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DEPARTMENT OF DEFENSE HANDBOOK

TEST REQUIREMENTS FOR LAUNCH, UPPER-STAGE, AND SPACE VEHICLES
Vol I : Baselines

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FOREWORD

1 .This Military Handbook is approved for use by all Departments and Agencies of the Department of Defense.

2. This handbook is for guidance only. This handbook cannot be sited as a requirement. If it is, the contractor does not have to comply.

3. The high reliability required of all launch and space equipment is achieved by the designs, design margins, and by the manufacturing process controls imposed at each and every level of assembly. The design and design margins should assure that the equipment is capable of performing in the launch and space environment. The manufacturing process controls are intended to assure that a known quality product is manufactured to meet the design requirements and that any changes required can be made based on a known baseline. Attention to every detail is required throughout development, manufacture, qualification, transportation, and preflight testing to assure successful operations of the launch and space equipment

4. For high-priority, long-life, complex space equipment, high reliability is usually achieved by strict compliance to the requirements and good practices that have historically resulted in successful missions. Programs for these types of space equipment are generally structured to provide extensive checks and balances, with detailed reviews of each step by independent personnel, to assure that no problem is overlooked. Particular attention is given in the design to eliminating single-point failure modes, wherever practicable. Special design analysis, special screening during manufacturing, and other quality provisions that will assure reliability, are implemented on any remaining single-point failure items to avoid latent defects. For these programs, a full qualification program is conducted on each level of assembly ranging from units, subsystems, space experiments, and on to each space vehicle involved.

5. Not all space equipment is high priority or long life. Many space programs are for a single mission that is of short duration, and the equipment may be relatively simple, or involve only one experiment. Expendable launch vehicles represent another program class where the mission duration is short, but weight constraints may eliminate redundancy. Regardless of the variations among launch and space programs, there is always a requirement for high reliability, because a flight failure is never cost effective. At the same time there is a constant drive to reduce the life cycle cost. Testing is a primary target for cost reduction because testing costs typically represent a high percentage of the total program cost. The real problem is to identify those cost-saving measures that are reasonable for each program and that will not increase risks in an unacceptable way. Although an in-flight failure can identify deficiencies in the test program used, mission success does not prove that the test program was optimum, cost effective, or even adequate. Nevertheless, mission success will typically be used to suggest that the testing conducted was excessive and should be reduced in the future. Even if the extent of testing was reasonable and justified, it will often be suggested that the costs were excessive!

6. This handbook, MIL-HDBK-340A, has been prepared to provide baseline test programs for launch and space vehicle equipment. The handbook is organized into two volumes. Volume I

presents a test baseline for high priority space and launch vehicles. The Volume I material previously was included in MIL-STD-1540C. For the convenience of contractors, the same weighting factors (3.5.12) have been retained to indicate the relative importance of the baseline requirements. Changing the weighting factor names to provide a softer guidance tone was rejected in the interest of those contractors who may want to extract sections of the baseline requirements for their own applications. Volume II of the handbook documents additional facets of information pertinent to the test baselines presented in Volume I and how the baseline requirements can be tailored for specific programs. The baselines in this handbook are intended to be used as guidance or as the starting point in developing a test program for a particular program. It should be emphasized that the information included is for general guidance and should not be followed if it does not accommodate the needs of a particular program.

7. Beneficial comments (recommendations, additions, or deletions) and any pertinent data which may be of use in improving this document should be addressed to: Space and Missile Systems Center, SMC/AXMP, 160 Skynet Street, Suite 2315, Los Angeles Air Force Base, El Segundo, CA 90245-4683 by using the Standardization Document Improvement Proposal (DD Form 1426) appearing at the end of this document or by letter.

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1.0 SCOPE

This handbook provides guidance for establishing uniform procedures for the control, determination, and documentation of product verification test requirements for launch, upper-stage, and space vehicles.

1.1 Purpose. Volume I of this handbook establishes environmental and structural ground test baselines for high priority launch vehicles, upper-stage vehicles, space vehicles, and for their sub-system and units. In addition, a uniform set of definitions of related terms is established. Volume II of this handbook addresses the tailoring of the testing baselines to meet the requirements of individual programs.

1.2 Application. Volume I of this handbook provides test baselines for use in developing or evaluating proposed test programs. Omissions or additions in proposed test programs when compared to the baselines in this handbook point to areas that may require justification. Volume II provides additional technical information for the testing requirements contained in Vol. I. In all cases, the testing requirements should be tailored to the specific program requirements. Therefore, the information provided is intended for guidance only and could change based on design complexity, design margins used, vulnerabilities, technical state-of-the-art, in-process controls, mission complexity, life cycle cost, number of vehicles involved, prior usage, and acceptable risk.

1.3 Test Categories. The tests are categorized as follows:

a. Development tests. Engineering characterization tests and tests to validate qualification and acceptance procedures (Section 5).

b. Qualification tests. Vehicle, subsystem, and unit levels (Section 6).

c. Acceptance tests. Vehicle, subsystem, and unit levels (Section 7).

d. Flightproof and protoqualification tests. Vehicle, subsystem, and unit levels (Section 8).

e. Prelaunch validation tests and follow-on operational tests and evaluations. Integrated system tests, initial operational tests and evaluations, and operational tests (Section 9).

