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TEST REQUIREMENTS FOR SPACE VEHICLES

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Test Requirements for Space Vehicles

MIL-STD-1540B (USAF)

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by using the self-addressed Standardization Document Improvement Proposal (DD Form 1426) appearing at the end of this document or by letter.

FOREWORD

All space vehicles are subjected to extensive ground testing to ensure their successful operational use. This standard establishes a uniform set of definitions and ground testing requirements for space vehicles, whether launched by an expendable launch vehicle or the recoverable Space Shuttle. Although this standard is not applicable to launch vehicles, it is recommended that the component test requirements be used for launch vehicle components to obtain the required high reliability. The test requirements specified are a composite of those tests currently used in achieving successful space missions. It is intended that these test requirements should be tailored to the specific space program or project considering design complexity, state of the art, mission criticality and acceptable risk.

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SECTION 1

SCOPE

1.1 PURPOSE

This standard establishes uniform definitions, environmental criteria, test requirements, and test methods for space vehicles and their subsystems and components.

1.2 APPLICATION

The tailored application of these test requirements to a particular space program is intended to assure a high level of confidence in achieving a successful space mission. This standard is intended for use in the procurement of space vehicle hardware, including space vehicles and airborne support equipment that remain in the Space Shuttle Orbiter during orbital flight, as well as orbital satellites.

1.3 CLASSIFICATIONS

The tests specified herein are classified as development, qualification, acceptance, prelaunch validation, or other tests.

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SECTION 2

REFERENCED DOCUMENTS

2.1 ISSUES OF DOCUMENTS

The following documents of the issue in effect on the date of invitation for bids or request for proposal form a part of this standard to the extent specified herein.

Military Standards

MIL-STD-1541(USAF)	Electromagnetic Compatibility Requirements for Space Systems

(Copies of specifications, standards, and publications required by contractors in connection with specific procurement functions should be obtained from the contracting officer or as directed by the contracting officer.)

2.2 ORDER OF PRECEDENCE

In the event of conflict between documents referenced herein and the contents of this standard, the contents of this standard shall be considered the superseding requirement.

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SECTION 3

DEFINITIONS

3.1 ACCEPTANCE TESTS

Acceptance tests are the required formal tests conducted to demonstrate acceptability of an item for delivery. They are intended to demonstrate performance to specification requirements and to act as quality control screens to detect deficiencies of workmanship, material, and quality.

3.2 AIRBORNE SUPPORT EQUIPMENT (ASE)

Airborne support equipment is the equipment installed in a recoverable launch vehicle to provide support functions and interfaces for the space vehicle during launch and orbital operations of the recoverable launch vehicle. This includes the hardware and software which provides the structural, electrical, electronic, and mechanical interfaces with the launch vehicle. ASE is recovered with the launch vehicle.

3.3 AMBIENT ENVIRONMENT

Ambient environment is defined as normal room conditions with temperature at 23 ± 10 deg C (73 deg ± 18 deg F), atmospheric pressure 101 + 2.0/-23 kilopascals (29.9 + 0.6/-6.8 in. Hg) and relative humidity of 50 ± 30 percent.

3.4 BURST PRESSURE

The burst pressure is the maximum test pressure that pressurized components withstand without rupture to demonstrate the adequacy of the design in a qualification test. It is equal to the product of the maximum expected operating pressure, burst pressure design factor, and a factor corresponding to the differences in material properties between test and design temperatures.

3.5 COMPONENT

A component is a functional unit that is viewed as an entity for purposes of analysis, manufacturing, maintenance, or record keeping. Examples are hydraulic actuators, valves, batteries, electrical harnesses, and individual electronic boxes such as transmitters, receivers, or multiplexers.

3.6 COMPUTER PROGRAM COMPONENT (CPC)

A computer program component (CPC) is a functionally or logically distinct part of a computer program configuration item (CPCI) that is distinguished for purpose of convenience in designing and specifying a complex CPCI as an assembly of subordinate elements.

3.7 DECIBEL (dB)

Amplitude and power ratios may be expressed in decibels (dB). Ten times the logarithm to the base 10 of the ratio of one power level to a reference power level is the value of the power, or power ratio, in dB. For pressures, electrical currents, voltages, sinusoidal vibration, and other parameters, the amplitude rather than power is frequently measured. Twenty times the logarithm to the base 10 of the ratio of one amplitude measurement to a reference amplitude is the value of the power ratio in dB. The referenced amplitude or referenced power level must be clear. For sound pressure levels the amplitude of the reference pressure is 0.0002 dynes per square centimeter.

3.8 DESIGN ENVIRONMENTS, SPACE VEHICLE

The design environments for a space vehicle are the composite of the various environmental stresses to which the space vehicle must be designed. Each of the design environments for a space vehicle is based upon:

a. The maximum and minimum predicted environments during the operational life of the space vehicle, plus

b. An environmental design margin (see 3.12) that increases the environmental range to provide an acceptable level of confidence that a failure will not occur during the service life of the space vehicle.

3.9 DESIGN ENVIRONMENTS, SPACE VEHICLE COMPONENTS

The design environments for space vehicle components are the composite of the various environmental stresses to which the space vehicle hardware components must be designed. Each of the design environments for a space vehicle component is based upon:

a. The maximum and minimum predicted environments during the operational life of the component, or for temperature, a standard thermal range between

-24 deg C and +61 deg C when the predicted range is less severe, plus

b. An environmental design margin (see 3.12) that increases the environmental range to provide an acceptable level of confidence that a failure will not occur during the service life of the component (see 3.37).

3.10 DEVELOPMENT TESTS

Development tests include all tests conducted to obtain information to aid in the design and manufacturing processes. Development tests are conducted to generate design parameters, validate design concepts, verify design criteria, determine design margins, identify failure modes, and to verify manufacturing processes. Development testing may be informal in that controlled design and test documentation, formal certification, formal retest requirements, and flight type hardware are usually not required.

3.11 DEVELOPMENT TEST VEHICLE (DTV)

A development test vehicle (DTV) is a space vehicle or subsystem fabricated to provide engineering design and test information concerning the validity of analytic techniques and assumed design parameters, to uncover unexpected system response characteristics, to evaluate design changes, to determine interface compatibility, to prove test procedures and techniques, and to determine if the equipment meets its performance specifications. DTVs include engineering test models, thermal models, and structural models.

3.12 ENVIRONMENTAL DESIGN MARGIN

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

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

b. To incorporate the allowable test conditions tolerances;

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

d. To help assure against fatigue failures due to repeated testing and operational use.

Unless otherwise specified, the test condition tolerances allowed by this standard are assumed to be incorporated in the environmental design margin. For example, space vehicle items are designed, unless otherwise specified, to thermal environments 10 deg C higher and 10 deg C lower than the maximum predicted thermal ranges, (see 3.25). This 10 deg C environmental design margin includes a ± 3 deg C tolerance for acceptance test conditions and a ± 3 deg C tolerance for qualification test conditions.

Unless otherwise specified, space vehicle items are also designed to acoustic noise and random vibration environments that are 6 dB above the maximum predicted levels. This 6 dB environmental design margin for acoustic noise and random vibration includes a ± 1.5 dB tolerance in the overall level (integrated root mean square value over the total frequency range of the test spectrum) for acceptance test conditions and a ± 1.5 dB tolerance for qualification test conditions.

When the qualification or acceptance tests are controlled using test condition tolerances with magnitudes less than specified herein (3 deg C or 1.5 dB), the environmental design margins (10 deg C or 6 dB) may be reduced accordingly. For example, if qualification and acceptance acoustic tests were both controlled to ± 1.0 dB, the design margin would be 5 dB instead of 6 dB. If larger test condition tolerances are allowed, then the design margins would be increased accordingly.

Other environmental design margins applicable to space vehicle items include 6 dB for shock, 6 dB for sinusoidal vibration, a factor of 2 for launch or injection acceleration, and a factor of 1.25 for maximum acceleration of deployed components on a spinning space vehicle.

Another element of the environmental design margin is the time the item is exposed to the design environmental levels. An increase in exposure time or number of cycles over that expected in operation is usually specified for vibration and acoustic design environments to increase confidence that wearout or fatigue failures will not occur. Of course, the environmental design margins may be changed to either higher or lower levels, or to longer or shorter exposure times, depending upon specific program requirements and allowable risk.

3.13 FLIGHT UNIT

A flight unit is a component, subsystem, or space vehicle intended for actual flight.

3.14 ITEM LEVELS

The item levels used in this standard, from the simplest division to the more complex, are: part, subassembly, component, subsystem, space vehicle, and system.

3.15 LAUNCH SYSTEM

A launch system is the composite of equipment, skills, and techniques capable of launching and boosting the space vehicle into orbit. The launch system includes the space vehicle, the launch vehicle, and related facilities, equipment, material, software, procedures, services, and personnel required for their operation.

3.16 LAUNCH VEHICLE, EXPENDABLE

An expendable launch vehicle is a composite of the booster initial stages, injection stages, space vehicle adapter, and fairing having the capability of a single launching and injection of a space vehicle or vehicles into orbit.

3.17 LAUNCH VEHICLE, RECOVERABLE

A recoverable launch vehicle is a composite of booster stages, injection stages, and a space vehicle carrier having the capability to carry a space vehicle to a parking orbit. Elements of the launch vehicle may be recovered for use and may return to earth with or without the space vehicle.

3.18 LIMIT LOAD

The limit load is the maximum anticipated load, or combination of loads, which a structure may be expected to experience during the performance of specified missions in specified environments. Since the actual loads that are experienced in service are in part random in nature, statistical methods for predicting limit loads are employed wherever appropriate.

3.19 MAXIMUM PREDICTED ACCELERATION

The maximum predicted acceleration is the acceleration value determined from the combined effects of the quasi-steady acceleration and the transient response of the vehicle to engine ignition, engine burnout, and stage separation. Where the natural frequency of the component mount or mounting structure may couple with these engine-initiated transients, the maximum predicted acceleration level shall account for the possible dynamic amplification. Different levels may be applied in different axes if the mounting orientation in the space vehicle is specified.

3.20 MAXIMUM PREDICTED ACOUSTIC ENVIRONMENT

The maximum predicted acoustic environment is the extreme value of fluctuating pressure occurring on the external surface of the space vehicle which occurs during liftoff, powered flight, or reentry. The maximum predicted acoustic environment test spectrum is specified based on one-third octave bands over a frequency range of 32 to 10,000 hertz (Hz). The duration of the maximum environment is the total period when the overall amplitude is within 6 dB of the maximum overall amplitude. Where sufficient data are available, the maximum predicted environment may be derived using parametric statistical methods. The data must be tested to show a satisfactory fit to the assumed underlying distribution. The maximum predicted environment is defined as equal to or greater than the value at the ninety-fifth percentile value at least 50 percent of the time. Where there are less than three data samples, a minimum margin of 3 dB is applied to the prediction to account for the variability of the environment.

3.21 MAXIMUM PREDICTED OPERATING PRESSURE

The maximum predicted operating pressure is the working pressure applied to a component by the pressurizing system with the pressure regulators and relief valves at their upper operating limit, including the effects of temperature, transient peaks, and vehicle acceleration.

3.22 MAXIMUM PREDICTED PYRO SHOCK ENVIRONMENT

The pyro shock environment imposed on the space vehicle components is due to structural response when the space or launch vehicle electro-explosive devices are activated. Resultant structural response accelerations resemble the form of superimposed complex decaying sinusoids which decay to a few percent of their maximum acceleration in 5 to 15 milliseconds.

The maximum predicted pyro shock environment is specified as a maximum absolute shock response spectrum determined by the response of a number of single-degree-of-freedom systems using Q = 10. The Q is the acceleration amplification factor at the resonant frequency for a lightly damped system. This shock response spectrum is determined at frequency intervals of one-sixth octave or less over a frequency range of 100 to 10,000 Hz. Where sufficient data are available, the maximum predicted environment may be derived using parametric statistical methods. The data must be tested to show a satisfactory fit to the assumed underlying distribution. The maximum predicted environment is defined as equal to or greater than the value at the ninety-fifth percentile value at least 50 percent of the time. Where there are less than three data samples, a minimum margin of 4.5 dB is applied to account for the variability of the environment.

3.23 MAXIMUM PREDICTED RANDOM VIBRATION ENVIRONMENT

The random vibration environment imposed on the space vehicle components is due to the lift-off acoustic field, aerodynamic excitations, and transmitted structure-borne vibration. The maximum predicted random vibration environment is specified as a power spectral density, based on a frequency resolution of 1/6 octave (or narrower) bandwidth analysis, over a frequency range of 20 to 2000 Hz. A different spectrum may be required for different equipment zones or for different axes. The component vibration levels are based on vibration response measurements made at the component attachment points during ground acoustic tests or during flight. The duration of the maximum environment is the total period during flight when the overall amplitude is within 6 dB of the maximum overall amplitude. Where sufficient data are available, the maximum predicted environment may be derived using parametric statistical methods. The data must be tested to show a satisfactory fit to the assumed underlying distribution. The maximum predicted environment is defined as equal to or greater than the value at the ninety-fifth percentile value at least 50 percent of the time. Where there are less than three data samples, a minimum margin of 3 dB is applied to account for the variability of the environment.

3.24 MAXIMUM PREDICTED SINUSOIDAL VIBRATION ENVIRONMENT

The sinusoidal vibration environment imposed on the space vehicle subsystems and components is due to sinusoidal and narrow band random forcing functions within the launch vehicle or space vehicle during flight or from ground transportation and handling. In flight sinusoidal excitations may be caused by unstable combustion, by coupling of structural resonant frequencies with propellant system resonant frequencies (POGO), or by imbalances in rotating equipment in the launch vehicle or space vehicle. Sinusoidal excitations may occur during ground transportation and handling due to the resonant response of tires and suspension systems of the transporter. The maximum predicted sinusoidal vibration environment is specified over a frequency range of 20 to 2000 Hz for flight excitation and 0.3 to 300 Hz for ground transportation excitation. Where sufficient data are available, the maximum predicted environment may be derived using parametric statistical methods. The data must be tested to show a satisfactory fit to the assumed underlying distribution. The maximum predicted environment is defined as equal to or greater than the value at the ninety-fifth percentile value at least 50 percent of the time. Where there are less than three data samples, a minimum margin of 3 dB is applied to account for the variability of the environment.

3.25 MAXIMUM AND MINIMUM PREDICTED COMPONENT TEMPERATURES

The maximum and minimum predicted component temperatures are the highest and lowest temperatures that can be expected to occur on each component of the space vehicle during all operational modes plus an uncertainty factor. The component temperatures are predicted by an analytical thermal model for all operational modes. This analytical model includes the effects of worst case combinations of equipment operation, internal heating, space vehicle orientation, solar radiation, eclipse conditions, ascent heating, and degradation of thermal surfaces during the life of the mission. The analytical model used in this prediction is usually validated by a space vehicle thermal balance test under the worst case operational modes. An appropriate margin for uncertainties is applied to the extreme component temperatures predicted by the analytical model, even after validation by a thermal balance test, to obtain the maximum and minimum predicted temperatures. This margin accounts for uncertainties in parameters such as complicated view factors, surface properties, contamination, radiation environment, joint conduction, and inadequate ground simulation. Because of these uncertainties, an uncertainty margin (see 3.45) of at least 11 deg C is included in all cases in determining the maximum or minimum predicted temperatures for space vehicle components. This 11 deg C thermal margin is applied to the temperature predictions made after the qualification thermal balance test. This implies that even larger thermal margins are required at the beginning of a program to accommodate changes that typically evolve from preliminary design to the final product.

3.26 MOVING MECHANICAL ASSEMBLIES

Moving mechanical assemblies are the mechanical or electromechanical devices that control the movement of one mechanical part of a space vehicle relative to another part. They include but are not limited to deployment mechanisms, pointing mechanisms, drive mechanisms, despin mechanisms, and the actuators, motors, linkages, latches, clutches, springs, cams, dampers, booms, gimbals, gears, bearings, and instrumentation that are an integral part of these mechanical assemblies.

