacuum environment. Both stabilized and transitions conditions will be produced, which induces stresses intended to uncover incipient problems.

Thermal-Vacuum testing of the hardware ***should*** occur after dynamic testing has been completed. This helps to ensure that incipient failures, induced by transient dynamic tests, are discovered during later test phases.

4.5.2.1 Component Level

Components ***shall*** operate as specified during exposure to the thermal vacuum environment defined in Table 4.5.1.1-1. Testing ***shall*** be performed at $\leq 10^{-4}$ Torr and ***shall*** have an adequate conduction path to ensure proper cycling. During pump down, it ***shall*** be demonstrated that components generate no corona that degrades their performance below specified limits while operating, if applicable, in the launch mode at vacuum. Transitions between temperature extremes ***shall*** be at a minimum rate of 1°C per minute.

Components ***shall*** be powered and critical parameters monitored for proper operation during the test except for the "turn-on" tests. Components ***should*** be turned off during the high and low dwells until the temperature stabilizes, for at least 30 minutes, and then turned back on. Components ***shall*** exhibit normal turn-on characteristics. Components ***shall*** be functionally tested at high and low temperature extremes after temperature stabilization. The temperature ***shall*** be measured at a representative location such as component mounting feet on the base plate. Care ***should*** be taken when returning the chamber to ambient to ensure that no condensation forms on the test items.

Note: *"Dwell" is defined as time at plateau temperature to allow unit internal temperatures to equilibrate and Soak as the total time at plateau temperature.*

Table 4.5.1.1-1
Component Thermal Vacuum Testing

Parameter	Acceptance	Qualification/ Protoflight

Temperature Range	-24°C to +61°C or Max. Allowable Range ±5°C	-34°C to +71°C or Max. Allowable Range ±10°C
Number of Cycles	8	8
Min. Temp. Rate of Change	1°C/minute	1°C/minute
Soak Time at Hot/Cold	4 hours	4 hours

4.5.2.2 Spacecraft Level

The spacecraft level temperature test conditions ***shall*** be based on the worst-case predicted flight environment plus margin as shown in Table 4.5.1.2-1.

Table 4.5.1.2-1
Spacecraft Thermal Vacuum Testing

Parameter	Acceptance	Qualification/ Protoflight
Temperature Range	Max predicted range ±5°C	Max predicted range ±10°C
Number of Cycles	4	4
Min. Temp. Rate of Change	Max. predicted for mission	Max. predicted for mission
Soak Time at Hot/Cold	8 hours	8 hours

4.6 Contamination Control

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

4.6.1 Applicability

The contamination control program is applicable to all spacecraft hardware during all mission phases (fabrication, assembly, integration, testing, transport, storage, launch site, launch, and on-orbit). In the case of spacecraft which are not sensitive to contamination, this program may still be required if there is a risk of cross-contamination to other payloads.

4.6.2 Summary of Verification Process

The following are performed in order:

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

Each of the above activities ***shall*** be documented and submitted for concurrence/approval.

4.6.3 Contamination Sensitivity

An assessment ***shall*** be made by PDR to determine whether the possibility exists that the item will be unacceptably degraded by molecular or particulate contaminants, or is a source of contaminants. The assessment ***shall*** take into account all the various factors during the entire development and flight including identification of materials (including quantity and location), manufacturing processes, integration, test, packing and packaging, transportation, and mission operations including launch and return to earth, if applicable.

In addition, the assessment ***should*** identify the types of substances that may contaminate and cause unacceptable degradation of the test item. If the assessment indicates a likelihood that contamination will degrade performance, a contamination control program ***should*** be instituted. The degree of effort applied ***shall*** be in accordance with the importance of the item's function to mission success, its sensitivity to contamination, and the likelihood of its being contaminated.

4.6.4 Contamination Allowance

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

4.6.5 Contamination Budget

A contamination budget ***shall*** be developed for each critical item that describes the quantity of contaminant and the degradation that may be expected during the various phases of the lifetime of the item. The phases include the mission itself. The budget ***should*** be stated in terms (or units) that can be measured during testing. The measure of contamination ***should*** be monitored as the program progresses to include the contamination and degradation experienced. The budget ***should*** be monitored to ensure that, given the actual contamination, the mission performance will remain acceptable. A contamination-sensitive item may be cleaned periodically to reestablish a budget baseline. Contamination avoidance methods, such as clean-rooms and instrument covers, will affect the budget and a general description of their usage ***should*** be included. The contamination budget ***should*** be negotiated among the cognizant parties (e.g., the Project Scientist, the instrument contractor and the spacecraft contractor).