2.0 APPLICABLE DOCUMENTS

2.1 General. The documents below are not necessarily all of the documents referenced herein, but are the ones that are needed in order to fully understand the information provided by this handbook.

2.2 Government Documents.

2.2.1 Specificaions, Standards, and Handbooks. The following standards and specifications form a part of this document to the extent specified herein. Unless otherwise specified, the issues of these documents are those listed in the issue of the Department of Defense Index of Specifications and Standards (DODISS) and supplement thereto, cited in the solicitation. When this handbook is used by acquisition, the application issue of the DoDISS must be sited in the solicitation.

Military Standards

MIL-STD-810	Environmental Test Methods and Engineering Guidelines.
MIL-STD-1522 (USAF)	Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems.
MIL-STD-1541 (USAF)	Electromagnetic Compatibility Requirements for Space Systems.
MIL-STD-1540D	Product Verification Requirements for Launch, Upper Stage, and Space Vehicles.

Military Handbooks

MIL-HDBK-340A Vol II	Test Requirements for Launch, Upper Stage, and Space Vehicles : Applications Guidelines.

(Unless otherwise indicated, copies of federal and military specifications, standards, and handbooks are available from the Standardization Document Order Desk, 700 Robbins Avenue, Building 4D, Philadelphia, PA 19111-5094.)

2.3 Order of Precedence. In the event of a conflict between the text of this document and the references sited herein, the text of this document takes precedence. Nothing in this document, however, supercedes applicable laws and regulations unless a specific exemption has been obtained.

3.0 DEFINITIONS

3.1 Item levels The categories of items in hierarchical order are defined in this section.

3.1.1 Part. A part is a single piece, or two or more joined pieces, which are not normally subject to disassembly without destruction or impairment of the design use. Examples: resistor, integrated circuit, relay, roller bearing.

3.1.2 Subassembly. A subassembly is a unit containing two or more parts which is capable of disassembly or part replacement. Examples: printed circuit board with parts installed, gear train.

3.1.3 Unit. A unit is a functional item that is viewed as a complete and separate entity for purposes of manufacturing, maintenance, or record keeping. Examples: hydraulic actuator, valve, battery, electrical harness, transmitter.

3.1.4 Subsystem. A subsystem is an assembly of functionally related units. It consists of two or more units and may include interconnection items such as cables or tubing, and the supporting structure to which they are mounted. Examples: electrical power, attitude control, telemetry, thermal control, and propulsion subsystems.

3.1.5 Vehicle. Any vehicle defined in this section may be termed expendable or recoverable, as appropriate.

3.1.5.1 Launch Vehicle. A launch vehicle is one or more of the lower stages of a flight vehicle capable of launching upper-stage vehicles and space vehicles, usually into a suborbital trajectory. A fairing to protect the space vehicle, and possibly the upper-stage vehicle, during the boost phase is typically considered to be part of the launch vehicle.

3.1.5.2 Upper-stage Vehicle. An upper-stage vehicle is one or more stages of a flight vehicle capable of injecting a space vehicle or vehicles into orbit from the suborbital trajectory that resulted from operation of a launch vehicle.

3.1.5.3 Space Experiment. A space experiment is usually part of the space vehicle payload and is therefore considered to be a lower level assembly of a space vehicle. However, a space experiment may be an integral part of a space vehicle, a payload that performs its mission while attached to a space vehicle, or even a payload that is carried by a host vehicle but performs some of its mission as a free-flyer. Whether complex space equipment is called a space experiment, a space instrument, or a space vehicle is discretionary and the nomenclature used should not affect the classification of the equipment or the requirements.

3.1.5.4 Space Vehicle. A space vehicle is an integrated set of subsystems and units 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 launch or upper-stage vehicle. The airborne support equipment (3.2.1), which is peculiar to programs utilizing a recoverable launch or upper-stage vehicle, is considered to be a part of the space vehicle.

3.1.5.5 Flight Vehicle. A flight vehicle is the combination of elements of the launch system that is flown; i.e., the launch vehicle(s), the upper-stage vehicle(s), and the space vehicle(s) to be sent to orbit.

3.1.6 System. A system is a 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. A system is typically defined by the System Program Office or the procurement agency responsible for its acquisition.

3.1.7 Combined Systems. Combined systems are interconnected systems that are required for program level operations or operational tests. The combined systems of interest are typically the launch system and the on-orbit system.

3.1.7.1 Launch System. A launch system is the composite of equipment, skills, and techniques capable of launching and boosting one or more space vehicles into orbit. The launch system includes the flight vehicle and related facilities, ground equipment, material, software, procedures, services, and personnel required for their operation.

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

3.2 SPECIAL ITEMS

3.2.1 Airborne Support Equipment (ASE). Airborne support equipment is the equipment installed in a flight vehicle to provide support functions and interfaces for the space or upper-stage vehicle during launch and orbital operations of the flight vehicle. This includes the hardware and software that provides the structural, electrical, electronic, and mechanical interfaces with the flight vehicle.