3.27 MULTIPACTING

Multipacting is the resonant back and forth flow of secondary electrons in a vacuum between two surfaces separated by a distance such that the electron transit time is an odd integral multiple of one half the period of the alternating voltage impressed on the surfaces. Multipacting requires an electron impacting one surface to initiate the action, and requires the secondary emission of one or more electrons at each surface to sustain the action. Multipacting is an unstable self-extinguishing action which can occur at pressures less than 6.65 pascals (0.05 Torr), except that it may become stable at pressures less than 0.0133 pascals (0.0001 Torr). The pitting action resulting from the secondary emission of electrons degrades the impacted surfaces. The secondary electron emission can also increase the pressure in the vicinity of the surfaces causing ionization (corona) breakdown to occur. These effects can cause degradation of performance or permanent failure of the radio frequency cavities, waveguides, or other devices involved.

3.28 ON-ORBIT SYSTEM

An on-orbit system is the composite of equipment, skills, and techniques permitting on-orbit operation of the space vehicle. The on-orbit system includes the space vehicle(s), the command and control network, and related facilities, equipment, material, software, procedures, services, and personnel required for their operation.

3.29 OPERATIONAL MODES

The operational modes for a component or space vehicle include all combinations of operational configurations or conditions that can occur during their service life (see 3.37). Some examples are: power on or power off, command modes, readout modes, attitude control modes, antennas stowed or deployed, and spinning or despun.

3.30 OPTIONAL TESTS

Optional tests are those tests which are not normally required, but which may be necessary due to special requirements of usage or to peculiarities of a particular configuration.

3.31 PART

A part is a single piece, or two or more pieces joined together, which are not normally subject to disassembly without destruction or impairment of the design use. Some examples are resistors, transistors, integrated circuits, relays, capacitors, gears, screws, and mounting brackets.

3.32 PASCAL

The pascal (Pa) is the unit of pressure in the international metric system. One pascal is equal to 0.000145 lbs per sq in. or 0.0075 Torr.

3.33 PRELAUNCH VALIDATION TESTS

Prelaunch validation tests are conducted at the launch base to assure readiness of the hardware, computer programs, personnel, and procedures to support launch and the program mission.

3.34 PROOF PRESSURE

The proof pressure is the test pressure that pressurized components can sustain without detrimental deformation. The proof pressure is used to give evidence of satisfactory workmanship and material quality, or to establish maximum possible flaw sizes. It is equal to the product of maximum expected operating pressure (see 3.21), proof pressure design factor, and a factor accounting for the difference in material properties between test and design temperature.

3.35 QUALIFICATION TESTS

Qualification tests are formal contractual demonstrations that the design, manufacturing, and assembly have resulted in hardware and computer programs conforming to specification requirements.

3.36 QUALIFICATION TEST VEHICLE

A qualification test vehicle is a space vehicle which is used for qualification tests. The qualification test vehicle should be representative of the flight hardware to the extent necessary to achieve a valid qualification test for the items being qualified.

3.37 SERVICE LIFE

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

3.38 SPACE VEHICLE

A space vehicle is a complete, integrated set of subsystems and components capable of supporting an operational role in space. A space vehicle may be an orbiting vehicle, a major portion of an orbiting vehicle, or a payload which performs its mission while attached to a recoverable launch vehicle. The airborne support equipment which is peculiar to programs utilizing a recoverable launch vehicle shall be considered a part of the space vehicle being carried by the launch vehicle.

3.39 SUBASSEMBLY

The term subassembly denotes two or more parts joined together to form a stockable unit which is capable of disassembly or part replacement. Examples are a printed circuit board with parts mounted, or a gear train.

3.40 SUBSYSTEM

A subsystem is an assembly of two or more components, including the supporting structure to which they are mounted, and any interconnecting cables or tubing. A subsystem is composed of functionally related components that perform one or more prescribed functions. Typical space vehicle subsystems are electrical power, attitude control, telemetry, instrumentation, command, structure, thermal control, and propulsion.

3.41 SYSTEM

A system is the composite of equipment, skills, and techniques capable of performing or supporting an operational role. A system includes all operational equipment, related facilities, material, software, services, and personnel required for its operation. Examples of systems that include space vehicles as a major subtier element are launch systems and on-orbit systems.

3.42 TEST DISCREPANCY

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

3.43 TEST ENVIRONMENTS

The test environments represent conditions the unit may experience during ground processing and flight operations. Such conditions normally include acoustic, vibration, acceleration, shock, thermal vacuum, nuclear radiation, and electromagnetic radiation, but should also include as appropriate humidity, rain, salt spray, fungus, internal pressure, and variations in interface power and signal characteristics.

3.44 TEST UNIT FAILURES

A failure of the test unit is defined as a test discrepancy (see 3.42) that is due to a design, workmanship, or quality deficiency in the unit being tested. Non-repeatable, momentary, or any other kind of discrepancy that occurs during testing is considered a failure of the test unit unless it can be determined to have been due to some other cause.

3.45 THERMAL UNCERTAINTY MARGIN

A thermal uncertainty margin is included in the thermal analysis of space vehicles to account for uncertainties in parameters such as complicated view factors, surface properties, contamination, radiation environments, joint conduction, and inadequate ground simulation. For components that have no thermal control, or have passive thermal control, the maximum predicted component temperatures should be at least 11 deg C above the maximum temperature estimated for each component based on measurements and analysis, and the minimum temperature should be at least 11 deg C below the minimum temperature estimated for each component based on measurements and analysis. The 11 deg C is the thermal uncertainty margin for the component. For active thermal control subsystems, a remaining control authority of at least 25 percent for either or both hot and cold limits is specified as the thermal uncertainty margin. It is used to provide a control margin equivalent to the 11 deg C uncertainty margin specified for passively controlled components. For example, if a 100 watt capacity proportional control heater is used, it should operate at 80 watts or less to maintain the component above tne minimum predicted temperature. The duty cycle should be less than 80 percent for an on-off heater. A control authority margin in excess of 25 percent should be demonstrated in cases where an 11 deg C change in the analytically predicted component temperatures would cause the temperature of any part of the actively controlled component to exceed an acceptable temperature limit.

3.46 ULTIMATE LOAD

The ultimate load is the maximum static load to which a structure is designed. It is obtained by multiplying the limit load (see 3.18) by the ultimate factor of safety.

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SECTION 4

GENERAL REQUIREMENTS

4.1 TESTING PHILOSOPHY

The complete test program for space vehicle hardware and software encompasses development, qualification, acceptance, prelaunch validation, and other categories of testing. Test methods, environments, and measured parameters shall be selected to permit the collection of empirical design parameters and the correlation of data throughout the complete test program. A satisfactory test program requires the completion of specific test objectives prior to the accomplishment of others. Design suitability is demonstrated in the development test program prior to the start of formal qualification testing. Qualification testing should be completed prior to the initiation of flight hardware acceptance testing.

4.1.1 Development Testing. Development tests are intended to validate hardware and computer program design concepts and to assist in the evolution of designs from the conceptual phase to the operational phase. An objective of these tests is to identify hardware and computer program problems early in their design evolution so that any required corrective actions can be taken prior to starting formal qualification testing. Development tests may be used to confirm performance margins, manufacturability, testability, maintainability, reliability, life expectancy, and compatibility with system safety. Where practicable, development tests should be conducted over a range of operating conditions that exceed the design limits to identify marginal design features.

4.1.2 Qualification Testing. Qualification tests demonstrate that adequate margins exist in the final product to assure that specification requirements are met. To accomplish this, the qualification test levels are usually specified as the design levels. These qualification test levels are established to exceed the range of environments and stresses expected in any subsequent use ranging from acceptance testing through mission application. For vibration testing, these margins must further asure that repeated acceptance testing, if necessary, will not jeopardize the integrity of the hardware. Qualification test durations for vibration or acoustic testing of reusable hardware shall be increased for each additional flight planned to provide a sufficiently high margin to assure equipment integrity after repeated environmental exposures. The qualification tests should validate the planned acceptance test program including test techniques, test procedures, test environments, ground support test equipment, and computer software.

4.1.3 Acceptance Testing. Acceptance tests are intended to demonstrate the flight-worthiness of each deliverable item. Acceptance tests should demonstrate acceptable performance over the specified range of mission requirements. The acceptance tests should measure performance parameters and should reveal inadequacies in the manufacturing process such as workmanship or material. The performance parameter measurements should establish a baseline that can be used to assure that there are no data trends established in successive tests which indicate a constant degradation of performance within specification limits that could result in unacceptable performance in flight. In-process production evaluation tests, burn-in or wear-in tests, and environmental stress screening tests are also considered to be acceptance tests.

4.1.4 Prelaunch Validation Testing. When a space vehicle is first delivered to the launch site, initial tests are conducted to assure space vehicle readiness for integration with the launch vehicle. Subsequent prelaunch validation testing demonstrates that successful integration of the space vehicle, launch vehicle, and launch facility has been accomplished, and that compatibility exists between the space vehicle hardware and computer software and the launch and on-orbit systems.

4.1.5 Other Testing. Other testing that is not development, qualification, acceptance, or prelaunch validation testing includes on-orbit tests and testing of reuseable flight hardware.

4.1.5.1 On-Orbit Testing. Testing requirements of space vehicles that are in orbit are an important consideration in the design of the space vehicle. However, the on-orbit tests are so program peculiar that specific requirements are not addressed in this standard.

4.1.5.2 Reuseable Flight Hardware Testing. Reusable space vehicle hardware consists of the space vehicles and components intended for repeated space missions. Airborne support equipment and space vehicles which perform their missions while attached to a recoverable launch vehicle are candidates for reuse, particularly for multiple mission programs. The reusable equipment would be subjected to repeated exposure to test, launch, flight, and recovery environments throughout its service life (see 3.37). The accumulated exposure time of space vehicles retained in the recoverable launch vehicle and of airborne support equipment is a function of the planned number of missions involving this equipment and the retest requirements between missions. Airborne support equipment environmental exposure time is further dependent on whether or not its use is required during the acceptance testing of each space vehicle. In any case, the service life of reusable hardware should include all planned reuses and all planned retesting between uses.

The testing requirements for reusable space hardware after the completion of a mission and prior to its reuse on a subsequent mission depends heavily upon the design of the reusable item and the allowable program risk. For those reasons, specific details are not presented in this standard. Similarly, orbiting space vehicles that have completed their useful life spans may be retrieved by means of a recoverable launch vehicle, refurbished, and reused. Until some insight is provided by experience as to the extensiveness of required refurbishment, detailed test guidelines cannot be provided. Based on present approaches, it is expected that the retrieved space vehicle would be returned to the contractor's factory for disassembly, physical inspection, and refurbishment. All originally specified acceptance tests should be conducted before reuse.

4.2 TEST PLANS AND PROCEDURES

4.2.1 Preparation. The contractor shall establish procedures for performing all required tests in accordance with detailed test plans approved by the contracting officer. The test plans shall be based upon a function by function mission analysis and the test requirements. The test plans should indicate the test requirements, testing approach for each item, related special test equipment, facilities, and system interface requirements. The test plans should identify the allocation of requirements to appropriate testable levels of assembly. Traceability shall be provided from the specified requirements to the test procedures. The test procedures shall cover all operations in enough detail so that there is no doubt as to what is to be done. The pass-fail test criteria shall be determined prior to the start of every test.

4.2.2 Test Condition Tolerances. The test condition tolerances allowed by this standard shall be applied to the nominal test values specified. Unless otherwise specified, the following maximum allowable tolerances on test conditions shall apply.

Temperature	± 3 deg C
Pressure	
Above 1.3×10^2 pascals (1 Torr)	± 10 percent
1.3×10^{-1} to 1.3×10^2 pascals (0.001 Torr to 1 Torr)	± 25 percent
Less than 1.3×10^{-1} pascals (0.001 Torr)	± 80 percent
Relative Humidity	± 5 percent
Acceleration	± 10 percent
Vibration Frequency	± 2 percent
Sinusoidal Vibration Amplitude	± 10 percent
Random Vibration Acceleration	
Power Spectral Density	
20 to 500 Hz (25 Hz or narrower)	± 1.5 dB
500 to 2000 Hz (50 Hz or narrower)	± 3.0 dB
Random Overall grms	± 1.5 dB
Sound Pressure Level	
1/3 Octave Band	± 3.0 dB
Overall	± 1.5 dB
Shock Response Spectrum (Q = 10)	
1/6 Octave Band Center Frequency Amplitude	± 6 dB with 30 percent of the response spectrum center frequency amplitudes greater than nominal test specification
Static Load	± 5 percent

4.2.3 Tailoring. This standard specifies test requirements that have been shown to assure high reliability in space missions. However, it is intended that these test baselines should be tailored to each space program considering design complexity, state of the art, mission criticality, cost, and acceptable risk. For some space programs this tailoring may relax the requirements in this standard, while for other programs the requirements may be made more stringent to reduce risk of on-orbit failures or to demonstrate with greater confidence that the space vehicle or components perform adequately when all parameters, environments, and related uncertainties are considered. For example, the optional tests shown in the test baseline tables should be added as required tests, where appropriate, as determined from considerations of design features, required lifetime, environmental exposure, and expected useage. The tailoring is a continuing process throughout the acquisition that should be implemented by the wording used to state the testing requirements in the specifications of the space system, space vehicles, and components or in other applicable contractual documents.

4.3 RETEST

If a test discrepancy (see 3.42) occurs, the test should be interrupted and the discrepancy verified. If the discrepancy is dispositioned as due to the test setup, software, or to a failure in the test equipment, the test being conducted at the time of the failure may be continued after the repairs are completed, as long as the discrepancy did not result in an overstress test condition. If the discrepancy is dispositioned as a failure in the item under test, the preliminary failure analysis and appropriate corrective action shall normally be completed before testing is resumed. If the failure occurs during system testing, the test may be continued if the discrepant area is not affected by the continuation of testing.

The conducting of a proper failure analysis plays an important part in the decision on the type of retest. It should include the determination of whether a failure occurred, the cause of the failure, the physics of the failure, and isolation of the failure to the smallest replaceable item. The degree of retest shall be determined for each case based upon the nature of the failure. In the case of a significant redesign of a component, all previous qualification tests shall be repeated. After significant component rework, all previous acceptance tests except burn-in shall be repeated. In the case of extensive component rework, repetition of the burn-in is also required. Where the redesign or rework of the component is very minor, it may be acceptable to only repeat functional testing and the test in which the failure occurred.

Where significant redesign or rework of components is required as the result of failure during system level testing, the system level test in which the failure occurred, as well as functional testing of the failed subsystem, shall be repeated. Repetition of system level environmental tests may be necessary if the redesign was extensive or the number of components changed out and connectors demated is so large as to reduce confidence in the space vehicle.

4.4 TEST DATA ANALYSIS

A test data bank containing all pertinent system, space vehicle, subsystem, and component test data taken throughout the program shall be maintained. To permit as complete an evaluation as possible of component, subsystem, and space vehicle performance under the various specified test conditions, all relevant test measurements and the environmental conditions imposed on the units shall be recorded on magnetic tape or by other suitable means. These records are intended for post-test analysis to supplement the real-time monitoring and to facilitate the mechanized accumulation of trend data for the critical test parameters. Test data shall be examined for out of tolerance values and for characteristic signatures. Transient responses and mode switching tests shall be examined for proper response. The test data shall also be compared across major test sequences for trends or evidence of anomalous behavior.

SECTION 5

DEVELOPMENT TESTING

5.1 GENERAL

The objective of the development tests is to assure that testing of critical items at all levels of assembly is sufficient to validate the design approach. Requirements for development testing therefore depend upon the maturity of the subsystems and components used and upon the operational requirements of the specific program. Development tests are necessary to validate new design concepts and the application of proven concepts and techniques to a new configuration. Development tests are also conducted to verify design criteria for structures and components and to determine design margins and failure modes. Development tests may be conducted on breadboard equipment, prototype hardware, or the development test vehicle equipment and software. When development tests are proposed on qualification or flight hardware, the approval of the contracting officer is required.