4.6.6 Contamination Control Plan

A contamination control plan ***shall*** be completed by PDR that describes the procedures that will be followed to control contamination and includes:

a. establishes the implementation and describe the methods that will be used to measure and maintain the levels of cleanliness required during each of the phases of the item's lifetime
b. specifically addresses outgassing requirements for the flight items, test chamber, and test support equipment and include a special section for lasers if incorporated into the design of the spacecraft.

4.6.7 Other Considerations

All nonmetallic materials ***should*** be selected for low out gassing characteristics consistent with those stipulated in ASTM E 595, JSC 0700 Vol. XIV, and NASA Reference Publication 1124. Relevant data can be found at the web site, http://outgassing.nasa.gov which is updated every three months.

Bake-outs of solar arrays, major wiring harnesses, and thermal blankets ***shall*** be performed unless it can be satisfactorily demonstrated that the contamination allowance can be met without bake-outs. Bake-outs of other components with large amounts of nonmetallic material, such as batteries, electronic boxes, and painted surfaces may also be necessary.

For information regarding outgassing testing refer to ASTM E 595, Test Method for Total Mass Loss and Collected Volatile Condensable Materials from Outgassing in a Vacuum Environment.

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

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

a. Non-flight GSE wiring and connector materials can contaminate the flight hardware during thermal testing.
b. Packaging material (plastic films and flexible foams) can contaminate hardware or cause corrosion during shipping and storage.
c. Plastic bags without anti-static properties can allow electrostatic discharges to damage electronic components on circuit boards.
d. Tygon tubing (or other non-flight tubing) used in purge systems can contaminate hardware or sensitive payloads when gasses or liquids extract plasticizers from the tubing.
e. Paints, sealants, and cleaning materials used to maintain clean rooms can contaminate or corrode flight hardware. To protect flight hardware, non-flight hardware that will be exposed to thermal vacuum testing with flight hardware (items such as cables, electronics, fixtures, etc.) ***should*** be fabricated from flight quality materials. Packaging materials ***should*** be tested to verify that they are non-corrosive, non-contaminating, and provide electrostatic protection, if required. All materials used in purge systems ***should*** be tested for cleanliness and compatibility with flight materials and/or payload as applicable.

5 VERIFICATION

Verification of many requirements of this document may be satisfied by project plans, procedures, design documents, drawings, analysis and test reports and should be referenced appropriately to reduce the need for generating unique verification closure reports (reference to appropriate sections of project documents in the verification compliance matrix is sufficient).

Compliance assessment is the responsibility of each project's Lead Systems Engineer. An assessment of compliance to the requirements in this document must be performed by the Lead Systems Engineer at the project's PDR, CDR, and Pre-Ship Review (PSR). In addition, ACE will conduct an independent assessment of compliance to the requirements during the CoFR process. The project ***shall*** also consult with the launch vehicle provider to determine the launch vehicle related verification products and schedule.

A Verification Matrix, Table 5-1, for the requirements in Sections 3 and 4 of this document ***shall*** be completed or incorporated into an overall project verification matrix. In many cases, multiple "*shall*" statements exist within one paragraph. The project may identify verification products by paragraph number and do not need to necessarily break out each "*shall*" statement to an individual verification line-item. This may be done at the discretion of the project.

It is important to establish verification requirements (method and approach for the method) early in the project development cycle taking into account project's mission classification so that the project's cost and schedule can be adequately determined. There are several cost and schedule items driven by the verification method chosen including the cost of testing, the cost of developing or procuring special test equipment and ground support equipment, and the time required to design, build, or procure these items.

5.1 Requirements Applicability

The first step in the verification process is to determine whether the requirements of this document are applicable or not. There is a third category reserved for requirements that are applicable but only in a modified form of the original requirement. For these exceptions to requirements the project must also provide appropriate rationale for the requested exception.

A	Applicable
N/A	Not Applicable
E	Exception (a modified version of the requirement)

5.2 Verification Method

For those requirements that are either applicable, or applicable with an exception, the project must determine an appropriate verification method such as Test, Analysis, Demonstration or Inspection.

Table 5-1
Verification Matrix

Requirements					Verification					
Par. No.	Paragraph Title	A N/A or E	Not Applicable/Exception Rationale or Special Comments		Method T-Test A-Analysis I-Inspection D-Demonstration				Compliant (yes, no, or N/A)	Documentation Reference
					T	A	I	D		

6 LIST OF PLANS

Table 6.1 below identifies the plans required to be developed by this document and the milestone they are required by.