3.2.2 Critical Unit. A critical unit is one whose failure can affect the system operation sufficiently to cause the loss of the stated vehicle objectives, a partial loss of the mission, or is a unit whose proper performance is essential from a range safety standpoint.

3.2.3 Development Test Article. A development test article is a representative vehicle, subsystem, or unit dedicated to provide design and test information. The information may be used to check the validity of analytic techniques and assumed design parameters, to uncover unexpected response characteristics, to evaluate design changes, to determine interface compatibility, to prove qualification and acceptance test procedures and techniques, or to determine if the equipment meets its performance specifications. Development test articles include engineering test models, thermal models, and structural static and dynamic models.

3.2.4 Explosive-ordnance Device. An explosive-ordnance device is a device that contains or is operated by explosives. A cartridge-actuated device, one type of explosive-ordnance device, is a mechanism that employs the energy produced by an explosive charge to perform or initiate a mechanical action.

3.2.5 Moving Mechanical Assembly (MMA). A moving mechanical assembly is a mechanical or electromechanical device that controls the movement of one mechanical part of a vehicle relative to another part. Examples: gimbals, actuators, despin and separation mechanisms, valves, pumps, motors, latches, clutches, springs, dampers, bearings.

3.2.6 Reusable Item. A reusable item is a unit, subsystem, or vehicle that is to be used for multiple missions. The service life (3.5.~~6~~7) of reusable hardware includes all planned reuses, refurbishment, and retesting.

3.3 ENVIRONMENTS

The complex flight environment involves a combination of conditions that are usually resolved into individual test environments. Each test environment should be based on actual flight data, scaled if necessary for differences in parameters, or if more reliable, by analytical prediction or a combination of analysis and flight data. The flight data may be from the current flight system, or from other flight systems if configuration variations are accounted for and properly scaled. The individual environments, which may be involved in qualification and acceptance, are described in this section.

3.3.1 Maximum and Minimum Expected Temperatures. The maximum and minimum expected temperatures are the highest and lowest temperatures that an item can experience during its service life (3.5.~~6~~ 7), including all operational modes. These temperatures are established from analytically determined extreme temperatures by adding a thermal uncertainty margin, discussed below. The analytically determined extreme temperatures are predicted from thermal models using applicable effects of worst-case combinations of equipment operation, internal heating, vehicle orientation, solar radiation, eclipse conditions, ascent heating, descent heating, and degradation of thermal surfaces during the service life.

For space and upper-stage vehicles, the analytical model is validated using results from a vehicle thermal balance test involving operational modes which include the worst-case hot and cold conditions. The thermal uncertainty margin is applied to the analytically determined extreme temperatures, even after validation by a thermal balance test. The thermal uncertainty margin accounts for uncertainties in parameters such as complicated view factors, surface properties, radiation environment, joint conduction, and unrealistic aspects of ground test simulation. The margins vary depending on whether passive or active thermal control techniques are used. Examples of each type, for purposes of uncertainty margin to be applied, appear in Table I. The margins to be applied are addressed in the following subparagraphs.

3.3.1.1 Margins for Passive Thermal Control Subsystems. For units that have no thermal control or have only passive thermal control, the recommended minimum thermal uncertainty margin is 17°C prior to achieving a validated analytical model. For space and upper-stage vehicles, the uncertainty margin may be reduced to 11°C after the analytical model is validated using results from a vehicle thermal balance test. To avoid significant weight and power increases of the power subsystem due to additional hardware or increased heater size, the uncertainty margin of 17°C may be reduced to 11°C.

For units that have large uncertainties in operational or environmental conditions or that do not require thermal balance testing, the thermal uncertainty margin may be greater than those stated above. Examples of these units for a launch vehicle are a vehicle heat shield, external insulation, and units within the aft skirt.

For passive cryogenic subsystems operating below minus 70°C, the thermal uncertainty margin may be reduced as presented in Table II. In addition, the following thermal-uncertainty heat-load margins are recommended: 50% in the conceptual phase, 45% for preliminary design, 35% for critical design review, and 30% for qualification.

3.3.1.2 Margins for Active Thermal Control Subsystems. For thermal designs in which temperatures are actively controlled, a heat-load margin of 25% may be used in lieu of the thermal margins specified in 3.3.1.1. This margin is applicable at the condition that imposes the maximum and minimum expected temperatures. For example, for heaters regulated by a mechanical thermostat or electronic controller, a 25% heater capacity margin may be used in lieu of the thermal margins at the minimum expected temperature and at minimum bus voltage, which translates into a duty cycle of no more than 80% under these cold conditions. Where an 11°C addition in the analytically determined extreme temperatures would cause the temperature of any part of the actively-controlled unit to exceed an acceptable temperature limit, a control-authority margin in excess of 25% should be demonstrated.