By its nature, development testing cannot be reduced to a standardized set of procedures. The development test requirements are necessarily unique to each new space vehicle. It is not the intent of this section to define the required development tests, but to provide guidelines for conducting appropriate tests when their need has been established.

5.2 COMPONENT DEVELOPMENT TESTS

The major portions of the development tests are conducted on breadboards and prototypes at the subassembly and component levels. The objective is early verification of the critical design concepts to reduce the risk involved in committing the design to qualification and flight hardware. The emphasis for hardware is on packaging design, electronic and mechanical performance, and capability to withstand environmental stress. New designs should be characterized across worst case voltage, frequency, and temperature variations at the breadboard level. Functional testing in thermal and vibration environments is normally conducted. Development tests of deployables and of the attitude control subsystem are normally conducted. Life tests of critical items which may have a wearout failure mode, such as moving mechanical assemblies (see 3.26), should also be conducted.

For electronic boxes, thermal mapping in a vacuum environment for known boundary conditions may be needed to verify the internal component thermal analysis. The correlated thermal model is then used to demonstrate that critical piece part temperature limits, consistent with reliability requirements and performance, are not exceeded. When electronic box packaging is not accomplished in accordance with known and accepted techniques relative to the interconnect system, parts mounting, board sizes and thickness, number of layers, thermal coefficients of expansion, or installation method, development tests should be performed. The tests should establish confidence in the design and manufacturing processes used. Temperature cycling and random vibration testing should be conducted to evaluate the entire package.

5.3 SPACE VEHICLE AND SUBSYSTEM DEVELOPMENT TESTS

Tests of structural and thermal development models are often necessary to confirm dynamic and thermal environmental criteria for design of subsystems, to verify mechanical interfaces, and to assess functional performance of deployment mechanisms and thermal control systems. Space vehicle development testing also provides an opportunity to develop handling and operating procedures as well as to understand system interactions.

For recoverable launch vehicles, a mechanical fit and operational interface test with the facilities at the launch site is recommended. The flight structure should be used if possible; however, a facsimile or portions thereof may be used to conduct the development tests at an early point in the schedule in order to reduce the impact of hardware design changes that may be necessary.

A modal survey is normally conducted to define or verify an analytically derived dynamic model of the space vehicle for use in launch vehicle flight loading event simulations and for use in examinations of post-boost configuration elastic effects upon control precision and stability. This test is conducted on a flight quality structural subsystem as augmented by mass simulated components. The data obtained should be adequate to define orthogonal mode shapes, mode frequencies, and mode damping ratios of all modes which occur within the frequency range of interest. In most instances, modes in the frequency range from zero to 50 Hz should be measured.

A structural development test is also needed to verify the stiffness properties of the space vehicle and to measure member load and stress distributions and deflections in structures with redundant load paths. The stiffness data are of particular interest where nonlinear structural behavior exist that is not fully exercised in the modal survey. This may include nonlinear bearings, elastic buckling of panels, gapping at preloaded interfaces, and slipping at friction joints. The member load and stress distribution data are used to experimentally verify the Loads Transformation Matrix (LTM). Deflection data are also used to experimentally verify the Deflection Transformmation Matrix (DTM). These matrices are used, in conjunction with the dynamic model, to calculate member loads (axial forces, bending moments, shears, torsional moments), and various stresses and deflections, which are converted into design and clearance margins for the space vehicle. This development test could be done in conjunction with the modal survey on the complete space vehicle or, alternatively, it could be done at a subcontractor's facility where significantly large subassemblies could be verified prior to shipping to the integrating contractor. This development test does not replace the structural static load test (see 6.3.1) that is required for vehicle qualification; however, the two tests may be incorporated into a single test sequence that encompasses the requirements of both tests.

Since high frequency vibration and shock responses are difficult to predict by analytical techniques, acoustic and shock development testing of the space vehicle may be necessary to verify the adequacy of the dynamic design criteria for components of the space vehicle. Space vehicle components that are not installed at the time of the test should be dynamically simulated as closely as practical with respect to mass, center of gravity, moments of inertia, interface stiffness, and geometric characteristics. For the acoustic test the space vehicle is normally exposed to the qualification environment in an acoustic chamber. Since the acoustic test is intended to excite a random vibration environment throughout the space vehicle which simulates the environment experienced during launch and flight, a test with the vehicle enclosure installed is highly desirable for vehicles that have an enclosure as a part of the flight configuration. For the pyro shock test all pyrotechnically operated devices and other equipment capable of imparting a significant shock impulse to the space vehicle should be operated. When significant shock levels are expected from subsystems not on board the space vehicle under test, such as the launch vehicle separation shock, the adaptor subsystem or suitable simulation shall be attached and appropriate pyrotechnics or other means used to simulate the shock imposed. Since the pyro shock environment may vary significantly between test exposures, a minimum of three tests of each dominant pyro-device should be used to simulate the maximum predicted shock environment (see 3.22).

A thermal balance development test may be necessary to verify the analytical modeling of the space vehicle and component thermal design criteria. For space vehicles in which structural thermal distortions are critical to mission success, this test also evaluates potential alignment problems. The configuration of the space vehicle should consist of a thermally equivalent structure with addition of equipment panels, thermal control insulation, finishes, and thermally equivalent models of electronic, pneumatic, and mechanical components. Testing is conducted in a space simulation test chamber capable of simulating the ascent, transfer orbit, and orbital thermal vacuum conditions.

The dynamic environment experienced by the space vehicle during road transportation is normally controlled to levels less than the maximum levels predicted for launch and flight. An analysis of space vehicle response to the transportation environment requires definition of the road surface. Since the latter is difficult to define, it is often necessary to conduct a development test of the space vehicle and transporter over a representative course to verify that the space vehicle would not be damaged. For such a test, a space vehicle development model or a simulator which has space vehicle mass properties.may be used, and instrumentation would be installed to measure both space vehicle and transporter dynamic responses.

SECTION 6

QUALIFICATION TESTING

6.1 GENERAL QUALIFICATION TEST REQUIREMENTS

6.1.1 Qualification Hardware. The space vehicle hardware used for qualification shall be produced from the same drawings, using the same materials, tooling, and methods as flight hardware, except as modified to accommodate minor changes that may be necessary to conduct the test. These minor changes may include adding such items as thermocouples, strain gages, and monitoring leads. The qualification hardware shall have demonstrated successful performance at acceptance levels in acoustic, vibration, and thermal environments prior to proceeding to the higher levels for formal qualification testing in that environment. For those environments which are applied by axis, both the acceptance and qualification tests can be completed in one axis before switching to another.

6.1.2 Qualification Test Levels. The qualification test levels shall at least equal the design levels. If the equipment is to be used by more than one program or in different space vehicle locations, the qualification test levels shall envelope the design levels of the various programs or space vehicle locations involved.

6.2 SPACE VEHICLE QUALIFICATION TESTS

The space vehicle qualification test baseline consists of all the required tests specified in Table I. The test baseline shall be tailored for each program, giving consideration to both the required and optional tests; however, deviations from the baseline requirements for the required tests shall be approved by the contracting officer. Additional special tests such as alignments, instrument calibrations, antenna patterns, and mass properties that are conducted as acceptance tests for flight vehicles shall be conducted on the qualification flight vehicle unit. If the space vehicle is controlled by on-board data processing, the flight version of the computer software shall be resident in the space vehicle computer for these tests. The verification of the operational requirements shall be demonstrated to the extent practicable.

Table I. Space Vehicle Qualification Tests

Test	Reference Paragraph	Suggested Sequence	Required (R) OR Optional (O)
Functional	6.2.1	1 (1)	R
EMC	6.2.2	2	R
Acoustic	6.2.3	5	R (2)
Vibration	6.2.4	5	O
Pyro Shock	6.2.5	4	R
Pressure	6.2.6	3, 6	R
Thermal Vacuum	6.2.7	9	R
Thermal Balance	6.2.8	8	R
Thermal Cycling	6.2.9	7	O (3)

Notes:

(1) Electrical and mechanical functional tests shall be conducted prior to and following each environmental test.

(2) Conduct vibration in place of acoustic test for vehicles of compact shape and with weight less than 180 kilograms.

(3) Required if thermal cycling acceptance test 7.1.8 is conducted.

6.2.1 <u>Functional Test, Space Vehicle Qualification</u>.

6.2.1.1 <u>Purpose</u>. This test verifies that mechanical and electrical performance of the space vehicle meets the specification requirements, verifies compatibility with ground support equipment, and validates all test techniques and software algorithms used in computer-assisted commanding and data processing.

6.2.1.2 <u>Mechanical Functional Test</u>. Mechanical devices, valves, deployables, and separable entities shall be functionally tested, with the space vehicle in the launch, orbital, or recovery configuration appropriate to the function. Alignment checks shall be made where appropriate. The maximum and minimum limits of acceptable performance shall be determined with respect to mechanics, time, and other applicable requirements. For each mechanical operation, such as appendage deployments, tests shall demonstrate positive margins of strength, torque margins, and that they function at conditions above and below specified operational limits. Where operation in a 1-g environment cannot be performed, a suitable ground test fixture may be utilized to permit operation and evaluation of the devices. Fit checks shall be made of the space vehicle physical interfaces with the launch vehicle by means of launch vehicle master gages or interface assemblies.

6.2.1.3 Electrical Functional Test. The space vehicle shall be in its flight configuration with all components and subsystems connected except pyrotechnic elements. This test shall verify the integrity of all electrical circuits in the space vehicle, including redundant paths, by the application of an initiating stimulus and the confirmation of the successful completion of the event. The test shall be designed to operate all components, primary and redundant, and all commands shall be exercised. The operation of all thermally controlled components, such as heaters and thermostats, shall be verified by test. The test shall demonstrate that all commands having preconditioning requirements (such as enable, disable, specific equipment configuration, specific command sequence, etc.) cannot be executed unless the preconditions are satisfied. Wherever possible, equipment performance parameters (such as power, voltage, gain, frequency, command and data rates) shall be varied over specification ranges to demonstrate the performance margins. Autonomous functions shall be verified to occur when the conditions exist for which they are designed. The space vehicle main bus shall be continuously monitored by a power transient monitor system. All telemetry monitors shall be verified, and pyrotechnic circuits shall be energized and monitored. A segment of this test shall operate the space vehicle through a mission profile with all events occurring in actual flight sequence to the extent practical. This sequence shall include the final countdown, launch, ascent, separation from the launch vehicle, orbital injection, orbital operation, and return from orbit as appropriate.

6.2.1.4 Supplementary Requirements. Mechanical and electrical functional tests shall be conducted prior to and after each of the environmental tests to detect equipment anomalies and to assure that performance meets the requirements of the specification. These tests do not require the mission profile sequence. Sufficient data analysis to verify the adequacy of the testing and the validity of the data shall be completed before disconnection from a particular environmental test configuration in order that any retesting required can be readily accomplished. During these tests, the maximum use of telemetry shall be employed for data acquisition, problem identification, and problem isolation.

6.2.2 Electromagnetic Compatibility Test, Space Vehicle Qualification

6.2.2.1 Purpose. This test demonstrates electromagnetic compatibility of the space vehicle and ensures the space vehicle has adequate margins in a simulated launch, orbital, and return from orbit electromagnetic environment.

6.2.2.2 Test Description. The operation of the space vehicle and selection of instrumentation shall be oriented toward determining the margin against malfunctions and unacceptable or undesired responses due to electromagnetic incompatibilities. The test shall demonstrate satisfactory electrical and electronic equipment operation in conjunction with the expected electromagnetic radiation from other systems or equipment such as the launch vehicle and the ground support equipment. The space vehicle shall be subjected to the required tests while in the launch, orbital, and return from orbit configurations and in all possible operational modes. Special attention shall be given to areas indicated to be marginal by analysis. Potential electromagnetic interference from the space vehicle to other systems shall be measured. The tests shall be conducted in accordance with the requirements of MIL-STD-1541. Electroexplosive devices with bridge wires, but otherwise inert, shall be installed in the system and monitored during all tests.

6.2.3 Acoustic Test, Space Vehicle Qualification

6.2.3.1 Purpose. This test demonstrates the ability of the space vehicle to withstand or, if appropriate, to operate in the design level acoustic environment which is the maximum level imposed in flight plus a design margin. This test also verifies the adequacy of component vibration qualification criteria.

6.2.3.2 Test Description. The space vehicle shall be installed in a reverberant acoustic cell capable of generating desired sound pressure levels. It shall be mounted on a flight-type support structure or reasonable simulation thereof. The mechanical configuration of the space vehicle shall be as it is during ascent (for example, solar arrays and antennas stowed). Where possible, ground handling equipment and test equipment shall be removed. Adequate dynamic instrumentation shall be installed to measure vibration responses at attachment points of critical and representative components.

6.2.3.3 Test Levels and Duration. The acoustic test spectrum shall be the design environment (see 3.8) which is the maximum predicted flight environment (see 3.20) plus the design margin (6 dB : see 3.12). However, the overall sound pressure level of the qualification test shall not be less than 144 dB. Exposure test time shall be at least three times the expected flight exposure time to the maximum flight environment, or three times the acceptance test duration if that is greater, but not less than 3 minutes. Operating time should be divided approximately equally between redundant circuits. Where insufficient time is available at the full test level to test all redundant circuits, all functions, and all modes, extended testing at a level 6 dB lower shall be conducted as necessary to complete functional testing.

6.2.3.4 Supplementary Requirements. During the test all electrical and electronic components, even if not operating during launch, shall be electrically energized and sequenced through operational modes to the maximum extent possible with the exception of components that may sustain damage if energized. Continuous monitoring of several perceptive parameters shall be provided to detect intermittent failures. Functional tests are required before and after the environmental exposure.

6.2.4 Vibration Test, Space Vehicle Qualification

6.2.4.1 Purpose. This test demonstrates the ability of a space vehicle to withstand or, if appropriate, to operate in the design level vibration environment which is the maximum level imposed in flight plus a design margin. This test also verifies the adequacy of component vibration qualification criteria. This test is only used for small vehicles with weights under 180 kg and with compact shapes.

6.2.4.2 Test Description. The space vehicle shall be attached to a vibration fixture by a flight type adapter. Vibration shall be applied at the base of the adapter via the fixture in each of three orthogonal directions, one direction being parallel to the thrust axis. Generally, antennas, booms, and solar arrays shall be stowed in the launch condition. Adequate dynamic instrumentation shall be installed to measure vibration responses in 3 axes at attachment points of critical and representative components.

6.2.4.3 Test Levels and Duration. The test levels shall be chosen to produce vibration responses within the equipment which are the maximum predicted flight environment (see 3.24) plus the design margin (6 dB : see 3.12). Structural responses shall be limited at resonant frequencies so that structural loads do not exceed design limit loads (see 3.18). Test duration in each axis shall be three times the expected flight exposure time to the maximum flight environment, or three times the acceptance test duration if that is greater, but not less than 3 minutes. Operating time should be divided approximately equally between redundant circuits. Where insufficient time is available at the full test level to test all redundant circuits, all functions, and all modes, extended testing at a level 6 dB lower shall be conducted as necessary to complete functional testing.

6.2.4.4 Supplementary Requirements. For small vehicles with weights of under 180 kilograms (kg) and compact shapes, a vibration test shall be conducted in lieu of an acoustic test. In these cases, special care is required to ensure that loads in excess of design loads are not imposed on the space vehicle. Items such as resonance effects of the structure, adapter type, table control techniques, and location of control accelerometers, shall be carefully considered when vibration is selected in lieu of acoustic testing. During the test, all electrical and electronic components, even if not operating during launch, shall be electrically energized and sequenced through operational modes to the maximum extent possible (with the exception of components that may sustain damage if energized). Continuous monitoring of several perceptive parameters shall be provided to detect intermittent failures. Functional tests are required before and after the environmental exposure.