Table 6.1

No.	Title	Milestone
1	EEE Part Control Plan	PDR
2	Integration and Test Plan	CDR
3	Contamination Control Plan	PDR

APPENDIX A. DEFINITIONS

Abbreviated Functional Test	A shortened version of the baseline functional test. It is a series of test operations that exercise the critical functional performance parameters of the component under test. This test is designed to verify that a component, subsystem, or system is operating in accordance with established parameters. The abbreviated functional test will be described specifically in the functional test procedure for each component.
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.
Functional Test	Functional Tests are a series of tests that exercise the functional of the unit under test. It is a series of test operations that exercise the critical functional performance parameters of the component under test. This test is designed to verify that a component, subsystem, or system is operating in accordance with established parameters. The functional test parameters will be described in a functional test procedure for each component, subsystem, or system (spacecraft).

Limit Load	The maximum anticipated load experienced by a structure during a loading event, load regime, or mission. Uncertainty factors associated with model uncertainty or forcing function uncertainty shall be incorporated into the limit load as reported. The factors of safety are not included in the limit load
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.
Performance Test	A performance test evaluates the performance of the system relative to the documented performance levels established in the project specific requirements documents.
Physical Tests	Physical tests are tests such as weight (mass), center of gravity, moment of inertia, envelope, physical attach interfaces, and other specified physical features of the component.
Protoflight Unit	Hardware of a new design that is used to qualify the design and is acceptable for flight. Typically, the proto-flight test levels and durations are between what

	a separate qualification unit is subjected to and what flight acceptance test is subjected to. Proto-flight units are sometimes referred to as "proto-qualification units".
Spacecraft	See Payload. Other terms used to designate this level of assembly are Laboratory, Observatory, and satellite.
Spacecraft Bus	See module.
Subassembly	A subdivision of an assembly. Examples are wire harness and loaded printed circuit boards.
Subsystem	A functional subdivision of a payload consisting of two or more components. Examples are structural, attitude control, electrical power, and communication subsystems. Also included as subsystems of the payload are the science instruments or experiments.
System	A composite of hardware and software capable of performing an operational role in its intended environment. It is a comprised of subsystems and components that together meet a specific set of mission objectives.
Temperature Stability	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.

Hierarchical Levels and Examples

Level Name	Examples

System	Spacecraft = instruments + structure + electrical power + C&DH + thermal control + ...
Subsystem	instruments, electrical power, thermal control, propulsion, C&DH, attitude control, structure
Component	electronics box, motor, actuator, battery, receiver, transmitter, antenna, solar panel, valve regulator
Part	solar cell, switch, connector, capacitor, resistor, IC, bolt, gasket, valve stem

APPENDIX B. ACRONYMS

ACS	Attitude Control System
ADCS	Attitude Determination and Control System
AIAA	American Institute of Aeronautics and Astronautics
ANSI	American National Standards Institute
APR	Ames Procedural Requirement
ARC	Ames Research Center
ASD	Acceleration Spectral Density
ASTM	American Society for Testing and Materials
CBE	Current Best Estimate
CCP	Contamination Control Plan
CDR	Critical Design Review
CLA	Couple Loads Analysis
CoFR	Certificate of Flight Readiness
COPV	Carbon Overwrapped Pressure vessel
CPU	Computer Processor Unit
CPT	Comprehensive Performance Test
CRES	Corrosion Resistant
DOD	Department of Defense
EEE	Electrical, Electronic, and Electromechanical
EED	Electro Explosive Device
EEPROM	Electrically Erasable Programmable Read Only Memory
ESD	Electro-Static Discharge
GEVS	General Environmental Verification Standard
GSFC	Goddard Space Flight Center
GSE	Ground Support Equipment
HEXFET	Hexagonal Field Effect Transistor
JPL	Jet Propulsion Laboratory
LPT	Limited Performance Test
NDE	Non-Destructive Evaluation
NTE	Not to Exceed
PDR	Preliminary Design Review
PEM	Plastic Encapsulated Microcircuits
PF	Protoflight
PROM	Programmable Read Only Memory
PSD	Power Spectral Density
RAM	Random Access Memory
RF	Radio Frequency
SEE	Single Event Effect
SEL	Single Event Latchup
SEU	Single Event Upset
SRR	System Requirements Review
SRS	Software Requirements Specification