For designs in which the temperatures are actively controlled to below minus 70°C by expendable coolants or refrigerators, the thermal uncertainty heat-load margin of 25% should be increased in the early phases of the development. For these cases, the following thermal-uncertainty heat-load margins are recommended: 50% in the conceptual phase, 45% for preliminary design, 35%for the critical design review, and 30% for qualification.

3.3.2 Statistical Estimates of Vibration, Acoustic, and Shock Environments. Qualification and acceptance tests for vibration, acoustic, and shock environments are based upon statistically expected spectral levels. The level of the extreme expected environment, used for qualification testing, is that not exceeded on at least 99% of flights, estimated with 90% confidence (P99/90 level). The level of the maximum expected environment, used for acceptance testing, is that not exceeded on at least 95% of flights, estimated with 50% confidence (P95/50 level). These statistical estimates are made assuming a lognormal flight-to-flight variability having a standard deviation of 3 dB, unless a different assumption can be justified. As a result, the P95/50 level estimate is 5 dB above the estimated mean (namely, the average of the logarithmic values of the spectral levels of data from all available flights). When data from N flights are used for the estimate, the P99/90 estimate in dB is $2.0 + 3.9/N^{1/2}$ above the P95/50 estimate. When data from only one flight are available, those data are assumed to represent the mean and so the P95/50 is 5 dB higher and the P99/90 level is 11 dB higher.

When ground testing produces the realistic flight environment (for example, engine operation or activation of explosive ordnance), the statistical distribution can be determined using the test data, providing data from a sufficient number of tests are available. The P99/90 and P95/50 levels are then determined from the derived distribution.

Extreme and maximum expected spectra should be specified for zones of the launch, upper-stage, and space vehicles to allow for repositioning of units within their zones without changing the expected environment. Particular spectra can be developed for specific units.

Table I. Categorization of Passive and Active Thermal Control Subsystems.

Passive	Active
Constant-conductance or diode heat pipes.	Variable-conductance heat pipes.
Hardwired heaters (fixed or variable-resistance, such as auto-trace or positive-temperature-coefficient thermistors).	Heat pumps and refrigerators. Stored-coolant subsystems.
Thermal storage devices (phase-change or sensible heat).	Resistance heater with commandable or mechanical or electronic controller.
Thermal insulation(multi-layer insulation, foams, or discrete shields).	Capillary-pumped loops. Pumped fluid loops.
Radiators (fixed, articulated, or deployable) with louvers or pinwheels.	Thermoelectric cooler.
Surface finishes (coatings, paints, treatments, second-surface mirrors).	

Table II. Thermal Uncertainty Margins For Passive Cryogenic Subsystems.

Predicted Temperature (°C)	Thermal Uncertainty Margin (°C)	
	Pre-validation	Post-validation
Above -70	17	11
-70 to -87	16	10
-88 to -105	15	9
-106 to -123	14	8
-124 to -141	13	7
-142 to -159	11	6
-160 to -177	9	5
-178 to -195	8	4
-196 to -213	6	3
-214 to -232	4	2
Below -232	2	1

3.3.3 **Fatigue Equivalent Duration.** For a time-varying flight acoustic or vibration environment, the fatigue equivalent duration is the time duration, at the maximum environment achieved during that flight, that would produce the same fatigue damage potential. For a given flight trajectory, the equivalent duration can be assumed to be independent of the maximum environment achieved during any particular flight. The fatigue damage potential is taken to be proportional to the fourth power of amplitude, unless another basis can be justified.

3.3.4 **Extreme and Maximum Expected Acoustic Environment.** The acoustic environment for an exterior or interior zone of a vehicle results from propulsive and aerodynamic excitations. The acoustic environment is expressed by a 1/3-octave-band pressure spectrum in dB (reference 20 micropascal) for center frequencies spanning a range of at least 31 to 10,000 Hz. For a time-varying environment, the acoustic spectrum used for test purposes is the envelope of the spectra for each of a series of 1-second time segments overlapped by at least 50%. Longer time segments may be used only if it is shown that significant smoothing of the time-dependent characteristics of the spectra (that is, large bias error) does not occur. The extreme and maximum expected acoustic environments (P99/90 and P95/50 acoustic spectra, respectively, per 3.3.2) are the bases for qualification and acceptance test spectra, respectively, subject to workmanship-based minimum spectra. The associated duration is the fatigue equivalent duration in flight (3.3.3).

3.3.5 **Extreme and Maximum Expected Random Vibration Environment.** The random vibration environment induced at the structural attachments of units is due to the direct or indirect action of the acoustic and aerodynamic excitations, to roughness in combustion or burning processes, and to machinery induced random disturbances. The random vibration environment is expressed as an acceleration spectral density in g^2/H (commonly termed power spectral density or simply PSD) over the frequency range of at least 20 to 2000 Hz. For a time-varying environment, the PSD used for test purposes is the envelope of the spectra for each of a series of 1-second time segments overlapped by at least 50%. Longer time segments may be used only if it is shown that significant smoothing of the time-dependent characteristics of the spectra (that is, large bias error) does not occur. Also, the resolution bandwidth is to be no greater than 1/6 octave, but need not be less than 5 Hz. The extreme and maximum expected vibration environments (P99/90 and P95/50 PSDs, respectively, per 3.3.2) are the bases for the qualification and acceptance test spectra, respectively, subject to workmanship-based minimum spectra. The associated duration is the fatigue equivalent duration in flight (3.3.3).