6.2.5 Pyro Shock Test, Space Vehicle Qualification

6.2.5.1 Purpose. This test demonstrates the capability of the space vehicle to withstand or, if appropriate, to operate in the design level pyro shock environments which are the levels predicted for flight plus a design margin. This test also verifies the adequacy of component pyro shock criteria.

6.2.5.2 Test Description. In this test or series of test segments, all pyrotechnically operated devices and other equipment capable of imparting a significant shock impulse to the space vehicle shall be operated. Separation subsystem shocks are often more severe than those from other pyrotechnic devices, and operation of the separation subsystems is therefore particularly significant. For these tests, the space vehicle shall be suspended or otherwise supported so as to preclude the possibility of recontact between separated portions thereof. When significant shock levels are predicted from subsystems not on board the space vehicle under test, such as the launch vehicle separation shock, the adapter subsystem or suitable simulation shall be attached and appropriate pyrotechnics or other means used to simulate the shock imposed. Adequate dynamic instrumentation shall be installed to measure pyro shock responses in 3 axes at attachment points of critical and representative components.

Support of the space vehicle varies with the configuration and may vary during the course of this test series. To permit optimum positioning and prevent damage to such items as deployment booms, paddles, and ejectables, a series of individual test setups or deployment restraints may be required. The test setup shall permit, as nearly as possible, flightlike dynamic response of the space vehicle structure.

6.2.5.3 Test Levels and Duration. All pyrotechnic devices (e.g., explosive bolt, nut, pin puller, marmon clamp, etc.) shall be fired at least one time. Those pyrotechnic devices producing shock levels within 6 dB of the maximum shock response measured from any of the devices shall be fired two additional times to provide the expected variability in the shock environment. Firing of both primary and redundant pyros shall be in the same sequence as they are designed to fire in flight.

6.2.5.4 Supplementary Requirements. Electrical and electronic components shall be operating and monitored to the maximum extent possible. Functional tests are required before and after environmental exposure.

6.2.6 Pressure Test, Space Vehicle Qualification

6.2.6.1 Purpose. This test demonstrates the capability of fluid subsystems to meet the flow, pressure, and leakage rate requirements specified.

6.2.6.2 Test Description. The space vehicle shall be placed in a facility that provides the services and safety conditions required to protect personnel and equipment during the testing of high pressure subsystems and in the handling of dangerous fluids. Tests shall be performed to verify compatibility with the test setup and to ensure that proper control of the equipment and test functions is provided. The requirements of the subsystem including flow, leakage, and regulation shall be measured while operating applicable valves, pumps, and motors. The flow checks shall verify that the plumbing configurations are adequate. Checks for subsystem cleanliness, moisture levels, and pH shall also be made. Where pressurized subsystems are assembled with other than brazed or welded connections, the specified torque values for these connections shall be verified prior to leak checks.

In addition to the high pressure test, propellant tanks and thruster valves shall be tested for leakage under propellant servicing conditions. The system shall be evacuated to the internal pressure normally used for propellant loading and the systems pressure monitored for any indication of leakage.

6.2.6.3 Test Levels and Duration. The subsystem shall be pressurized to proof pressure (see 3.34) and held for 5 minutes, then the pressure shall be reduced to the maximum predicted operating pressure (see 3.21). Unless specified otherwise, the proof pressure equals 1.5 times the maximum operating pressure. This sequence shall be conducted three times. Inspection for leakage after these cycles shall be at the maximum operating pressure. The duration of the evacuated propulsion system leak test shall not exceed the time that this condition is normally experienced during propellant loading.

6.2.6.4 Supplementary Requirements. Applicable safety standards shall be followed in conducting all tests. Specially formulated bubbleforming solutions are suitable for detecting external leakage at such locations as joints, fittings, plugs, and lines, where the allowable limits are from 0.00001 to 0.01 cubic centimeters per second (cubic cm/sec). Solutions that are used for detecting leaks shall be compatible with the media being leak tested or with the media which could contact any residues. Liquid displacement methods may be used for detecting leakage through poppet seats and internal seals for measurement requirements of 0.1 to 30 cubic cm/sec. Helium or radioactive tracer gas leak detectors may be used for leakage rates from 0.0000001 to 0.0001 cubic cm/sec. The use of halogen gas detectors for liquid propulsion subsystems shall be avoided. Leak tests shall be conducted only after satisfactory proof pressure tests have been completed. Leak detection and measurement procedures may require vacuum chambers, bagging of the entire space vehicle or localized areas, or other special techniques to achieve the required accuracies.

6.2.7 Thermal Vacuum Test, Space Vehicle Qualification

6.2.7.1 Purpose. This test demonstrates the ability of the space vehicle to meet design requirements under vacuum conditions and temperature extremes which simulate those predicted for flight plus a design margin.

6.2.7.2 Test Description. The space vehicle shall be placed in a thermal vacuum chamber and a functional test performed to assure readiness for chamber closure. The vehicle shall be zoned into separate equipment areas based on the location of critical components within each area. Components that operate during ascent shall be monitored for corona, and multipacting (see 3.27) as applicable, as the pressure is reduced to the lowest specified level. Equipment that does not operate during launch shall have electrical power applied after the test pressure level has been reached. A temperature cycle begins with the space vehicle at ambient temperature. The temperature is reduced to the specified low level and stabilized. Component temperature stabilization has been achieved when the rate of temperature change is no more than 3 deg C/hour. Following the cold soak, the temperature shall be raised to the highest specified level and stabilized. Following the high temperature soak, the space vehicle shall be returned to ambient temperatures to complete one temperature cycle. Functional tests shall be conducted during the first and last temperature cycle at both the high and low temperature limits with functional operation and monitoring of perceptive parameters during all other cycles. In addition to the temperature cycles, the chamber shall be programmed through various orbital operations. Operational sequences shall be coordinated with expected orbital environments, and a complete cycling of all equipment shall be performed including the operating and monitoring of redundant equipment and paths. System electrical equipment shall be operating and monitored throughout the test. Strategically placed temperature monitors shall assure attainment of temperature limits. Strategically placed witness plates and quartz crystal microbalances or other instrumentation shall be installed in the test chamber to assure that outgassing from the space vehicle and test equipment does not degrade system performance beyond specified limits.

6.2.7.3 Test Levels and Duration. Temperatures in various equipment areas shall be controlled by the external test environment and internal heating resulting from equipment operation so that during the hot cycle the temperature on at least one component in each equipment area at its design high temperature and one component during the cold cycle is at its design low temperature. The temperature extremes shall be established by a survey of predicted temperatures in various equipment areas and may have to be adjusted to the performance of the most sensitive components in a particular area. Temperatures on the components shall not be allowed to exceed the design levels for the components. The pressure shall be maintained at 0.0133 pascals (0.0001 Torr) or less. All orbital operational conditions and all equipment functional modes including redundancy shall be tested. The qualification test shall include at least eight complete hot-cold cycles at the maximum predicted orbital rate of temperature change and with at least an 8-hour soak at each temperature extreme. Operating time should be divided approximately equally between redundant circuits.

6.2.7.4 Supplementary Requirements. Since the purpose of the more severe temperature extreme is to demonstrate an adequate design margin, it may be necessary to force temperature extremes at certain locations by altering thermal boundary conditions locally or by altering the operational sequence to provide additional heating or cooling. Adjacent equipments may be turned on or off; however, any special conditioning within the space vehicle shall generally be avoided. External baffling, shadowing, or heating shall be utilized to the extent feasible.

6.2.8 Thermal Balance Test, Space Vehicle Qualification

6.2.8.1 Purpose. This test verifies the analytical thermal model and demonstrates the ability of the space vehicle thermal control subsystem to maintain components, subsystems, and the entire space vehicle within the specified operational temperature limits. This test also verifies the adequacy of component thermal design criteria.

6.2.8.2 Test Description. The qualification space vehicle shall be tested to simulate the thermal environment seen by the space vehicle during the transfer orbit and orbital mission phases. Tests shall be conducted over the full mission range of seasons, equipment duty cycles, solar angles, and eclipse combinations so as to include the worst case high and low temperature extremes for all space vehicle components. Special emphasis shall be placed on defining the test conditions expected to produce the maximum and minimum battery temperatures. Sufficient measurements shall be made on the space vehicle internal and external components to effect verification of the space vehicle thermal design and analyses. The power requirements of all thermostatically controlled heaters shall be verified during the test. The test chamber, with the test item installed, shall provide a pressure of 0.0133 pascals (0.0001 Torr), or less. Where appropriate, provisions shall be made to prevent the test item from "seeing" warm chamber walls by using black-coated cryogenic shrouds of sufficient area and shape that are capable of approximating liquid nitrogen temperatures. The space vehicle thermal environment may be supplied by one of the following three methods:

a. Method I. Absorbed Flux. The absorbed solar, albedo, and planetary irradiation is simulated using heater panels or IR spectrum adjusted for the external thermal coating properties and projected by IR lamps or heater panels.

b. Method II. Incident Flux. The intensity, spectral content, and angular distribution of the incident solar, albedo, and planetary irradiation is simulated.

c. Method III. Combination. Thermal environment is supplied by a combination of incident and absorbed irradiation.

The selection of the method and fidelity of the simulation depends upon details of the space vehicle thermal design such as vehicle geometry, the size of internally produced heat loads compared with those supplied by the external environment, and the thermal characteristics of the external surfaces. Instrumentation shall be incorporated down to the component level to evaluate total space vehicle performance within operational limits as well as to identify component problems. The space vehicle shall be operated and monitored throughout the test. Dynamic orbital simulation of the space vehicle thermal environment shall be provided unless the external space vehicle temperature does not vary significantly with time. For example, static simulation is usually adequate for spinning space vehicles.

6.2.8.3 Test Levels and Duration. Test conditions and durations for this test are dependent upon the space vehicle configuration, design, and mission details. Normally, boundary conditions for evaluating thermal design shall include: (a) maximum external absorbed flux plus maximum internal dissipation, and (b) minimum external absorbed flux plus minimum internal power dissipation. The thermal time constant of the subsystems and orbital maneuvering both influence the time required for the space vehicle to achieve thermal equilibrium and hence the test duration. Thermal equilibrium has been achieved when the equipment temperature change is no more than 3 deg C/hour. The tests should simulate the full range of seasons, equipment duty cycles, solar angles, and eclipse combinations so as to produce the worst case high and low temperature extremes for all space vehicle components.

6.2.8.4 Supplementary Requirements. This test augments and validates the detailed thermal analysis. Pass criteria depend not only on survival and operation of each equipment within specified temperature limits, but also on correlation of the test with theoretical thermal models. As a goal, correlation of test results to the thermal model predictions should be within ± 3 deg C. Lack of correlation with the theoretical models may indicate either a deficiency in the model, test setup, or space vehicle hardware. The thermal balance test can be combined with the thermal vacuum test. The correlated thermal math model is then used to make the final temperature predictions for the various mission phases, e.g., prelaunch, ascent, on-orbit, etc. The thermal margins are then based on these final temperature predictions.

6.2.9 Thermal Cycling Test, Space Vehicle Qualification

6.2.9.1 Purpose. This test demonstrates the ability of the space vehicle to withstand the thermal stressing environment of the space vehicle thermal cycling acceptance test (7.1.8) plus a design margin.

6.2.9.2 Test Description. The space vehicle shall be placed in a thermal chamber at ambient pressure, and a functional test shall be performed to assure readiness for the test. The space vehicle shall be operated and monitored during the entire test, except that space vehicle power may be turned off if necessary to reach stabilization at the cold temperature. Space vehicle operation shall be asynchronous with the temperature cycling, and redundant circuits shall be operated with approximately equal time on each redundant circuit. Unfavorable combinations of temperature and humidity shall be avoided so there is no moisture deposition either on the exterior surfaces of the space vehicle or inside spaces where the humidity is slow to diffuse, e.g., multilayer insulation and enclosed electronic equipment. When the relative humidity of the inside spaces of the space vehicle is below the value at which the cold test temperature would cause condensation, the temperature cycling shall begin. One complete temperature cycle is a period beginning at ambient temperature then cycling to one temperature extreme and stabilizing, then to the other temperature extreme and stabilizing, and then returning to ambient temperature. Strategically placed temperature monitors installed on components shall assure attainment and stabilization of the temperature extremes at several components. Auxiliary heating and cooling may be employed for selected temperature-sensitive components, e.g., batteries. If it is necessary to achieve the temperature rate of change, parts of the space vehicle such as solar panels and passive thermal equipment may be removed for the test. The last temperature cycle shall be a soak cycle during which the space vehicle shall remain at each temperature extreme while a functional test, including testing of redundant circuits, is conducted.

6.2.9.3 Test Levels and Duration. The space vehicle temperature range from hot to cold shall be the maximum possible within the constraints of the component design temperatures. The minimum space vehicle temperature range should be 70 deg C. Auxiliary heating and cooling may be used to protect selected temperature sensitive components. The average rate of change of temperature from one extreme to the other shall be as rapid as possible. The test shall include 25 percent more thermal cycles than the thermal cycling acceptance test (7.1.8).

6.3 SUBSYSTEM QUALIFICATION TESTS

Subsystem level qualification tests shall be conducted on those subsystems that are subjected to environmental acceptance tests. Qualification tests shall also be conducted at the subsystem level where this level of testing provides a more realistic test simulation, such as the required structural static load test. At the other extreme, the qualification of certain components such as interconnect tubing or wiring may be most readily completed at the subsystem level of assembly rather than at the component level. In this case, the subsystem may be treated as a component, and the appropriate component level tests conducted at the subsystem level to complete required component qualification tests.

6.3.1 Structural Static Load Test, Subsystem Qualification

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

6.3.1.2 Test Description. The structural configuration, materials, and manufacturing processes employed in the qualification test specimens shall be identical to those of flight articles. When structural items are rebuilt or reinforced to meet specific strength or rigidity requirements, all modifications shall be structurally identical to the changes incorporated in flight articles. The support and load application fixture shall consist of an adequate replication of the adjacent structural section to provide boundary conditions which simulate those existing in the flight article. Static loads representing the design limit load and the design ultimate load (see 3.46) shall be applied to the structure, and measurements of the strain and deformation shall be recorded. Strain and deformation shall be measured before loading, after removal of the limit loads, and at several intermediate levels up to limit load for post-test diagnostic purposes. The test conditions shall include the combined effects of acceleration, pressure, preloads, and temperature. These effects can be simulated in the test conditions as long as the failure modes and design margins are enveloped by the simulations. For example, temperature effects, such as material degradation and additive thermal stresses, can often be accounted for by increasing mechanical loads. Analysis of flight profiles shall be used to determine the proper sequencing or simultaneity for application of thermal stresses. When prior loading histories affect the structural adequacy of the test article, these shall be included in the test requirements. If more than one ultimate load condition is to be applied to the same test specimen, a method of sequential load application shall be developed by which each condition may, in turn, be tested to progressively higher load levels. The final test may be taken to failure to substantiate the capability to accommodate internal load redistribution, to provide data for any subsequent design modification effort, and to provide data for use in any weight reduction programs. Failures at limit load shall include material yielding or deflection which degrade mission performance and at ultimate load shall include rupture or collapse.

6.3.1.3 Test Levels and Duration

a. Static Loads. The loads, other than internal pressure in pressure vessels, shall be increased until failure occurs or until the specified test loads are reached.

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

c. Duration of Loading. Loads shall be applied as closely as possible to actual flight loading times, with a minimum dwell time sufficient to record test data such as stress, strain, deformation, and temperature.