TMR	Tripple Module Redundancy
VDD	Version Description Document

APPENDIX C. REQUIREMENTS TRACE MATRIX

Paragraph	Title	Trace
3	**FLIGHT SYSTEM DESIGN**	
3.1	Design Margins and Fault Tolerance	
3.1.1	Mechanical	
3.1.1.1	Mass	JPL Design Principles Par. 6.3.2.3 Gold Rules Table 1.06-1
3.1.1.2	Deployment Systems	JPL Design Principles Par. 4.2.3.1 Gold Rules 4.15
3.1.1.3	Actuator Design	JPL Design Principles Par. 4.2.3.3
3.1.1.4	Stroke for Linear Actuators	JPL Design Principles Par. 4.2.3.6
3.1.1.5	Mechanism Cycle Life	reduced JPL Design Principles Par. 4.2.3.9 Gold Rules 4.23
3.1.1.6	Enclosed Volume	JPL D-26086 rev. D
3.1.2	Electrical	
3.1.2.1	Power	JPL Design Principles Par. 6.3.3.3 Gold Rules 1.06-1
3.1.2.2	Depth of Discharge	ARC Best Practices, JPL Design Principles Par. 4.3.3.6.3
3.1.2.3	Power Distribution Circuit Margin	JPL Design Principles Par. 6.3.6.1
3.1.2.4	Flight Electronics Hardware Margins	JPL Design Principles Par. 6.3.8
3.1.2.5	Pyrotechnic Systems	JPL Design Principles Par. 6.3.6
3.1.2.5.1	Pyrotechnic Circuit Margin	JPL Design Principles Par. 6.3.6.1
3.1.2.5.2	Pyrotechnic Circuit Fault Protection	ARC Best Practices
3.1.3	Thermal	
3.1.3.1	System Level Temperature Margin	Gold Rules 4.25
3.1.3.2	Component Level Temperature Margin	Gold Rules 4.27

3.1.4	Propulsion	
3.1.4.1	Liquid Propellant Design Criteria	NASA-STD-5012
3.1.4.2	Propellant Volume	Gold Rules 1.06
3.1.4.3	Propellant Freezing	JPL Design Principles Par. 4.8.2.6
3.1.4.4	Propellant Condensation	JPL Design Principles Par. 4.8.2.7
3.1.4.5	Cryogenic Design Margin	JPL Design Principles Par. 4.8.2.11
3.1.4.6	Component Cycle Life	JPL Design Principles Par. 4.7.3.1
3.1.4.7	Attitude Control System	ARC Best Practice
3.1.5	Attitude Determination and Control System (ADCS)	
3.1.5.1	Controller Stability Margins	Gold Rules 1.30, JPL Design Principles Par. 4.6.1.2
3.1.5.2	Actuator Sizing Margins	Gold Rules 1.31
3.1.5.3	Flexible Body Systems	Gold Rules 1.30
3.1.5.4	Passive Attitude Control System	ARC Best Practice
3.1.6	Telemetry & Command	
3.1.6.1	Telemetry & Command Hardware Data Channels and RF Link	Gold Rules 1.06-1
3.1.7	Flight Software and Computing System	
3.1.7.1	Use of Analysis in Lieu of Measurement	ARC Best Practice
3.1.7.2	Flight System Computing Resource Margin	Gold Rules 3.07 & Table 3.07-1, JPL Design Principles Par. 6.3.5
3.1.8	Safety	
3.1.8.1	Catastrophic Hazards	Gold Rules 1.26
3.1.8.2	Critical Hazards	Gold Rules 1.26
3.2	Mechanical	
3.2.1	Structural and Mechanical	Gold Rules 4.14
3.2.1.1	Loads	Derived
3.2.1.2	Stiffness Requirements	