3.3.6 **Extreme and Maximum Expected Sinusoidal Vibration Environment.** The sinusoidal vibration induced at the structural attachments of units may be due to periodic excitations from rotating machinery and from instability involving pogo (interaction of structural and propulsion dynamics), flutter (interaction of structural dynamics and aerodynamics), or combustion. Periodic excitations may also occur during ground transportation. The sinusoidal vibration environment is expressed as an acceleration amplitude in g over the frequency range for which amplitudes are significant.

Namely, those whose acceleration amplitude exceeds 0.016 times the frequency in Hz. This is based on a response velocity amplitude of 1.27 meters per second (50 inches per second) when the vibration is applied to a single-degree-of-freedom system having a Q of 50. The resolution bandwidth should be no greater than 10% of the lowest frequency sinusoidal component present. The extreme and maximum expected sinusoidal vibration environments (P99/90 and P95/50 amplitude spectra, respectively, per 3.3.2) are the basis for qualification and acceptance spectra, respectively. The associated duration is the fatigue equivalent duration (3.3.3), including flight and transportation.

When combined sinusoidal and random vibration during service life (3.5.7) can be more severe than sinusoidal and random vibration considered separately, the combined environment is applicable.

3.3.7 Extreme and Maximum Expected Shock Environment. Shock transients result from the sudden application or release of loads associated with deployment, separation, impact, and release events. Such events often employ explosive-ordnance devices resulting in generation of a pyroshock environment, characterized by a high-frequency acceleration transient which decays typically within 5 to 15 milliseconds. The shock environment is expressed as the derived shock response spectrum in g, based upon the maximum absolute acceleration or the equivalent static acceleration induced in an ideal, viscously damped, single-degree-of-freedom system. Its natural frequency should span the range from at least 100 Hz to 10,000 Hz for pyroshock or comparable shock disturbances, at intervals of no greater than 1/6 octave, and for a resonant amplification (Q) of 10. The extreme and maximum expected shock environments (P99/90 and P95/50 shock response spectra, respectively, per 3.3.2) are the bases for qualification and acceptance test spectra, respectively.

3.4 STRUCTURAL TERMS

3.4.1 Burst Factor. The burst factor is a multiplying factor applied to the maximum expected operating pressure to obtain the design burst pressure. Burst factor is synonymous with ultimate pressure factor.

3.4.2 Design Burst Pressure. The design burst pressure is a test pressure that pressurized components must withstand without rupture in the applicable operating environments. It is equal to the product of the maximum expected operating pressure and a burst factor.

3.4.3 Design Factor of Safety. The design factor of safety is a multiplying factor used in the design analysis to account for uncertainties such as material properties, design procedures, and manufacturing procedures. The design factor of safety is often called the design safety factor, factor of safety, or, simply, the safety factor. In general, two types of design factors of safety are specified: design yield factor of safety and design ultimate factor of safety.

3.4.4 Design Ultimate Load. The design ultimate load is a load, or combinations of loads, that the structure must withstand without rupture or collapse in the applicable operating environments. It is equal to the product of the limit load and the design ultimate factor of safety.

3.4.5 Design Yield Load. The design yield load is a load, or combinations of loads, that a structure must withstand without experiencing detrimental deformation in the applicable operating environments. It is equal to the product of the limit load and the design yield factor of safety.

3.4.6 Limit Load. A limit load is the highest load, or combinations of loads, that may be applied to a structure during its service life (3.5.7) and acting in association with the applicable operating environments produces a design or extreme loading condition for that structure. When a statistical estimate is applicable, the limit load is that load not expected to be exceeded on at least 99% of flights, estimated with 90% confidence.

3.4.7 Maximum Expected Operating Pressure (MEOP). The MEOP is the highest gage pressure that an item in a pressurized subsystem is required to experience during its service life (3.5.7) and retain its functionality, in association with its applicable operating environments. The MEOP is synonymous with limit pressure or maximum operating pressure (MOP) or maximum working pressure (MWP). Included are the effects of maximum ullage pressure, fluid head due to vehicle quasi-steady and dynamic accelerations, waterhammer, slosh, pressure transients and oscillations, temperature, and operating variability of regulators or relief valves.

3.4.8 Maximum Predicted Acceleration. The maximum predicted acceleration (its extreme value), defined for structural loads analysis and test purposes, is the highest acceleration determined from the combined effects of quasi-steady acceleration, the vibroacoustic environment, and the dynamic response to such significant transient flight events as liftoff; engine ignitions and shutdowns; transonic and maximum dynamic pressure traversal; gust; and vehicle separation. The frequency range of concern is usually limited to below 50 Hz for structural loads resulting from the noted transient events, and to below 300 Hz for secondary structural loads resulting from the vibration and acoustic environments. Maximum accelerations are predicted for each of three mutually perpendicular axes in both positive and negative directions. When a statistical estimate is applicable, the maximum predicted acceleration is at least that acceleration not expected to be exceeded on 99% of flights, estimated with 90% confidence (P99/90).