6.3.1.4 Supplementary Requirements. Pretest analysis shall be conducted to identify the locations of minimum design margins and associated failure modes which correspond to the selected critical test load conditions. This analysis shall be used to locate instrumentation, to determine the sequence of loading conditions, and to afford early indications of anomalous occurrences during the test. This analysis shall also form the basis for judging the adequacy of the test loads. Internal loads resulting from the limit test conditions shall envelop all critical internal loads expected in flight; however, excessive internal loads peculiar to the test shall be avoided. In cases where a load or other environment has a relieving effect, the minimum, rather than the maximum, expected value shall be used in defining limit test loading conditions. In some instances, where only a small number of flight vehicles have been included in the program, the cost of a dedicated test article may represent an unacceptably high percentage of the program cost. In such cases, the failure test would not be conducted, and it would be necessary to subject flight hardware to test loads prior to flight. In this event, special precautions shall be taken to ensure that the structure can still withstand its predicted flight environment after it has been subjected to the test loads. Such precautions shall include at least the special design requirement that no permanent deformation detrimental to mission performance shall occur and the inspection requirement that sufficient nondestructive testing be conducted after the test to ensure the integrity of the structure. Alternatively, each flight vehicle shall be proof-tested; proof levels may be less than ultimate levels but shall exceed limit levels. In this case, the vehicle shall be designed to withstand the proof levels without permanent deformation detrimental to mission performance, and a thorough post-test inspection of each flight vehicle shall be conducted to ensure the integrity of the structure.

6.4 COMPONENT QUALIFICATION TESTS

The space vehicle component qualification test baseline consists of all the required tests specified in Table II. The test baseline shall be tailored for each program, giving consideration to both the required and optional tests; however, deviations from the baseline of required tests shall be approved by the contracting officer. Each component that is acceptance tested as a component shall undergo comparable qualification tests as a component. Component qualification tests shall normally be accomplished entirely at the component level. However, in certain circumstances, required component qualification tests may be conducted partially or entirely at the subsystem or space vehicle levels of assembly. Tests of components such as interconnect tubing, radio frequency circuits, and wiring harnesses are examples where at least some of the tests can usually be accomplished at higher levels of assembly.

Where components fall into two or more categories of Table II, the required tests specified for each category shall be applied. For example, a star sensor may be considered to fit both "Electronic Equipment" and "Optical Equipment" categories.

Table II. Component Qualification Tests

TEST	REFERENCE PARAGRAPH	SUGGESTED SEQUENCE	ELECTRONIC OR ELECTRICAL EQUIPMENT	ANTENNAS	MOVING MECHANICAL ASSEMBLY	SOLAR PANEL	BATTERIES	VALVES	FLUID OR PROPULSION EQUIPMENT	PRESSURE VESSELS	THRUSTERS	THERMAL EQUIPMENT	OPTICAL EQUIPMENT
FUNCTIONAL	6.4.1	1(1)	R	R	R	R	R	R	R	R	R	R	R
THERMAL VACUUM	6.4.2	9	R	R	R	R	R	R	R	O	R	R	R
THERMAL CYCLING	6.4.3	8	R	O	O	O	O	O	O	—	—	—	—
SINUSOIDAL VIBRATION	6.4.4	5	O	O	O	—	O	O	O	O	O	O	O
RANDOM VIBRATION	6.4.5	4	R	R(3)	R	—	R	R	R	R	R	R	R
ACOUSTIC	6.4.6	4	—	R(3)	—	R	—	—	—	—	—	—	—
PYRO SHOCK	6.4.7	3	R	O	O	O	O	O	O	—	O	O	O
ACCELERATION	6.4.8	7	O	R	O	O	O	—	—	O	—	—	R
HUMIDITY	6.4.9	10	O	O	O	O	O	O	O	O	O	O	O
PRESSURE	6.4.10	11	—	—	R	—	R(2)	R	R	R	R	—	—
LEAK	6.4.11	2,6,12	R(2)	—	R	—	R(2	R	R	R	O	O	—
EMC	6.4.12	13	R	O	O	—	—	—	—	—	—	—	—
LIFE	6.4.13	14	O	O	O	O	O	O	—	O	O	O	O

LEGEND: R = REQUIRED
O = OPTIONAL TEST
— = NO REQUIREMENT

Notes: (1) Functional tests shall be conducted prior to and following environmental test
(2) Required only on sealed or pressurized equipment
(3) Either random vibration or acoustic test required with the other optional

In this example, a thermal cycling test would be conducted since it is required for electronic equipment, even though there is no requirement for thermal cycling optics. Similarly, an electric motor-driven actuator fits both "Electrical Equipment" and "Moving Mechanical Assembly" categories. The former makes thermal cycling a required test, even though this test is optional for the moving mechanical assembly category.

6.4.1 <u>Functional Test, Component Qualification</u>

6.4.1.1 <u>Purpose</u>. This test verifies that the electrical and mechanical performance of the component meets the requirements of the component specification.

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

6.4.1.3 Supplementary Requirements. A functional test shall be conducted prior to each of the environmental tests to assure that performance meets the requirements of the particular specification. The same functional test shall be conducted after each environmental test.

6.4.2 Thermal Vacuum Test, Component Qualification

6.4.2.1 Purpose. This test demonstrates the ability of the component to perform in a thermal vacuum environment which simulates the design environment for the component (see 3.9).

6.4.2.2 Test Description. The component shall be mounted in a vacuum chamber on a thermally controlled heat sink or in a manner similar to its actual installation in the space vehicle. The component surface finishes shall be identical to those on the flight models. A temperature sensor shall be attached to the component baseplate for conduction-dominated internal designs or to a representative case location(s) for a component cooled primarily by radiation. This sensor shall be used to determine and control the test temperature. Temperature stability has been achieved when the rate of change is no more than 3 deg C/hour. The component heat transfer to the thermally controlled heat sink and the radiation heat transfer to the environment shall be controlled to the same proportions as calculated for the flight environment. The components that are required to operate during ascent shall be operating and monitored for arcing and corona during the initial reduction of pressure to the specified lowest levels. The time for reduction of chamber pressure from ambient to 20 pascals (0.15 Torr) shall be at least 10 minutes to allow sufficient time in the region of critical pressure. Components only operational during ascent may be turned off after the test pressure level has been reached. Equipment that does not operate during launch, and that may otherwise be subject to corona damage, shall have electrical power applied after the test pressure level has been reached. With the chamber at the test pressure level, RF equipment shall be monitored to assure that multipacting does not occur. A temperature cycle begins with the chamber at ambient temperature. The temperature is reduced to the specified low level and stabilized. All components which operate in orbit shall be turned off, then cold started after a soak period sufficient to ensure the component internal temperature has stabilized at the specified temperature, and then functionally tested. With the component operating, the component temperature shall be increased to the upper temperature level. After the component temperature has stabilized at the specified level, the component shall be turned off, then hot started after electrical circuits have been discharged, and then functionally tested. The temperature of the chamber shall then be reduced to ambient conditions. This constitutes one complete temperature cycle.

6.4.2.3 Test Levels and Duration

a. Pressure. The pressure shall be reduced from atmospheric to 0.0133 pascals (0.0001 Torr) or less.

b. Temperature. The component temperature shall be at the design high temperature during the hot cycle and at the design low temperature during the cold cycle.

c. Duration. A minimum of three temperature cycles shall be used. Each cycle shall have a 12 hour or longer dwell at the high and at the low temperature levels during which the unit is turned off until the temperature stabilizes and then turned on.

6.4.2.4 Supplementary Requirements. Functional tests shall be conducted at the high and low temperature levels during the first and last cycle and after return of the component to ambient temperature. During the remainder of the test, electrical and electronic components, including all redundant circuits and paths, shall be monitored for failures and intermittents to the maximum extent possible. Monitoring of the RF output for corona shall be conducted using spectrum monitoring instrumentation during chamber pressure reduction. The RF component shall be operated at maximum power and at design frequency. The force or torque design margin shall be measured on moving mechanical assemblies at the environmental extremes. Compatibility of thrusters with their operational fluids shall be verified at test temperature extremes during thermal vacuum testing.

6.4.3 Thermal Cycling Test, Component Qualification

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

6.4.3.2 Test Description. A thermal cycle begins with the component at ambient temperature. With the component operating (power on) and while perceptive parameters are being monitored, the chamber temperature shall be reduced to the specified low temperature level as measured at a representative location on the component such as the mounting point on the baseplate for conduction-dominated internal designs or at a representative location(s) on the case for radiation-controlled designs. After the component temperature has stabilized at less than 3 deg C/hour rate of change, the unit shall be turned off and then cold started. With the component operating, the chamber temperature shall be increased to the upper temperature level. After the component temperature has stabilized at the specified level, the component shall be turned off and then hot started. The temperature of the chamber shall then be reduced to ambient conditions. This constitutes one thermal cycle.

6.4.3.3 Test Levels and Duration

a. Pressure. Ambient pressure shall normally be used. When unsealed components are being tested, the chamber shall be flooded with dry air or nitrogen to preclude condensation on and within the component at low temperature. If convenient, this test may be performed in thermal vacuum and combined with the test of 6.4.2, provided that the temperature limits, number of cycles, rate of temperature change, and dwell times conform to this test.

b. Temperature. The component temperatures shall be at the design high temeprature during the hot cycle and at the design low temperature during the cold cycle.

c. Duration. Three times the number of thermal cycles as used for acceptance testing but not less than 24 cycles total, of which the last 4 shall be failure free. Each cycle shall have a

1-hour minimum dwell at the high and at the low temperature levels during which the unit shall be turned off until the temperature stabilizes and then turned on. The dwell time at the high and low levels shall be long enough to obtain internal thermal equilibrium. The transitions between low and high temperatures shall be at an average rate of at least 1 deg C per minute.

6.4.3.4 Supplementary Requirements. Functional tests shall be conducted during the first and last thermal cycles at high and low temperatures and after return of the component to ambient. During the remainder of the test, electrical components, including all redundant circuits, shall be cycled through various operational modes and perceptive parameters monitored for failures and intermittents to the maximum extent possible. Compatibility of valves and fluid or propulsion equipment with their operational fluids shall be verified at test temperature extremes during thermal cycling tests.

6.4.4 Sinusoidal Vibration Test, Component Qualification

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

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

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

c. To evaluate fixtures or for diagnostic purposes.

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

6.4.4.3 Test Levels and Duration. Tests conducted to determine resonant conditions or to evaluate fixtures (purposes "b" or "c" above) shall be conducted using test levels and durations sufficient to provide diagnostic capability. Sinusoidal excitation may be applied as a dwell at discrete frequencies or as a frequency sweep with the frequency varying at a logarithmic rate. The sweep rate for diagnostic tests shall be slow enough to allow identification of significant resonances. Tests conducted to demonstrate the degree of ruggedness (purpose "a" above) shall use two (2) minutes per octave unless the sweep rates and dwell times can be based on the persistence of the environment in service use. The vibration levels shall be sufficient to cover the severity of the maximum design levels.

6.4.4.4 Supplementary Requirements. Sinusoidal vibration tests to demonstrate the degree of ruggedness (purpose "a" above) shall be considered a required test where significant sinusoidal vibration is expected in service usage. A functional test shall be conducted before the sinusoidal vibration test and after its completion. Electrical components shall be powered during the test and perceptive parameters monitored for failures or intermittents. When monitoring during the test is not practical, a limited functional test shall be performed after the vibration test for each axis is completed. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test.

6.4.5 Random Vibration Test, Component Qualification

6.4.5.1 Purpose. This test demonstrates the ability of the component to withstand the design level random vibration environment.

6.4.5.2 Test Description. The component shall be mounted to a rigid fixture through the normal mounting points of the component. The component shall be tested in each of three mutually perpendicular axes. Propulsion system valves shall be pressurized to operating pressure for this test and monitored for internal pressure decay if pressurized during ascent.

6.4.5.3 Test Levels and Duration. The random vibration shall be the design level for the component. The minimum overall test level for components weighing 22.7 kg (50 lb) or less shall be 12 grms. The minimum test level for components weighing more than 22.7 kg (50 lb) shall be evaluated on an individual basis. The test duration in each of the three orthogonal axes shall be three times the expected flight exposure time to the maximum predicted environment or three times the component random vibration acceptance test time if that is greater, but not less than 3 minutes per axis. Where insufficient time is available at the full test level to test all redundant circuits, all functions, and all modes, extended testing at a level 6 dB lower shall be conducted as necessary to complete functional testing.

6.4.5.4 Supplementary Requirements. A functional test shall be conducted before and after the completion of the random vibration test. During the test, electrical and electronic components, including all redundant circuits, shall be electrically energized and functionally sequenced through various operational modes to the maximum extent possible. Several perceptive parameters shall be monitored for failures or intermittents during the test. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test and vibration test levels controlled at the input to the isolators.

6.4.6 Acoustic Test, Component Qualification

6.4.6.1 Purpose. This test demonstrates the ability of the component to withstand the design level acoustic environment. Acoustic tests shall be conducted only on components with large surfaces which are likely to be susceptible to acoustic noise excitations.

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

6.4.6.3 Test Levels and Duration. The sound pressure level shall be at least the design level, but not less than 144 dB overall. The duration shall be three times the expected flight exposure time to the maximum predicted environment or three times the acoustic acceptance test duration, whichever is greater, but not less than three minutes. Where insufficient time is available at the full test level to test all redundant circuits, all functions, and all modes, extended testing at a level 6 dB lower shall be conducted as necessary to complete functional testing.

6.4.6.4 Supplementary Requirements. A functional test shall be conducted before and following the acoustic test. During the test, electrical and electronic components, including all redundant circuits, shall be electrically energized and functionally sequenced through various operational modes to the maximum extent possible. Several perceptive parameters shall be monitored for failures or intermittents during the test.

Components characterized by large ratios of surface area to volume, such as large antennas and solar arrays, cannot be tested in a manner which suitably simulates imposition of the service dynamic environment by employing mechanical vibration. For such component configurations acoustic testing shall be required. When acoustic component testing is required, random vibration component testing shall not be required.

6.4.7 Pyro Shock Test, Component Qualification

6.4.7.1 Purpose. This test demonstrates the capability of the component to withstand the design level pyrotechnic shock environment.

6.4.7.2 Test Description. The component shall be mounted to a fixture through the normal mounting points of the component. The selected test method shall be capable of meeting the required shock spectrum with a transient which has a duration comparable to the duration of the expected inflight shock. Numerous test techniques are currently in use for the performance of pyrotechnic shock testing. Methods such as shaped pulses, complex decaying sinusoids, impact devices, and pyrotechnically excited fixtures have been successfully employed. Electrodynamic shakers with control electronics capable of synthesizing the shock transient with a prescribed spectrum have been developed. A mounting of the equipment on actual or dynamically similar structure provides a more realistic test than does a mounting on a rigid structure such as a shaker armature or slip table. A rigid structure tends to overly correlate the input shock motions at the equipment attachment points and thereby can lead to higher responses within the equipment than intended. Sufficient development testing shall be conducted to validate the proposed test method before testing qualification hardware. These test techniques as well as others that may be devised are acceptable, provided the following conditions are met:

a. A transient with the prescribed shock spectrum can be generated within specified tolerances.

b. The applied shock transient provides a simultaneous application of the frequency components as opposed to a serial application. Toward this end, it shall be a goal for the duration of the shock transient to approximate the duration of the actual shock event. In general, the duration of the shock employed for the shock spectrum analysis shall not exceed 20 milliseconds unless a longer duration is required to simulate the actual shock event.

6.4.7.3 Test Levels and Exposure. The shock spectrum in each direction along each of the three orthogonal axes shall be at least the design level for that direction. A sufficient number of shocks shall be imposed to meet the amplitude criteria in both directions on each of the three orthogonal axes at least three times. However, if a suitable test environment can be generated to satisfy the amplitude requirement in all six axial directions by a single application, this test environment shall be imposed three times. If, on the other hand, an imposed shock meets the amplitude requirement in only one direction of a single axis, the shock test shall be conducted a total of 18 times in order to get three valid test amplitudes in both directions of each axis.

6.4.7.4 Supplementary Requirements. A visual inspection shall be made before and after the test. The visual inspection shall not entail the removal of component covers nor any disassembly. Electrical and electronic components, including redundant circuits, shall be energized and monitored to the maximum extent possible. A functional test shall be performed before and after all shock tests, and several perceptive parameters monitored during the shocks to evaluate performance and to detect any failures. Relays shall not transfer and shall not chatter in excess of specification limits. If the component is to be mounted on dynamic isolators in the space vehicle, the component shall be mounted on these isolators during the qualification test.