3.2.1.2.1	Primary Structure	Derived based on Pegasus, Minotaur, and Atlas payload User's Guides
3.2.1.2.2	Secondary Structure & Components	ARC Best Practice
3.2.1.3	Factors of Safety	NASA-STD-5001, Gold Rules 4.03
3.2.1.3.1	Joint slip factor of safety	JPL Design Principles Par. 4.2.5.3
3.2.1.3.2	Fitting factor of safety	JPL Design Principles Par. 4.2.5.4
3.2.1.3.3	Thermally-induced loading factor of safety	JPL Design Principles Par. 4.2.5.5
3.2.1.4	Proof testing Non-Metallic Structures	NASA-STD-5001, Gold Rules 4.12
3.2.2	Fasteners & Pins	
3.2.2.1	Torque Limits and Preload	NASA-STD-5020
3.2.2.2	Locking Requirements	Gold Rules 4.20, NASA-STD-5020
3.2.2.3	Lubrication	NASA-STD-5020
3.2.2.4	Fastener Materials	ARC Best Practice, NASA-STD-5020, NASA-STD-6016
3.2.2.5	Flat Head Screws	ARC Best Practice
3.2.2.6	Use of Alignment Pins	ARC Best Practice
3.2.2.7	Use of Double Eccentric Fasteners	ARC Best Practice
3.2.3	Mechanisms	NASA-STD-5017
3.2.3.1	Spring Energized Bolt Release for Pyrotechnic Separation Nuts	JPL Design Principles Par. 4.2.3.8
3.2.3.2	Explosive Ordnances	DOD-E-83578
3.2.4	Materials	
3.2.4.1	Environmental Effects on Material Selection	ARC Best Practice
3.2.4.2	Metallic Materials	ARC Best Practice, NASA-STD-6016
3.2.4.3	Composite Materials	ARC Best Practice, CMH-17
3.2.4.4	Galvanic Corrosion	ARC Best Practice, MIL-STD-889
3.2.4.5	Stress Corrosion Cracking	ARC Best Practice, MIL-STD-1568
3.2.5	Fluids Systems	
3.2.5.1	Materials Compatibility	ARC Best Practice

3.2.6	Mechanical Integration	
3.2.6.1	Component Precision Location	Gold Rules 4.22
3.2.6.2	Sensor and Antenna Blockage	Gold Rules 1.37
3.3	Electrical	GSFC 500-PG-8700.2.2
3.3.1	General	
3.3.1.1	Minimum Operating Time	
3.3.1.1.1	Prior to Spacecraft Integration	ARC Best Practice
3.3.1.1.2	Integrated Spacecraft	ARC Best Practice, JPL Design Principles Par. 8.1.1, Gold Rules 2.01
3.3.1.2	Printed Wiring Boards	ARC Best Practice
3.3.1.3	Electrostatic Discharge (ESD) Control	ANSI/ESD S20.20, ARC Best Practice
3.3.1.4	RF Component Immunity to Multipaction	Gold Rules 5.04
3.3.1.5	Voltage/Temperature Margin Test	ARC Best Practice
3.3.1.6	Polarity Checks of Critical Components	Gold Rules 1.33
3.3.2	EEE Parts	
3.3.2.1	EEE Parts Control	APR 8730.2
3.3.3	Digital Design	
3.3.3.1	Synchronous Designs	JPL Design Principles Par. 4.12.3.1
3.3.3.2	ASIC/FPGA Synchronous Designs	ARC Best Practice, GSFC 500-PG-8700.2.7
3.3.4	Use of Plastic Encapsulated Microcircuits (PEMs)	ARC Best Practice
3.3.5	Power Systems Design	
3.3.5.1	Wiring Design	ARC Best Practice, EEE-INST-002
3.3.5.2	Power Quality	AIAA S-122
3.3.5.3	Solar Array	ARC Best Practice, Gold Rules 2.24
3.3.5.4	Batteries	JPL Design Principles Par. 4.3.3.6
3.3.6	Power Converters and Supply	
3.3.6.1	Power Converter Synchronization	JPL Design Principles Par. 4.12.4.1
3.3.6.2	Power Cycle Capability	ARC Best Practice
3.3.6.3	Power Supply Transient Analysis	ARC Best Practice

3.3.6.4	Power System Charging	NASA-STD-4005, NASA-HDBK-4006
3.3.7	Circuit Protection	ARC Best Practice, JPL Design Principles Par. 4.12.1.8
3.3.8	Grounding, Bonding, and Isolation	
3.3.8.1	Local Single Point Ground	ARC Best Practice
3.3.8.2	Primary Circuit Return Path	JPL Design Principles Par. 4.10.4.1
3.3.8.3	Bonding	ARC Best Practice
3.3.8.4	Electrical Isolation	MIL-STD-1542 Par. 5.1.3.2
3.3.9	Connectors	
3.3.9.1	Inadvertent Mating	Gold Rules 2.13, JPL Design Principles Par. 4.12.5.4
3.3.9.2	Powered Connections	ARC Best Practice
3.3.9.3	Blind Mate Connection	ARC Best Practice, Gold Rules 2.13
3.3.9.4	Connector Pin Populatioin	ARC Best Practice
3.3.10	Radiation Tolerance	JPL Design Principles Par. 4.12.1.5
3.3.10.1	Single Event Effect (SEE)	JPL Design Principles Par. 4.12.1.5, ARC Best Practice
3.3.10.2	Total Ionizing Dose (TID)	Narrative
3.3.10.2.1	General Shielding	JPL Design Principles 4.12.1.5.1
3.3.10.2.2	Spot Shielding	JPL Design Principles 4.12.1.5.2
3.3.11	Pyrotechnic Functions	
3.3.11.1	Enabling of Pyrotechnic Functions	JPL Design Principles Par. 4.3.4.3
3.4	Thermal	
3.4.1	General Design Approach	
3.4.1.1	Design Tailored to Specific Application	JPL Design Principles Par. 4.8.1.1
3.4.1.2	Passive vs. Active Thermal Control Systems	ARC Best Practice
3.4.1.3	Motor and Actuator Self-Heating	JPL Design Principles Par. 4.8.2.12
3.4.1.4	Thermal Coatings Properties	Gold Rules 4.06
3.4.2	Use of Heaters	ARC Best Practice