3.4.9 Operational Deflections. Operational deflections are the deflections imposed on a structure during operation (for example, by engine thrust-vector gimballing, thermal differentials, flight accelerations, and mechanical vibration).

3.4.10 Pressure Component. A pressure component is a unit in a pressurized subsystem, other than a pressure vessel, that is structurally designed largely by the acting pressure. Examples are lines, tubes, fittings, valves, bellows, hoses, regulators, pumps, and accumulators.

3.4.11 Pressure Vessel. A pressure vessel is a structural component whose primary purpose is to store pressurized fluids and one or more of the following apply:

a. Contains stored energy of 19,310 joules (14,240 foot-pounds) or greater based on adiabatic expansion of a perfect gas.

b. Contains a gas or liquid that would endanger personnel or equipment or create a mishap (accident) if released.

c. May experience a design limit pressure greater than 690 kilopascals (100 psi).

3.4.12 Pressurized Structure. A pressurized structure is a structure designed to sustain both internal pressure and vehicle structural loads. A main propellent tank of a launch vehicle is a typical example.

3.4.13 Pressurized Subsystem. A pressurized subsystem consists of pressure vessels (3.4.11) or pressurized structures (3.4.12), or both, and pressure components (3.4.10). Excluded are electrical or other control units required for subsystem operation.

3.4.14 Proof Factor. The proof factor is a multiplying factor applied to the limit load, or maximum expected operating pressure, to obtain the proof load or proof pressure for use in a proof test.

3.4.15 Proof Test. A proof test is an acceptance test used to prove the structural integrity of a unit or assembly, or to establish maximum possible flaw sizes for safe-life determination. The proof test gives evidence of satisfactory workmanship and material quality by requiring the absence of failure or detrimental deformation. The proof test load and pressure compensate for the difference in material properties between test and design temperature, if applicable.

3.4.16 Structural Component. A mechanical unit is considered to be a structural component if its primary function is to sustain load or maintain alignment.

3.5 OTHER DEFINITIONS

3.5.1 Ambient Environment. The ambient environment for a ground test is defined as normal room conditions with temperature of 23 ± 10°C (73 ± 18°F), atmospheric pressure of 101 +2/-23 kilopascals (29.9 +0.6/-6.8 in. Hg), and relative humidity of 50 ± 30%.

3.5.2 Contamination Tolerance Level. The contamination tolerance level is the value of contaminant particle size, or level of contamination, at which a specified performance, reliability, or life expectancy of the item is adversely affected.

3.5.3. 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 electronis 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.

3.5.4 Operational Modes. The operational modes for a unit, assembly, subsystem, or system include all combinations of operational configurations or conditions that can occur during its service life (3.5.~~6~~ 7). Examples: power condition, command mode, readout mode, attitude control mode, redundancy management mode, safe mode, and spinning or despun condition.

3.5.5 "Required," "Other," and "Not-required" Tests. "Required," "other," and "not-required" tests for each vehicle category are indicated by an "R," "O," and "-," respectively, in Tables VII, IX, X, XII, and XIII. The following basis has been used

a. "Required" tests are the baseline tests that are required because they are generally effective.

b. "Other" tests are those that are usually ineffective and have a low probability of being required. Such tests must be evaluated on a case-by-case basis. If the evaluation shows that an "other" test is effective, it becomes a "required" test for that case.

c. "Not-required" tests are generally ineffective and are not required.

3.5.6 Qualification Margin. An environmental qualification margin is the increase in an environmental condition, over that expected during service life (3.5.7), including acceptance testing, to demonstrate that adequate ruggedness exists in the design and in its implementation. A margin may include an increase in level or range, an increase in duration or cycles of exposure, as well as any other appropriate increase in severity. Environmental qualification margins are intended to demonstrate the ability to satisfy all of the following on a single qualification item:

a. Be tolerant of differences in ruggedness and functionality of flight items relative to the qualification item, due to reasonable variations in parts, material properties, dimensions, processes, and manufacturing.

b. Be immune to excessive degradation (such as fatigue, wear, loss of material properties or functionality) after enduring a specified maximum of acceptance testing prior to operational use of a flight item.

c. Meet requirements under extreme conditions of flight, which when expressed statistically are the P99/90 estimates (3.3.2, 3.4.8).

3.5.7 **Service Life.** The service life of an item starts at the completion of fabrication and continues through all acceptance testing, handling, storage, transportation, prelaunch testing, all phases of launch, orbital operations, disposal, reentry or recovery from orbit, refurbishment, retesting, and reuse that may be required or specified.

3.5.8 **Temperature Stabilization.** For thermal cycle and thermal vacuum testing, temperature stabilization for a unit is achieved when the unit baseplate is within the allowed test tolerance on the specified test temperature (4.6), and the rate of change of temperature has been less than 3^{m}C per hour for 30 minutes. For steady-state thermal balance testing, temperature stabilization is achieved when the unit having the largest thermal time constant is within 3^{m}C of its steady state value, as determined by numerical extrapolation of test temperatures, and the rate of change is less than 1^{m}C per hour.