6.4.8 Acceleration Test, Component Qualification

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

6.4.8.2 Test Description. The component shall be mounted to a test fixture through the normal mounting points of the component. The component shall be tested in each of three mutually perpendicular axes. The specified accelerations apply to the geometric center of the test item. If a centrifuge is used, the arm (measured to the geometric center of the test item) shall be at least five times the dimension of the test item measured along the arm. Inertial components such as gyros and platforms may require counter rotating fixtures on the centrifuge arm.

6.4.8.3 Test Levels and Duration

a. Acceleration Level. The test acceleration level shall be at least the design levels (see 3.12) in each direction for each of the three orthogonal axes. For the axis in the direction of the launch acceleration, the test level shall not be less than 20 g for tests qualifying components for the launch and injection environments.

b. Duration. Five minutes each axis in each direction.

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

6.4.9 Humidity Test, Component Qualification

6.4.9.1 Purpose. This test demonstrates that the component is capable of surviving without excessive degradation the design value of humidity which might be imposed upon the component during fabrication, test, shipment, storage, and preparation for launch.

6.4.9.2 Test Description and Levels. The component shall be installed in the chamber.

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

b. Cycle 1. The temperature shall be increased to +35 deg C over a 1-hour period; then the humidity shall be increased to not less than 95 percent over a 1-hour period with the temperature maintained at +35 deg C. These conditions shall be held for 2 hours. The temperature shall be reduced to 2 deg C over a 2 hour period with the relative humidity stabilized at not less than 95 percent. These conditions shall be held for 2 hours.

c. Cycle 2. The above cycle shall be repeated except that the temperature shall be increased from +2 deg C to +35 deg C over a 2 hour period (moisture is not added to the chamber until +35 deg C is reached).

d. Cycle 3. The chamber temperature shall be increased to +35 deg C over a 2 hour period without adding any moisture to the chamber. The test component shall then be dried with air at room temperature and 50 percent maximum relative humidity by blowing air through the chamber for 6 hours. The volume of air used per minute shall be equal to one to three times the test chamber volume. A suitable container may be used in place of the test chamber for drying the test component.

e. Cycle 4. The component shall be placed in the test chamber and the temperature increased to +35 deg C and the relative humidity increased to 90 percent over a 1 hour period and then these end conditions shall be maintained for at least 1 hour. The temperature shall be reduced to +2 deg C over a 1 hour period with the relative humidity stabilized at 90 percent and these conditions maintained for at least 1 hour. A drying cycle should follow (see Cycle 3).

6.4.9.3 Supplementary Requirements. The component shall be functionally tested prior to the test and at the end of Cycle 3 (within 2 hours after the drying) and visually inspected for deterioration or damage. The component shall be functionally tested during the Cycle 4 periods of stability; after the 1-hour period to reach +35 deg C and 90 percent relative humidity, and again after the 1-hour period to reach the +2 deg C and 90 percent relative humidity condition. The component shall be visually inspected for deterioration or damage after removal from the chamber.

6.4.10 Pressure Test, Component Qualification

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

6.4.10.2 Test Description

a. Proof Pressure. For such items as pressure vessels, pressure lines, and fittings, the temperature of the component shall be consistent with the critical use temperature and subjected to a minimum of one cycle of proof pressure. A proof pressure cycle shall consist of raising the internal pressure (hydrostatically or pneumatically, as applicable) to the proof pressure, maintaining it for 5 minutes and then decreasing the pressure to zero. Evidence of permanent set or distortion that exceeds 0.2 percent or failure of any kind shall indicate failure to pass the test.

b. Proof Pressure for Valves. With the valve in the open and closed positions (if applicable), the proof pressure shall be applied for a minimum of three cycles to the inlet port for 5 minutes (hydrostatically or pneumatically, as applicable). Following the 5-minute pressurization period, the inlet pressure shall be reduced to ambient conditions. The exterior of the unit shall be visually examined. Evidence of deformation that exceeds 0.2 percent or any failure shall indicate failure to pass the test. The test may be conducted at room ambient temperature.

c. Burst Pressure (see 3.4). For such items as pressure vessels, pressure lines, and fittings, the temperature of the component shall be consistent with the critical use temperature, and the component shall be pressurized (hydrostatically or pneumatically, as applicable and safe) to design burst pressure or greater. The internal pressure shall be applied at a uniform rate such that stresses are not imposed due to shock loading.

d. Burst Pressure for Valves. With the valve in the open or closed position, as applicable, the design burst pressure shall be applied to the inlet port for 5 minutes (hydrostatically or pneumatically, as applicable). Following the 5-minute pressurization period, the inlet pressure shall be reduced to ambient conditions. The exterior of the unit shall be visually examined for indications of deformation or failure. The test may be conducted at room ambient temperature.

6.4.10.3 Test Levels

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

b. Proof Pressure. Unless otherwise specified, the proof pressure equals 1.5 times the maximum operating pressure.

c. Burst Pressure. Unless otherwise specified, the burst pressure equals two times the maximum operating pressure.

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

6.4.11 Leakage Test, Component Qualification

6.4.11.1 Purpose. This test demonstrates the capability of pressurized components to meet the design leakage rate constraints specified in the component specifications.

6.4.11.2 Test Description and Alternatives. Component leak checks shall be made prior to initiation of, and following the completion of, component qualification thermal and vibration tests. Proof pressure tests per 6.4.10 shall be successfully completed before conducting leakage tests. The test method employed shall have sensitivity and accuracy consistent with the specified maximum allowable leak rate. One of the following recommended methods shall be used:

a. Method I (gross leak test). The component shall be completely immersed in a liquid so that the uppermost part of the test item is 5 ± 2.5 cm (2 ± 1 inches) below the surface of the liquid. The critical side or side of interest of the component shall be in a horizontal plane facing up. The liquid, pressurizing gas, and the test item shall be 23 ± 10 deg C (73 ± 18 deg F). The gas used for pressurizing shall be clean and dry with a dewpoint of at least -32 deg C (-25 deg F). Any observed leakage during immersion as evidenced by a continuous stream of bubbles emanating from the component indicates a failure of seals.

b. Method II (fine leak test). The component shall be purged with nitrogen and then charged with helium to the required pressure (as specified in the component detail specification) before being sealed. The component shall then be placed in a suitable vacuum chamber and tested for helium leakage with a helium leak detector. The leakage rate shall be used to determine seal integrity and shall not exceed the amount specified in the detailed component specification. This method is applicable to tape recorders and similar components.

c. Method III (for battery cases or pressurized components). The component shall be pressurized with dry nitrogen or other appropriate gas to the specified value. The pressure shall be monitored by a gage (or pressure transducer) for the required time. The drop in pressure shall not exceed the permitted amount as specified under the component specification.

d. Method IV (for hermetically sealed alkaline storage batteries). The battery shall be cleaned with alcohol while in the discharged state. A suitable indicator (e.g., dilute solution of phenolphthalein or other suitable color change indicator) shall be applied to all seams, terminals, and pinch tubes subject to leakage of electrolyte. A change in the color of the indicator shall be an indication of a leak. After testing, the test solution shall be removed (e.g., with distilled water).

e. Method V (for components of pressurized fluid systems). The components shall be pressurized to their maximum working pressure in each of the functional modes. Leakage shall be detected using an appropriate method (6.2.6.4). Propulsion system tanks and thrusters shall also be evacuated to the internal pressure normally used for propellant loading and the internal pressure monitored for indications of leaking.

6.4.11.3 Test Levels and Duration. The leak tests shall be performed with the component pressurized at the maximum operating pressure and then at the minimum operating pressure if the seals are dependent upon pressure for proper sealing. The test duration shall be sufficient to detect any significant leakage. The test levels and duration for the typical methods of 6.4.11.2 are:

a. Method I. The duration of immersion shall be 60 minutes at each pressure.

b. Method II. The external test pressure shall be 0.133 pascals (0.001 Torr) or less and the duration of the test shall be 4 hours (for equipment that is operational in orbit for more than one day).

c. Method III. The test pressure is usually less than 343 kilopascals (50 psi). The pressure drop shall not exceed the specified amount (typically about 6.9 kilopascals (1 psi) in a 6-hour period at room temperature).

d. Method IV. The test results are visable within seconds.

e. Method V. The duration of the evacuated propulsion system component leak test shall not exceed the time that this condition is normally experienced during propellant loading.

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

6.4.12 EMC Test, Component Qualification

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

6.4.12.2 Test Description. The test shall be conducted in accordance with the requirements of MIL-STD-1541. An evaluation shall be made of each component to determine which tests shall be performed as the baseline requirements.

6.4.13 Life Test, Component Qualification

6.4.13.1 Purpose. This test demonstrates the reliability of the component and increases confidence that components which may have a wearout, drift, or fatigue-type failure mode have the capability to withstand the maximum duration or cycles of operation to which they are expected to operate during repeated ground testing and in flight without degradation of their function outside of allowable limits.

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

6.4.13.3 Test Levels and Duration

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

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

c. Duration. The total operating time or number of operational cycles for a component life test shall be twice that predicted during the service life, including ground testing, in order to demonstrate an adequate margin.

d. Functional Duty Cycle. Complete functional tests shall be conducted before the test begins, after each 168 hours of operation and during the last 2 hours of the test. An abbreviated functional test shall be conducted periodically to ascertain that the component is functioning within specification limits.

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

6.5 SUBASSEMBLY LEVEL QUALIFICATION TESTS

Subassembly level qualification tests shall be conducted on those subassemblies that are subjected to environmental acceptance tests at the subassembly level. For other subassemblies, qualification tests are to be considered as optional unless specified otherwise in the contract. Functional or environmental qualification tests may be conducted at the subassembly level to detect material and workmanship defects, or to measure critical parameters, that cannot be accomplished satisfactorily at higher levels of assembly. When subassembly level qualification tests are planned, the subassemblies may be tested to similar requirements as components, or if more stringent requirements are used for acceptance test stress screening then the more stringent levels shall be the basis for the quaification tests. In general, all parts shall be qualified to maximum and minimum environmental levels well in excess of the levels predicted for their specific application in the space vehicle.

SECTION 7

ACCEPTANCE TESTS

7.1 SPACE VEHICLE ACCEPTANCE TESTS

The space vehicle acceptance test baseline consists of all the required tests specified in Table III. The test baseline shall be tailored for each program, giving consideration to both the required and optional tests; however., deviations from the baseline requirements for the required tests shall be approved by the contracting officer.

Table III. Space Vehicle Acceptance Tests

Test	Reference Paragraph	Suggested Sequence	Required (R) OR Optional (O)
Functional	7.1.1	1 (1)	R
EMC	7.1.2	2	O
Acoustic	7.1.3	5	R (2)
Vibration	7.1.4	5	O
Pyro Shock	7.1.5	4	O
Pressure	7.1.6	3, 6	R
Thermal Vacuum	7.1.7	8	R (3)
Thermal Cycling	7.1.8	7	O
Storage Tests	7.1.9	-	O
Special Tests	7.1	-	O

Notes:

(1) Electrical functional tests shall be conducted prior to and following each environmental test.

(2) Conduct vibration in place of acoustic test for vehicles of compact shape and with weight less than 180 kilograms.

(3) Requirements are modified if Thermal Cycling test 7.1.8 is conducted.

Additional special tests normally conducted by space vehicle programs include alignments, instrumentation calibrations, and measurements of mass properties, antenna patterns, and magnetic field. Since performance and accuracy requirements are generally program peculiar, and test methods are typically contractor peculiar, these tests are not included in this standard.

If the space vehicle is controlled by on-board data processing, the flight version of the computer software shall be resident in the space vehicle computer for these tests. The verification of the operational requirements shall be demonstrated in these tests to the extent practicable.

7.1.1 Functional Test, Space Vehicle Acceptance

7.1.1.1 Purpose. This test verifies that the electrical and mechanical performance of the space vehicle meets the performance requirements of the specifications and detects any anomalous condition.

7.1.1.2 Mechanical Functional Test. Same as the mechanical functional test for space vehicle qualification (6.2.1.2), except tests are only necessary at nominal operational conditions.

7.1.1.3 Electrical Functional Test. Same as the electrical functional test for space vehicle qualification (6.2.1.3), except tests are only necessary at nominal operational conditions.

The final ambient functional test conducted prior to shipment of the space vehicle to the launch base provides the data to be used as success criteria during launch base testing. For this reason, the functional test should be designed so that it can be duplicated, as nearly as possible, at the launch base. The results of all factory functional tests and of those conducted at the launch base shall be used for trend analysis.

7.1.1.4 Supplementary Requirements. Same as 6.2.1.4.

7.1.2 EMC Test, Space Vehicle Acceptance. Limited EMC acceptance testing shall be accomplished on space vehicles to check on marginal EMC compliance indicated during space vehicle EMC qualification testing and to verify that major changes have not occurred on successive production equipment. The limited tests shall include measurements of power bus ripple, peak transients, and monitoring of selected critical circuit parameters.

7.1.3 Acoustic Test, Space Vehicle Acceptance

7.1.3.1 Purpose. This test simulates the acoustic and vibration environment imposed on a space vehicle in flight in order to detect material and workmanship defects that might not be detected in a static test condition.

7.1.3.2 Test Description. Same as 6.2.3.2.

7.1.3.3 Test Levels and Duration. The acoustic spectrum shall represent the maximum predicted flight environment as defined in 3.20. The overall sound pressure level for acceptance testing shall not be less than 138 dB. The exposure time at full acceptance test level shall equal or exceed the maximum expected flight exposure time, but the test time shall not be less than 1 minute. Operating time should be divided approximately equally between redundant circuits. Where insufficient time is available at the full test level to test all redundant circuits, all functions, and all modes, extended testing at a level 6 dB lower shall be conducted as necessary to complete functional testing.

7.1.3.4 Supplementary Requirements. Same as 6.2.3.4.

7.1.4 Vibration Test, Space Vehicle Acceptance

7.1.4.1 Purpose. This test simulates the dynamic vibration environment imposed on a vehicle in flight in order to detect material and workmanship defects. This test is only used for small vehicles with weights under 150 kg and compact shapes.

7.1.4.2 Test Description. Same as 6.2.4.2.

7.1.4.3 Test Levels and Duration. Random vibration test levels shall be at the maximum predicted environmental levels as defined in 3.23. Test duration shall be equal to or exceed the expected flight exposure time to the maximum predicted environment but not less than 1 minute in each of three axes. Different axes may have different levels applied. Sinusoidal test levels shall be at the maximum predicted environmental levels as defined in 3.24, and test duration shall be equal to the expected flight exposure to the maximum predicted environment. Where insufficient time is available at the full test level to test all redundant circuits, all functions, and all modes, extended testing at a level 6 dB lower shall be conducted as necessary to complete functional testing.

7.1.4.4 Supplementary Requirements. Same as 6.2.4.4.

7.1.5 Pyro Shock Test, Space Vehicle Acceptance

7.1.5.1 Purpose. This test simulates the dynamic shock environment imposed on a space vehicle in flight in order to detect material and workmanship defects.

7.1.5.2 Test Description. Same as 6.2.5.2.

7.1.5.3 Test Levels and Duration. Pyrotechnic shock acceptance testing of space vehicles shall be required in those instances where the shock-producing mechanism can be readily refurbished for flight, as is often the case for explosive nuts, bolts, pinpullers, and clamps. One firing of those pyrotechnic devices causing significant shocks to critical and shock sensitive components shall be conducted. Firing of both primary and redundant pyros is required in the same relationship as they will be used in flight. However, where the pyrotechnic mechanism explosively severs structure by detonation of detonating fuse or shaped charge, such testing shall not be included or required. To aid in fault detection, the pyro shock test shall be conducted with subsystems operating and monitored to the maximum extent practical.