3.4.2.1	Power Density Limit for Film Heaters	JPL Design Principles Par. 4.8.2.9
3.4.2.2	Maximum Duty Cycle of Heaters	JPL Design Principles Par. 4.8.2.10, Gold Rules 4.25, ARC Best Practice
3.4.3	Thermal Environments	
3.4.3.1	Thermal Environment for Earth Orbiting Missions	
3.4.3.1.1	Direct Solar Radiation	Spacecraft Thermal Control Handbook
3.4.3.1.2	Earth Albedo	Spacecraft Thermal Control Handbook
3.4.3.1.3	Earth Infrared Radiation	Spacecraft Thermal Control Handbook
3.4.3.2	Thermal Environment for Interplanetary Missions	Spacecraft Thermal Control Handbook
3.5	Propulsion	
3.5.1	Design, Test, and Analysis Requirements	
3.5.1.1	Fuses for Propulsion System	Gold Rules 1.27
3.5.1.2	Plume Impingement Analysis	Gold Rules 1.32
3.5.2	Sizing	
3.5.2.1	Propellant Tanks	JPL Design Principles Par. 4.7.2.1
3.5.2.2	Propellant Quantity	JPL Design Principles Par. 4.7.2.2
3.5.3	Pressurized Components	
3.5.3.1	General Design and Test Requirements	JPL Design Principles Par. 4.7.1.1
3.5.3.2	Tubing	ARC Best Practice
3.5.3.3	Joints	Gold Rules 1.20
3.5.3.4	Access for Cleaning and Testing	ARC Best Practice
3.5.3.5	Separable Fittings	ARC Best Practice
3.5.3.6	Pressure Surge Prevention (Liquid Systems)	Gold Rules 1.21
3.5.4	Propulsion Safety	
3.5.4.1	Fiber-Reinforced Composite Over-wrapped Pressure Vessels (COPV)	JPL Design Principles Par. 4.7.4.1

3.5.4.2	Use of Passive Isolation in Bi-Propellant Systems	JPL Design Principles Par. 4.7.4.2
3.5.4.3	Use of Gas Regulators	JPL Design Principles Par. 4.7.4.3
3.5.4.4	Propulsion Ignition	Gold Rules 1.28
3.5.4.5	Residual Test Fluids	Gold Rules 1.22
3.5.4.6	Propulsion System Safety Electrical Disconnect	Gold Rules 1.24
3.6	Attitude Determination and Control System (ADCS) Design	
3.6.1	Control Authority	ARC Best Practice
3.6.2	Sampled Control System Timing	JPL Design Principles Par. 4.6.1.4
3.7	Telemetry and Command	
3.7.1	Mission Critical Telemetry and Commands	Gold Rules 1.14
3.7.2	Spacecraft State Management	ARC Best Practice
3.7.2.1	Explicit Commanding of States	JPL Design Principles Par. 4.4.4, ARC Best Practice
3.7.2.2	Command Logging	ARC Best Practice
3.7.2.3	Critical Command Locking	ARC Best Practice
3.7.2.4	Power-On Reset (POR) State	JPL Design Principles Par. 4.12.7.1
3.7.2.5	Power-On Reset (POR) State Visibility	JPL Design Principles Par. 4.12.7.2
3.7.2.6	Visibility of Spacecraft State	JPL Design Principles Par. 4.4.6.4
3.7.2.7	Visibility of Health Status	JPL Design Principles Par. 4.4.6.1
3.7.2.8	Visibility for Anomaly Determination and Reconstruction	JPL Design Principles Par. 4.4.6.1
3.7.2.9	Visibility of Mission-Unique Functions	JPL Design Principles Par. 4.4.6.1, 4.4.6.4, 4.4.6.8, 4.4.6.10
3.8	Software and Information Systems	
3.8.1	NASA Software Engineering Requirements	APR 7150.2
3.8.2	Flight Software Design	
3.8.2.1	Software Modifiability	ARC Best Practice