3.5.9 **Test Discrepancy.** A test discrepancy is a functional or structural anomaly that occurs during testing, which may reveal itself as a deviation from specification requirements for the test item. A test discrepancy may be a momentary, unrepeatable anomaly; or it may be a permanent failure to respond in the predicted manner to a specified combination of test environment and functional test stimuli. Test discrepancies include those associated with functional performance, premature operation, failure to operate or cease operation at the prescribed time, and others that are unique to the item.

A test discrepancy may be due to a failure of the test item, or may be due to some unintended cause such as from the test setup, test instrumentation, supplied power, test procedures, or computer software used.

3.5.10 Test Item Failure. A failure of a test item is defined as a test discrepancy that is due to a design, workmanship, or quality deficiency in the item being tested. Any test discrepancy is considered to be a failure of the test item unless it can be determined to have been due to some unintended cause (3.5.9).

3.5.11 Thermal Soak Duration. The thermal soak duration of a unit at the hot or cold extreme of a thermal cycle is the time that the unit is operating and its baseplate is continuously maintained within the allowed tolerance of the specified test temperature.

3.5.12 Weighting Factors. Even for the required tests, not all of the testing requirements have an equal importance or equal weight. To avoid overstating testing requirements, and hence avoid excessive costs, various categories of weighting factors are associated with the requirements. The primary weighting factors that are incorporated in the Handbook are:

a. Weighting factor “a”. “Will” designates the most important weighting level, the prime requirements.

b. Weighting factor “b”. “Will, where practicable” designates requirements or practices at the second highest weighting level. Alternative requirements or practices may be used for specific applications. When the use of the alternative is substantiated by documented technical trade studies.

c. Weighting factor “c”. “Should” designates the third weighting level.

d. Weighting factor “d”. “May” designates the lowest weighting level. In some cases, these “may” requirements are stated as examples of acceptable practices.

SECTION 4.

GENERAL REQUIREMENTS

This section addresses general requirements applicable to all test categories. Included are the validation process, testing philosophy, propulsion equipment tests, firmware tests, inspections, test condition tolerances, test plans and procedures, retest, and documentation.

4.1 VALIDATION PROCESS.

The development of space and launch systems involves the design, manufacture, test, and integration of very complex equipment and software by a large number of people in many independent organizations. Successful space and launch systems rely heavily on a rigorous prelaunch verification process because repairs after launch are practically impossible and failures tend to be extremely expensive. Documentation required for the validation process also serves as a way to coordinate the many people and activities involved in a space system acquisition. The system engineering process allocates system performance requirements and tolerances to items at lower levels of assembly. A service life cycle profile is developed for each item to document the various phases that the item might encounter in its life. These phases start with release from manufacturing and progress through operational use to removal from service. The phases may include handling, shipping, storage, system integration, prelaunch validation, launch, injection into orbit, operational use, standby use, and return from orbit. For each phase, the possible configurations and operational modes are noted and the possible environments and variations in the environmental range throughout the service life is determined for each item. The life cycle profile data is used in the design development process where hardware and software is designed to perform the allocated functions. The designs, allocated functions, life-cycle profile data, and the associated manufacturing processes and procedures need to be documented in sufficient detail so as to provide a baseline for system validation and a baseline for any subsequent corrective actions or changes that may be needed. By this point, most space programs should have prepared a validation plan to identify all necessary steps to be taken to assure a successful mission. Essentially a list of all required functions, interfaces and other requirements will be prepared. For each item on this requirements list, the method of verifying the requirement will be indicated, as well as the level of assembly involved. Typical validation methods include analysis, inspection, similarity, test, demonstration, and simulation. Note that this handbook does not address many of these validation methods. The handbook focuses only on those requirements where tests are needed to verify the design and manufacturing (qualification tests), and the tests needed for product verification (acceptance).

4.2 TESTING PHILOSOPHY

The complete test program for launch vehicles, upper-stage vehicles, and space vehicles encompasses development, qualification, acceptance, prelaunch validation, and follow-on operational tests and evaluations. Test methods, environments, and measured parameters should 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. The test program encompasses the testing of progressively more complex assemblies of hardware and computer software. Design suitability should be demonstrated in the earlier development tests prior to testing the next more complex assemblies or combinations in the progression and prior to the start of formal qualification testing. All qualification testing for an item should be completed, and consequential design improvements incorporated, prior to the initiation of flight hardware acceptance testing for that item. In general, hardware items subjected to qualification tests are themselves not eligible for flight, since there has been no demonstration of remaining life from fatigue and wear standpoints. Section 8 describes higher risk, alternative strategies which may be used to tailor a qualification test program. The integrated system prelaunch validation tests, described in Section 9, are intended to be combined with or incorporated with the Step 3 integrated system tests, and the Step 4 and 5 operational tests that include the applicable ground equipment and associated computer software.