7.1.6 Pressure Test, Space Vehicle Acceptance

7.1.6.1 Purpose. This test demonstrates the capability of fluid subsystems to meet the flow, pressure, and leakage requirements specified in the space vehicle specification.

7.1.6.2 Test Description. Same as 6.2.6.2.

7.1.6.3 Test Levels and Duration. The leak checks shall be performed by pressurizing the subsystem to maximum operating pressure and holding at this pressure for a period commensurate with the leakage method being employed.

7.1.6.4 Supplementary Requirements. Same as 6.2.6.4.

7.1.7 Thermal Vacuum Test, Space Vehicle Acceptance

7.1.7.1 Purpose. This test detects material, process, and workmanship defects that would respond to thermal vacuum and thermal stress conditions and verifies thermal control.

7.1.7.2 Test Description. Same as 6.2.7.2.

7.1.7.3 Test Levels and Duration. Temperatures in various equipment areas shall be controlled by the external test environment and internal heating resulting from equipment operation so that the hot (or cold) temperature on at least one component in each equipment area equals the maximum (or minimum) predicted temperature as defined in 3.25. The temperature extremes shall be established by a survey of predicted temperatures in various equipment areas and may have to be adjusted to performance of the most sensitive components in a particular area. The pressure shall be maintained at 0.0133 pascals (0.0001 Torr) or less. Duration shall be sufficient to test all orbital operational conditions and all equipment functional modes including redundancy. Operating time should be divided approximately equally between redundant circuits. If the thermal cycling test (7.1.8) is not conducted, the thermal vacuum acceptance test shall include at least four complete hot-cold cycles at the maximum predicted orbital rate of temperature change and have at least an 8-hour soak at each temperature extreme of each cycle.

During one temperature cycle, thermal equilibrium shall be achieved at both hot and cold extremes to allow verification of performance of the thermostats, louvers, heat pipes, electric heaters, and the control authority of active thermal systems. Thermal equilibrium has been achieved when equipment temperature change is not more than 3 deg C/hour.

7.1.7.4 Supplementary Requirements. It may be necessary to force temperature extremes at certain locations by altering thermal boundary conditions locally or by altering the operational sequence to provide additional heating or cooling. Any special conditioning within the space vehicle shall generally be avoided. External baffling, shadowing, or heating shall be utilized to the extent possible.

7.1.8 Thermal Cycling Test, Space Vehicle Acceptance

7.1.8.1 Purpose. This test detects material, process, and workmanship defects by subjecting the space vehicle to a thermal cycling environment.

7.1.8.2 Test Description. Same as 6.2.9.2.

7.1.8.3 Test Levels and Duration. The space vehicle temperature range from hot to cold shall be the maximum possible within the constraints of the components acceptance temperatures. The minimum space vehicle temperature range shall be 50 deg C. Auxiliary heating and cooling may be used to protect selected temperature sensitive components. The average rate of change of temperature from one extreme to the other shall be as rapid as possible. Operating time should be divided approximately equally between redundant circuits. The minimum number of thermal cycles shall normally be 40.

7.1.8.4 Supplementary Requirements. If this test is implemented, only one thermal cycle is required in the thermal vacuum acceptance test specified in 7.1.7. Consideration should be given to conducting this test where considerable disassembly for rework of components has occurred or if maximum confidence in the system is required.

7.1.9 Storage Tests, Space Vehicle Acceptance. The space vehicle is sometimes stored at the contractor's facility for several months until needed for replenishment of an on-orbit operational system. Careful planning of environmental conditions and any testing during storage is necessary. Systematic checks of the space system health shall be made at periodic intervals while in storage. Age-sensitive parts shall be accounted for and reported on a schedule basis. Components having rotating elements may require periodic operation. If the space vehicle is removed from storage and design modifications made, consideration shall be given to conducting all of the required acceptance tests specified in Table III. Upon the vehicle's removal from storage for shipment to the launch site, a complete ambient functional acceptance test shall be conducted prior to shipment.

7.2 SUBSYSTEM ACCEPTANCE TESTS

Subsystem level acceptance tests are considered optional. These tests are often cost-effective because failures detected at this level usually are less costly to correct than are those detected at the space vehicle level. Acceptance tests should be conducted at the subsystem level where this level provides a more perceptive test than would be possible at the space vehicle level. The desirability of conducting these subsystem acceptance tests should be evaluated considering such factors as the relative accessibility of the subsystem and its component and the retest time at the vehicle level. The possible necessity to repeat other portions of vehicle level tests because of their invalidation due to a subsystem failure corrective action should also be considered. When subsystem level tests are performed, the test requirements shall be based on the space vehicle level test requirements.

7.3 COMPONENT ACCEPTANCE TESTS

The space vehicle component acceptance test baseline consists of all the required tests specified in Table IV. The test baseline shall be tailored for each program, giving consideration to both the required and optional tests; however,

TABLE IV. Component Acceptance Tests

TEST	REFERENCE PARAGRAPH	SUGGESTED SEQUENCE	ELECTRONIC OR ELECTRICAL EQUIPMENT	ANTENNAS	MOVING MECHANICAL ASSEMBLY	SOLAR PANEL	BATTERIES	VALVES	FLUID OR PROPULSION EQUIPMENT	PRESSURE VESSELS	THRUSTERS	THERMAL EQUIPMENT	OPTICAL EQUIPMENT
FUNCTIONAL	7.3.1	1(1)	R	R	R	R	R	R	R	R	R	R	R
THERMAL VACUUM	7.3.2	7	R(2)	O	R	O	R	R	R	O	R	R	R
THERMAL CYCLING	7.3.3	6	R	O	O	O	O	O	O	—	—	—	—
RANDOM VIBRATION	7.3.4	4	R	R(4)	R	—	O	R	R	O	R	R	R
ACOUSTIC	7.3.5	4	O	R(4)	—	O	—	—	—	—	—	—	—
PYRO SHOCK	7.3.6	3	O	—	—	—	—	—	—	—	—	—	O
PRESSURE	7.3.7	9	—	—	O(3)	—	R(3)	R	R	R	O	—	—
LEAK	7.3.8	2,5,10	R(3)	—	R(3)	—	R(3)	R	R	R	O	—	—
BURN-IN	7.3.9	8	R	—	O	—	—	R	—	—	R	—	—

LEGEND: R = REQUIRED
O = OPTIONAL TEST
— = NO REQUIREMENT

Notes: (1) Functional tests shall be conducted prior to and following environmental test
(2) Required only on unsealed units and on high power, RF equipment
(3) Required only on sealed or pressurized equipment
(4) Either random vibration or acoustic test required with the other optional

deviations from the baseline of required tests shall be approved by the contracting officer. Component acceptance tests shall normally be accomplished entirely at the component level. However, in certain circumstances, the required component acceptance tests may be conducted partially or entirely at the subsystem or space vehicle levels of assembly. Acceptance tests of components such as interconnect tubing, radio frequency circuits, and wiring harnesses are examples where at least some of the tests can usually be accomplished at higher levels of assembly.

Where components fall into two or more categories of Table IV, the required tests specified for each category shall be applied. For example, a star sensor may be considered to fit both "Electronic Equipment" and "Optical Equipment" categories. In this example, a thermal cycling test would be conducted since it is required for electronic equipment, even though there is no requirement for thermal cycling optics. Similarly, an electric motor-driven actuator fits both "Electrical Equipment" and "Moving Mechanical Assembly" categories. The former makes thermal cycling a required test, even though this test is optional for the moving mechanical assembly category.

7.3.1 Functional Performance Test, Component Acceptance

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

7.3.1.2 Test Description. Same as 6.4.1.2.

7.3.1.3 Supplementary Requirements. Same as 6.4.1.3.

7.3.2 Thermal Vacuum Test, Component Acceptance

7.3.2.1 Purpose. This test detects material and workmanship defects prior to installation into a space vehicle, by subjecting the unit to a thermal vacuum environment.

7.3.2.2 Test Description. Same as 6.4.2.2.

7.3.2.3 Test Levels/Duration

a. Pressure. The pressure shall be reduced from atmospheric to 0.0133 pascals (0.0001 Torr) or less.

b. Temperature. The high temperature shall be the maximum predicted but not less than +61 deg C, and the low temperature shall be the minimum predicted temperature but not higher than -24 deg C, except where the temperature extreme would result in physical deterioration of the materials in the component such as in tape recorders and batteries.

c. Duration. A minimum of one temperature cycle shall be used. The cycle shall have a 12-hour dwell at the high and at the low temperature levels during which the unit is turned off until the temperature has stabilized and then turned on.

7.3.2.4 Supplementary Requirements. Same as 6.4.2.4.

7.3.3 Thermal Cycling Test, Component Acceptance

7.3.3.1 Purpose. This test detects material and workmanship defects prior to installation of the component into a space vehicle, by subjecting the component to thermal cycling.

7.3.3.2 Test Description. Same as 6.4.3.2.

7.3.3.3 Test Levels and Duration

a. Pressure. Ambient pressure shall normally be used. When unsealed components are being tested, the chamber shall be flooded with dry air or nitrogen to preclude condensation on and within the component at low temperature. If convenient, this test may be performed in thermal vacuum and combined with the test of 7.3.2, provided that the temperature limits, number of cycles, rate of temperature change, and dwell times conform to this test.

b. Temperature. The high temperature shall be the maximum predicted but not less than +61 deg C, and the low temperature shall be the minimum predicted but not higher than -24 deg C, except where the temperature extreme would result in physical deterioration of the materials in the component such as in tape recorders and batteries.

c. Duration. The minimum number of temperature cycles shall be eight, of which the last four shall be failure free. Each cycle shall have a 1-hour minimum dwell at the high and at the low temperature levels during which the unit shall be turned off until the temperature stabilizes and then turned on. The dwell time at the high and low levels shall be long enough to obtain internal thermal equilibrium. The transitions between low and high temperatures shall be at an average rate of at least 1 deg C per minute.

7.3.3.4 Supplementary Requirements. Same as 6.4.3.4.

7.3.4 Random Vibration Test, Component Acceptance

7.3.4.1 Purpose. This test detects material and workmanship defects prior to installation of the component into a flight system, by subjecting the unit to a dynamic vibration environment.

7.3.4.2 Test Description. Same as 6.4.5.2.

7.3.4.3 Test Levels and Duration. The vibration test spectrum shall be equal to or greater than a smooth spectral representation for the maximum predicted environment as defined in 3.23. The minimum overall test level for components weighing 22.7 kg (50 lb) or less shall be 6 grms. The minimum test level for components weighing more than 22.7 kg (50 lb) shall be evaluated on an individual basis. The test duration in each of the three orthogonal axes shall equal or exceed the expected flight exposure time, but shall not be less than 1 minute per axis. Where insufficient time is available at the full test level to test all modes, extended testing at a level 6 dB lower shall be conducted as necessary to complete functional testing.

7.3.4.4 Supplementary Requirements. Same as 6.4.5.4.

7.3.5 Acoustic Test, Component Acceptance

7.3.5.1 Purpose. This test detects material and workmanship defects that might not be detected in a static test condition.

7.3.5.2 Test Description. Same as 6.4.6.2.

7.3.5.3 Test Levels and Duration. The component shall be exposed to sound pressure levels equal to the maximum predicted levels as defined in 3.21, but not less than 138 dB overall. The duration shall equal or exceed the expected flight exposure time to the maximum predicted environment but shall not be less than 1 minute.

7.3.5.4 Supplementary Requirements. Same as 6.4.6.4.

7.3.6 Pyro Shock Screening Test, Component Acceptance

7.3.6.1 Purpose. This test is intended to detect intermittents due to conducting particles in electronic components. It is effective for particle detection since shock can dislodge the particle and the extended duration vibration allows the particle to move to a position where a short may occur and permit detection. It may also detect intermittents due to cracked or loose dies in electronic parts that would not be found in normal acceptance tests.

7.3.6.2 Test Description. The screening test consists of pyro shock followed by random vibration testing. The component shall be mounted to a rigid fixture through the normal mounting points of the component. The component shall be tested in each direction of each of three mutually perpendicular axes (6 tests).

The component shall be electrically energized and functionally sequenced through all possible operating modes, including redundancy, during the testing. Circuits should be monitored for intermittents. A functional test shall be conducted before and after the pyro shock test and after the vibration screening test.

7.3.6.3 Test Levels and Duration. A pyro shock test shall be performed once in both directions of each of three mutually perpendicular axes at maximum predicted levels as defined in 3.22. The screening vibration test level shall be 3 dB below the acceptance test level specified in 7.3.4.3, but no lower than 4.5 g rms. Each test shall consist of 5 minutes dwell test followed by vibration bursts consisting of 10 seconds "on" and 10 seconds "off." The number of bursts shall be such that all circuits are monitored at least 10 times during the burst sequence with a minimum number of 20 bursts.

7.3.6.4 Supplementary Requirements. This test has proved to be very perceptive for detecting intermittents in guidance components and should be considered where the component is critical to mission success. Rather than basing the screening vibration test on the expected flight vibration spectrum shape, a special purpose spectrum may be developed. It is believed by some researchers that the screening vibration spectrum imposed on the component should cause circuit boards within to vibrate at a level of 5 - 10 g rms with a flat spectrum from 10 - 450 Hz. The required input spectrum may be defined from results of developmental vibration survey tests or analytic response predictions.

The screening test incurs exposure of the component to vibration in addition to that experienced in the acceptance vibration test. The additional exposure is defined by the screening spectrum based on the expected flight shape, or by the specially developed spectrum if that option is selected. In either case, the component qualification vibration levels and durations specified in this standard shall be assessed to assure that they provide a sufficient margin to safely conduct the screening test.

7.3.7 Pressure Test, Component Acceptance

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

7.3.7.2 Test Description. This test is the same as described in 6.4.10.2a and b, except that only one cycle shall be required, and test at elevated temperature is optional.

7.3.7.3 Test Levels. Same as 6.4.10.3.

7.3.7.4 Supplementary Requirements. Applicable safety standards shall be followed in conducting all tests.

7.3.8 Leakage Test, Component Acceptance

7.3.8.1 Purpose. This test demonstrates the capability of pressurized components to meet the leakage rate requirements specified in the component specifications.

7.3.8.2 Test Description and Alternatives. The component leak checks shall be made before and after exposure to each environmental acceptance test. The test method employed shall have sensitivity and accuracy consistent with the components specified maximum allowable leak rate. One of the methods given in 6.4.11.2 shall be used.

7.3.8.3 Test Levels and Duration. Same as 6.4.11.3.

7.3.9 Burn-In Test, Component Acceptance

7.3.9.1 Purpose. The purpose of the burn-in test shall be to detect material and workmanship defects which occur early in the component life.

7.3.9.2 Test Description. A modified thermal cycling test shall be used to accumulate the additional operational hours required for the burn-in test of electronic and electrical components. While the component is operating (power on) and while perceptive parameters are being monitored, the temperature of the unit shall be reduced to the specified low temperature level. The unit shall be operated at the low temperature level for 1 hour or longer. The unit temperature shall then be increased to the specified high temperature level and operated for 1 hour or longer. The temperature shall then be reduced to ambient to complete one cycle of the burn-in test. The transitions between low and high temperatures shall be at an average rate greater than 1 deg C per minute.

For valves, thrusters, and other items where the number of cycles of operation rather than hours of operation is a better method to ensure detecting infant mortality failures, functional cycling shall be conducted at ambient temperature. For thrusters, a cycle is a hot firing which includes a start, steady-state operation, and shutdown. For hot firings of thrusters utilizing hydrazine propellants, action shall be taken to assure that the flight valves are thoroughly cleaned of all traces of hydrazine following the test firings. Devices that have extremely limited life cycles such as positive expulsion tanks are excluded from burn-in test requirements.