3.8.2.2	Protection from Unintended Software Modification	JPL Design Principles Par. 4.11.4.3
3.8.2.3	Compatibility with COTS Tools	JPL Design Principles Par. 4.11.1.4
3.8.2.4	Start-Up Response	JPL Design Principles Par. 4.11.2.1, ARC Best Practice
3.8.2.5	Software Design Robustness	JPL Design Principles Par. 4.11.4
3.8.2.5.1	Command Validation and Acknowledgement	JPL Design Principles Par. 4.11.4.2
3.8.2.5.2	Detection and Response to Radiation Events	ARC Best Practice
3.8.2.5.3	Predictable Behavior When Stressed	JPL Design Principles Par. 4.11.5.4
3.8.2.5.4	Response to Resource Over-Subscription	JPL Design Principles Par. 4.11.4.5
3.8.2.5.5	Response to Missing Inputs	JPL Design Principles Par. 4.11.4.6
3.8.2.5.6	Response to Failed I/O	JPL Design Principles Par. 4.11.4.7
3.8.2.5.7	Response to Nominal and Off-Nominal Inputs	JPL Design Principles Par. 4.11.4.6
3.8.2.6	Protection against Incorrect Memory Use	JPL Design Principles Par. 4.11.4.11
3.8.2.6.1	Data Set Consistency	JPL Design Principles Par. 4.11.4.12
3.8.2.6.2	Self-Test Capability and Fault Diagnostic	JPL Design Principles 4.11.6.1, 4.11.6.2
3.8.2.6.3	Measurement of Constrained Resources	JPL Design Principles 4.11.6.3
3.8.3	On-Board Data Management	
3.8.3.1	Protection of Critical Data	JPL Design Principles Par. 4.4.3.1, 4.4.3.4
3.8.3.2	Redundant Handling of Critical Data	JPL Design Principles Par. 4.4.1.1
3.8.3.3	Compatibility with Tracking Outages	JPL Design Principles Par. 4.4.3.2
3.8.3.4	Multiple Restart Flight Software Initialization	JPL Design Principles Par. 4.11.2.2
3.9	Fault Management	
3.9.1	General	NASA-HDBK-1002
3.9.1.1	Management of Credible Single Faults	JPL Design Principles Par. 4.9.1.2

3.9.1.2	Fault Protection Response During Time-Critical Mission Activities	JPL Design Principles Par. 4.9.2.3
3.9.1.3	Fault Protection Response During Non-Time-Critical Mission Activities	JPL Design Principles Par. 4.9.2.2 Gold Rules 1.17
3.9.1.4	In-Flight Ability to Command and Parameter Visibility	JPL Design Principles Par. 4.9.3.1
3.9.1.5	Fault Indication Filtering	JPL Design Principles Par. 4.9.4.1
3.10	Ground Support Equipment	
3.10.1	General	
3.10.1.1	Acceptability for Interface to Flight Hardware	ARC Best Practice
3.10.1.2	GSE Testing	ARC Best Practice
3.10.1.3	GSE Use at Launch Site	Gold Rules 1.41
3.10.1.4	Electrical Fault Propagation	ARC Best Practice
3.10.1.5	GSE Software	ARC Best Practice
3.10.1.6	Fluid System GSE Fault Tolerance	Gold Rules 5.05
3.10.1.7	Connector Savers	ARC Best Practice
3.11	Parts, Materials, and Processes	
3.11.1	General	ARC Best Practice
3.11.2	Parts, Materials, and Processes List	ARC Best Practice
3.11.3	Limited Shelf-Life Materials	
3.11.3.1	Limited Life Item Tracking	ARC Best Practice
3.11.3.2	Limited Life Item Marking	ARC Best Practice
3.11.4	Workmanship	Listed NASA and IPC Standards
3.11.5	Parts & Material Traceability	ARC Best Practice
3.11.6	Outgassing	ARC Best Practice
3.11.7	Mechanical Parts Selection	ARC Best Practice
3.11.8	Finishes	ARC Best Practice
3.11.9	Prohibited Materials	NASA-STD-6016
3.11.10	Dissimilar Materials	ARC Best Practice, MIL-STD-889

4	**INTEGRATION AND ENVIRONMENTAL TEST**	
4.1	Integration and Test Plan	ARC Best Practice , Gold Rules 1.09, NASA-STD-7002
4.1.1	System Level Functional & Performance Verification	
4.1.1.1	Release Mechanisms for Flight Deployables	Gold Rules 1.47, ARC Best Practice
4.1.1.2	End-to-End Data System Testing	Gold Rules 1.08 & 3.03
4.1.1.3	Transmitter RF Power Output Testing	ARC Best Practice
4.1.1.4	ADCS Sensors and Actuators	Gold Rules 1.07, JPL Design Principles Par. 8.3.3.5
4.1.1.5	Mechanical Clearances	Gold Rules 4.24, JPL Design Principles Par. 8.3.3.3
4.1.1.6	Software Regression Testing	JPL Design Principles Par. 8.3.2.2
4.1.2	General I&T Requirements	
4.1.2.1	Capping of Test Points and Plugs	Gold Rules 2.14
4.1.2.2	Non-Condensing Environment	ARC Best Practice
4.1.3	Qualification of Heritage Hardware	Gold Rules 1.11
4.2	Environmental Verification	
4.2.1	Level of Assembly Test Requirements	ARC Best Practice, GEVS
4.2.2	Test Sequence	ARC Best Practice
4.2.3	Test Tolerances	GEVS 1.13
4.2.4	Test Article Types	ARC Best Practice
4.2.4.1	Qualification Testing	Narrative
4.2.4.2	Protoflight Qualification Testing	Narrative
4.2.4.3	Acceptance Testing	Narrative
4.3	Structural and Mechanical	Gold Rules 4.11, 4.14
4.3.1	Test Factors and Durations	GEVS Table 2.2-2
4.3.2	Random Vibration	NASA-STD-7001

4.3.2.1	Component Level	Gold Rules 4.10 GEVS Table 2.4-3
4.3.2.2	Spacecraft Level	GEVS Table 2.2-2 and Section 2.4.2
4.3.3	Acoustic Vibration	NASA-STD-7001
4.3.3.1	Component Level	ARC Best Practice, (envelopes values shown in user's guides for Pegasus, Minotaur, & Atlas launch vehicles), spacecraft and launch vehicle specific
4.3.3.2	Spacecraft Level	Spacecraft and launch vehicle specific
4.3.4	Sine Vibration Test	GEVS 2.4.3
4.3.4.1	Component Level	GEVS 2.4.3, Launch vehicle specific
4.3.4.2	Spacecraft Level	GEVS 2.4.3, Launch vehicle specific
4.3.5	Shock	GEVS 2.4.4, NASA-STD-7003
4.3.5.1	Component Level	GEVS Figure 2.4-1 and launch vehicle specific
4.3.5.2	Spacecraft Level	GEVS 2.4.4.2, Launch vehicle specific
4.3.6	Pressure Decay/Venting	GEVS 2.4.6
4.4	Electrical	
4.4.1	Electrical Functional & Performance Tests	GEVS 2.3
4.4.1.1	Electrical Interface Tests	GEVS 2.3.1
4.4.1.2	Comprehensive Performance Tests	GEVS 2.3.2
4.4.1.3	Limited Performance Tests	GEVS 2.3.3
4.4.2	Electromagnetic Interference and Compatibility (EMI/EMC)	GEVS 2.5, MIL-STD-461G
4.4.2.1	Conducted Susceptibility	GEVS 2.5.2.2, MIL-STD-461G
4.4.2.2	Radiated Emissions	GEVS 2.5.2.3, MIL-STD-461G
4.4.2.3	Radiated Susceptibility	GEVS 2.5.2.4, MIL-STD-461G
4.5	Thermal	GEVS 2.6
4.5.1	Thermal Balance	ARC Best Practice
4.5.2	Thermal Vacuum/Cycling	ARC Best Practice

4.5.2.1	Component Level	Gold Rules 4.27 & 4.29, GEVS 2.6
4.5.2.2	Spacecraft Level	Gold Rules 4.27 & 4.29, GEVS 2.6
4.6	Contamination Control	GEVS 2.7.1
4.6.1	Applicability	GEVS 2.7.1
4.6.2	Summary of Verification Process	GEVS 2.7.1.1
4.6.3	Contamination Sensitivity	GEVS 2.7.1.3
4.6.4	Contamination Allowance	GEVS 2.7.1.4
4.6.5	Contamination Budget	GEVS 2.7.1.5
4.6.6	Contamination Control Plan	Gold Rules 4.01, 5.08, GEVS 2.7.1.6
4.6.7	Other Considerations	GEVS 2.7.1.7
5	VERIFICATION	ARC Best Practice
5.1	Requirements Applicability	
5.2	Verification Method	
6	List of Plans	