7.3.9.3 Test Levels and Duration

a. Pressure. Ambient pressure should normally be used.

b. Temperature. For cycling of electronic and electrical components, the extreme temperatures specified in 7.3.3.3.b shall be used.

c. Duration. The total operating time for electronic and electrical component burn-in shall be 300 hours including the operating time during thermal cycling per 7.3.3. The minimum number of temperature cycles shall be 18 including those conducted during the thermal cycling acceptance test. Additional test time beyond that required for thermal cycling shall be conducted at either maximum or minimum temperature. The last 100 hours of the component burn-in test shall be free of failures. For valves, thrusters, and other components where functional cyclic testing is a better burn-in method, a minimum of 100 cycles shall be conducted.

d. Functional Duty Cycle. Functional tests shall be conducted at the start of this test to provide a baseline reference for determining if performance degradation occurs. The functional test shall be repeated after 150 hours of operation and during the last 2 hours of the thermal cycling test. Perceptive parameters for all circuits, including all redundancy, shall be monitored to the maximum extent possible during the entire test sequence. On-off cycling of the electronics component shall be conducted during the test to simulate operational usage.

7.3.9.4 Supplementary Requirements. The reduction of system level failures by burn-in at the component level has a favorable impact on costs and schedules by stabilizing the failure rate at or near its minimum and ensuring the highest probability of mission success.

7.4 SUBASSEMBLY LEVEL ACCEPTANCE TESTS

These tests are to be considered as optional unless specified otherwise in the contract. However, subassembly level acceptance tests are often cost-effective measures for reducing or avoiding failures in higher level tests and possibly in orbital operations. Acceptance test should be conducted at the subassembly level where this level provides a more perceptive test than would be possible at either the part or the component level. Functional or environmental acceptance tests are usually conducted at the subassembly level to detect material and workmanship defects, or to measure critical parameters, that cannot be accomplished satisfactorily at higher levels of assembly. When these acceptance tests are planned on subassemblies, they may be tested to similar requirements as components, or more stringent requirements for stress screening may be used.

SECTION 8

FLIGHT USE OF QUALIFICATION EQUIPMENT

Qualification tests are conducted to demonstrate that the design, manufacturing, and assembly have resulted in hardware conforming to specification requirements. The qualification tests required by this document incorporate the environmental design margin into the test levels to assure that flight units will meet the operational requirements for their service life. The vibration tests, acoustic tests, and thermal tests produce cyclic stresses that can encroach on the fatigue margins of interconnect wiring, solder joints, structural members and similar items in the qualification test units. If equipment that has been subjected to qualification testing is planned for subsequent flight use, it is possible that the remaining fatigue margins are so low as to present a high risk of failure during flight. This is primarily due to the use of high test levels and long test durations during the baseline qualification tests. Therefore, the actual vehicle used for the 6.2 vehicle qualification tests or the components used for the 6.4 component qualification tests may not be suitable for subsequent flight.

Nevertheless, initial program costs and schedule constraints may force the consideration of ways to make units used for qualification testing acceptable for flight. It should be recognized that the use of qualification items for flight always presents a higher risk than the use of standard acceptance tested items for flight. This risk may be reduced by various strategies such as reducing qualification test levels and durations to reduce the encroachment on fatigue and wearout margins. The strategy used should be based upon specific program considerations. One method has been to replace all components on the qualification vehicle with "new" components that have passed component acceptance tests (see 8.3). Another way was to lower the space vehicle qualification test levels and test duration to avoid excessive encroachment on margins (see 8.2). On some programs, one or more qualification components have been used as flight components (see 8.1). In such cases where program considerations are overriding, the contract may direct, or the contracting officer may approve, the use of qualification units for flight. Some of the strategies that have been used are presented in the following examples.

8.1 USE OF THE QUALIFICATION COMPONENTS FOR FLIGHT

When the qualification components are planned for flight use, the component qualification test program shall be modified from that specified in Section 6 to reduce cyclic stress levels. In addition, the component qualification testing shall be conducted on flight spares so that flight use is delayed or possibly never required. The flight space vehicle in which these qualification components are installed shall be acceptance tested in accordance with the requirements of 7.1. This space vehicle qualification would be based on the requirements of 6.2.

8.1.1 <u>Component Qualification Tests</u>. When the component qualification tests are conducted on a component intended for subsequent flight, the component acceptance tests required by this standard are waived, except for the burn-in acceptance test of 7.3.9, and only the qualification test baseline specified in 6.4 is required with the following exceptions:

a. For the component thermal vacuum test (6.4.2), the temperature extremes shall be 5 deg C beyond the minimum and maximum predicted temperatures.

b. For the component thermal cycling test (6.4.3), the temperature cycles shall be conducted at 5 deg C beyond the acceptance temperature extremes (7.3.3.b)

c. For the component vibration qualification test (6.4.5), the test level shall be 3 dB greater than the maximum predicted level but not less than 9 grms.

d. For the component acoustic qualification test (6.4.6), the test level shall be 3 dB greater than the maximum predicted level but not less than 141 dB overall.

e. For the component pyro shock test (6.4.7), the shock spectrum shall be 3 dB greater than the maximum predicted level.

f. For the component pressure test (6.4.10) only proof pressure tests per 6.4.10.3 a and b shall be conducted.

8.1.2 <u>Component Certification for Flight</u>. Upon completion of the modified qualification test program, the component test history shall be reviewed for excessive test time and potential fatigue type failures to determine if the unit is acceptable for flight or if refurbishment is required. Mission and safety critical qualification components should not be used for flight in systems where a redundant component is not provided.

8.2 USE OF THE FLIGHT VEHICLE FOR SPACE VEHICLE LEVEL QUALIFICATION

When the flight vehicle is also used for the vehicle level qualification tests, the space vehicle qualification test levels and durations shall be reduced as defined in 8.2.1. The components installed in this flight vehicles shall be acceptance tested in accordance with the requirements of 7.3. The component qualifications would be based on the requirements of 6.4.

8.2.1 Space Vehicle Qualification Tests. If the space vehicle qualification tests are to be combined with the flight vehicle acceptance tests, the space vehicle level acceptance tests required by this standard are weived and only the qualification test baseline in 6.2 is required with the following exceptions:

a. For the space vehicle acoustic qualification test (6.2.3), the test level shall be 3 dB greater than the maximum predicted level but not less than 141 dB overall. The duration of the test shall be the same as for the space vehicle acoustic acceptance test (7.1.3.3).

b. For the space vehicle vibration qualification test (6.2.4), the test levels shall produce vibration responses in the equipment which are 3 dB greater than the maximum predicted level. The duration of the test shall be the same as for the space vehicle vibration acceptance test (7.1.4.3).

c. For the space vehicle thermal vacuum qualification test (6.2.7), the number of hot-cold cycles shall be four and the temperature extremes shall be 5 deg C beyond the minimum and maximum predicted temperatures.

d. If the optional space vehicle thermal cycling test (6.2.9) is adopted as baseline, the minimum space vehicle temperature range shall be 60 deg C. The test should include 15 percent more thermal cycles than the space vehicle thermal cycling acceptance test, (7.1.8.3).

8.2.2 Space Vehicle Certification for Flight. Upon completion of the modified space vehicle qualification test program, the vehicle test history shall be reviewed for excessives test time and potential fatigue type failure to determine if the vehicle is acceptable for flight or if refurbishment is required. If significant modifications are incorporated or numerous components are refurbished or replaced with new components subsequent to qualification testing, the space vehicle acceptance baseline specified in 7.1 shall be required prior to launch certification.

8.3 USE OF THE QUALIFICATION VEHICLE FOR FLIGHT

When the space vehicle used for vehicle level qualification testing of 6.2 is planned for subsequent flight use, all components shall be replaced with "new" components that have passed the component acceptance tests. The space vehicle is certified for flight when it satisfactorily completes the vehicle level acceptance tests of Section 7.

8.4 OTHER

Various combinations of strategy may be considered depending on specific program considerations and the degree of risk deemed acceptable. For example, method 8.1.1 may be combined with a vehicle qualified at reduced levels per 8.2.1 or with the qualification vehicle per 8.3. In such cases, the provisions of both methods apply and the resultant risk would be increased appropriately.

SECTION 9

PRELAUNCH VALIDATION TESTS

9.1 GENERAL REQUIREMENTS

Prelaunch validation testing is accomplished at the factory and at the launch base, with the objective of demonstrating launch system readiness. The test series first establishes space vehicle baseline data in the factory preshipment test of 7.1.1.3. At the launch base, these tests verify that no changes have occurred in space vehicle parameters as a result of handling and transportation to the launch base. Further, a prelaunch validation test of the launch system is performed to verify that interfaces between the space vehicle and launch vehicle, and between these vehicles and the launch facilities, are within specified limits. To the greatest extent possible, the tests are to exercise all systems through every operational mode, in order to ensure that all mission requirements can be satisfied.

The space vehicle may be delivered as a complete vehicle or it may be assembled at the launch base with orbital injection subsystems, payload subsystems, propulsion subsystems, and other space vehicle subsystems being delivered separately. All factory test acceptance data shall accompany delivered flight hardware. In any case, the total launch system is first assembled at the launch site. The prelaunch validation tests are unique for each program in the extent of the operations necessary to ensure that all interfaces are properly tested. For programs which ship a complete space vehicle to the launch site, these tests primarily confirm vehicle performance, check for transportation damage, and demonstrate interface compatibility.

9.2 PRELAUNCH VALIDATION TEST FLOW

The test flow shall follow a progressive growth pattern to ensure proper operation of each space vehicle element prior to progressing to a higher level of assembly and test. In general, tests should follow the launch base buildup cycle. As successive systems or subsystems are verified, assembly proceeds to the completed space vehicle. Following testing of the space vehicle and its interfaces with the launch vehicle, these vehicles are electrically and mechanically mated and integrated into the launch system. Space vehicles employing a recoverable launch vehicle shall utilize a launch vehicle simulator to perform mechanical and electrical interface tests prior to integration with the launch vehicle. Following integration of the space vehicle and launch vehicle, functional tests of the space vehicle shall be conducted to ensure its proper operation following the handling operations involved in mating the vehicles. Space vehicle cleanliness shall be monitored by use of witness plates.

9.3 PRELAUNCH VALIDATION TEST CONFIGURATION

During each test, the space vehicle should be in flight configuration to the maximum extent possible, consistent with safety, control, and monitoring requirements. For programs utilizing a recoverable launch vehicle, the test configuration shall include any airborne support equipment required for the launch, ascent, and space vehicle deployment phases. This equipment shall be mechanically and electrically mated to the space vehicle in its launch configuration. Whenever possible, ground support equipment should have a floating point ground scheme that is connected to the flight vehicle single point ground. Isolation resistance tests shall be run to verify the correct grounding scheme prior to connection to the flight vehicle. This reduces the possibility of ground equipment interference with vehicle performance. All ground equipment shall be validated prior to being connected to any flight hardware, to preclude the possibility of faulty ground equipment causing damage to the flight hardware or inducing ambiguous or invalid data. Test provisions shall be made to verify integrity of circuits into which flight jumpers, arm plugs, or enable plugs have been inserted.

9.4 PRELAUNCH VALIDATION TEST DESCRIPTIONS

The prelaunch validation tests shall exercise and demonstrate satisfactory operation of the space vehicle through all of its mission phases, to the maximum extent practical. Test data shall be compared to corresponding data obtained in factory tests to identify trends which indicate performance degradation within specification limits. Each test procedure used shall include test limits and success criteria sufficient to permit a rapid determination as to whether or not processing and integration of the vehicle should continue. However, the final acceptance or rejection decision, in most tests, depends upon the results of post-test data analysis.

9.4.1 Functional Test. Electrical functional tests shall be conducted that duplicate, as nearly as possible, the factory functional tests of 7.1.1.2. Mechanical tests for leakage, valve and mechanism operability, and fairing clearance shall be conducted.

9.4.1.1 Simulators. Simulation devices shall be carefully controlled and shall be permitted only when there is no feasible alternative for conducting the test. When it is necessary to employ simulators in the conduct of prelaunch validation tests of the space vehicle, the interfaces disconnected in the subsequent replacement of the simulators with flight hardware shall be revalidated. Simulators shall be used for the validation of ground support equipment prior to connecting it to flight hardware.

9.4.1.2 Explosive Circuits. When explosive circuits are involved, approved simulation devices shall be used where appropriate. Before connection of pyro devices to their respective circuits, line continuity checks shall be made for the presence of the "Fire" signal at the squib connection when commanded. A line continuity stray voltage check shall be made immediately prior to the connection of any pyro device, and this check shall be repeated whenever that connection is opened and prior to reconnection.

9.4.2 Propulsion System Leak and Functional Test. A functional test of the space vehicle propulsion subsystem shall be conducted to verify, to the maximum practical extent, the proper operation of all components. Propulsion system leakage rates shall be verified to be within allowable limits.

9.4.3 Integrated System Tests. Total launch system readiness shall be demonstrated through an integrated, fully assembled launch systems test prior to flight. This test shall include an evaluation of radio frequency (rf) interference between system elements, electrical power interfaces, and the command and control subsystems. On a new space vehicle design or a significant design change to the telemetry, tracking, or receiving subsystem of an existing space vehicle, a test shall be run on the first vehicle to ensure nominal operation and that pyrotechnics (simulators) do not fire when the vehicle is subjected to the worst case range electromagnetic interference environment.

9.4.4 Compatibility Test, On-Orbit System

9.4.4.1 Purpose. This test validates the compatibility of the space vehicle and the on-orbit command and control network. For the purpose of establishing this testing baseline, it is assumed that the on-orbit command and control network is (or operationally interfaces with) the Air Force Satellite Control Facility (AFSCF). This test demonstrates the ability of the space vehicle, when in orbit, to properly respond to the AFSCF hardware, software, and operations team as specified in the AFSCF Orbital Support Plan. For programs which have a dedicated ground station, compatibility tests shall also be performed with the dedicated ground station.

9.4.4.2 Test Description. Facilities to perform on-orbit system compatibility tests exist at the Western Test Range (WTR) and the Eastern Test Range (ETR). At both locations, the AFSCF can command the space vehicle and process telemetry from the space vehicle as well as perform tracking and ranging, thus verifying the rf compatibility, the command software, and the telemetry modes. The tests include the following:

a. Verification of rf, analog, and digital compatibility of command, telemetry, and tracking links.

b. Verification of AFSCF capability to control the space vehicle using single, block, unsecure, and secure commands as required for on-orbit support.

c. Verification of AFSCF capability to process, display, and record space vehicle telemetry link or links as required for on-orbit support.

d. Verification of AFSCF capability to track the space vehicle using angle, doppler, and range tracking as required for on-orbit support.

9.4.4.3 Supplementary Requirements. This test should be run as soon as feasible after the space vehicle arrives at the launch base. The test is made with every space vehicle to verify system interface compatibility. The test shall be run using the software model versions that are integrated into the operational on-orbit software of the space vehicle under test. A preliminary compatibility test may be run prior to the arrival of the space vehicle at the launch base by the use of prototype subsystems, components, or simulators as required to prove the interface. Preliminary compatibility tests may be run using preliminary software. Normally, a preliminary compatibility test is run once for each series of space vehicles to check design compatibility, and is conducted well in advance of the first launch to permit orderly correction of hardware, software, and procedures as required. Changes in the interface from those tested in the preliminary test shall be checked by the compatibility tests conducted just prior to launch.

The following Data Item Descriptions are among those most frequently used in the Contract Data Requirements List (DD Form 1423) to establish detail requirements for the preparation of test plans, procedures, and reports.

DI-T-3701	System Test Plan
DI-T-3702	Contractor Test Plans/Procedures
DI-T-3703	Computed Program Coinfiguration Item (CPCI) Test Plan/Procedures (Computer Programs)
DI-T-3704	Electromagnetic Compatibility Test Plan
DI-T-3705	Structural Test Plans
DI-T-3707	General Test Plan/Procedures
DI-T-3708	Environmental Test Plans/Procedures
DI-T-3714	Acceptance Test Procedures
DI-T-3716	Final Test Report
DI-T-3717	Computer Program Configuration Item (CPCI) Development Test and Evaluation Test Report
DI-T-3718	Test Reports General

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MIL-STD-1540B(USAF) Test Requirements for Space Vehicles

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