Environments other than those specified in this Handbook can be sufficiently stressful as to warrant additional qualification and possibly acceptance testing. These include environments such as nuclear and electromagnetic radiation, as well as climatic conditions not specified such as lightning.

The environmental tests specified are intended to be imposed sequentially, rather than in combination. Nevertheless, features of the hardware design or of the service environments may warrant the imposition of combined environments in some tests. Examples: combined temperature, acceleration, and vibration when testing units employing elastomeric isolators in their design; and combined shock, vibration, and pressure when testing pressurized components. In formulating the test requirements in these situations, a logical combination of environmental factors should be imposed to enhance test effectiveness.

4.3 PROPULSION EQUIPMENT TESTS

In general, tests of solid rocket motors and tests of liquid rocket engines are not addressed in this Handbook. However, units which comprise a vehicle propulsion subsystem, including units which are integral to or mounted on a motor or engine, are covered by this Handbook in that they will be qualified and acceptance tested to the applicable unit requirements specified herein. Testing of a unit on an engine during the engine acceptance test firing may be substituted for part of the unit level acceptance test if it can be established that the environments and duration meet the intent of the individual acceptance test criteria, or if such units are not amenable to testing individually. Environmental testing of thrusters (such as staging rockets, retro-motors, and attitude control thrusters) will meet the applicable unit requirements of this Handbook.

4.3.1 Engine Line Replaceable Unit (LRU) Acceptance Testing. An engine LRU is an engine unit which may be removed from an engine and replaced by a new unit without requiring re-acceptance test firing of the engine with the new unit. If the unit being replaced was included in an engine acceptance test firing as part of its acceptance test, then the replacement unit will either be subjected to such a test on an engine, or will undergo equivalent unit level acceptance testing. Equivalent testing will consider all appropriate environments such as temperature, vibration, pressure, vacuum, and chemical. Testing will demonstrate functionality of the unit under conditions similar to those achieved in the engine acceptance test firing and flight.

4.3.2 Engine Line Replaceable Unit (LRU) Qualification Testing. All engine LRUs will be qualified at a unit level to the requirements of this Handbook.

4.4 FIRMWARE TESTS

Firmware is the combination of a hardware device and computer instructions or computer data that reside as read-only software on the hardware device. The software cannot be readily modified under program control. Firmware that falls under the intent and purpose of a Commercial Off the Shelf item (COTS) should be tested as COTS. Firmware that is not COTS should be tested as a development item subject to the test requirements of this document. The software element of firmware should be tested as software, and the hardware element of firmware should be tested as hardware.

4.5 INSPECTIONS

All units and higher levels of assembly should be inspected to identify discrepancies before and after testing, including tests performed at the launch site. The inspections of flight hardware should not entail the removal of unit covers nor any disassembly, unless specifically called out in the test procedures. Included should be applicable checks of finish, identification markings, and cleanliness. Weight, dimensions, fastener tightness torques and breakaway forces and torques should be measured, as applicable, to determine compliance with specifications.

4.6 TEST CONDITION TOLERANCES.

Unless stated otherwise, the specified test parameters should be assumed to include the maximum allowable test tolerances listed in Table III. For conditions outside the ranges specified, the tolerances should be appropriate for the purpose of the test.

4.7 TEST PLANS AND PROCEDURES

The test plans and procedures should be documented in sufficient detail to provide the framework for identifying and interrelating all of the individual tests and test procedures needed.

4.7.1 Test Plans. The test plans should provide a general description of each test planned and the conditions of the tests. The test plans should be based upon a function-by-function mission analysis and any specified testing requirements. To the degree practicable, tests should be planned and executed to fulfill test objectives from development through operations. Test objectives should be planned to verify compliance with the design and specified requirements of the items involved, including interfaces. The test plans should incorporate by reference, or directly document, the following:

a. A brief background of the applicable project and descriptions of the test items covered (such as the systems, vehicles, and subtier equipment).

b. The overall test philosophy, testing approach, and test objective for each item, including any special tailoring or interpretation of design and testing requirements.

c. The allocation of requirements to appropriate testable levels of assembly. Usually this is a reference to a requirements traceability matrix listing all design requirements and indicating a cross reference to a verification method and to the applicable assembly level.

d. The identification of separate environmental test zones (such as the engine, fairing, or payload).

Table III. Maximum Allowable Test Tolerances.

Test Parameters			Test Tolerance
Temperature			
-54°C to +100°C			± 3°C
Relative Humidity			± 5%
Acceleration			+10/-0%
Static Load and Pressure			+ 5/-0%
Atmospheric Pressure			
Above	133 pascals	(>1 Torr)	±10%
133 to	0.133 pascals	(1 Torr to 0.001 Torr)	±25%
Below	0.133 pascal	(<0.001 Torr)	±80%
Test Time Duration			+10/-0%
Vibration Frequency			± 2%
Sinusoidal Vibration Amplitude			±10%
Random Vibration Power Spectral Density			
<u>Frequency Range</u>	<u>Maximum Control Bandwidth</u>		
20 to 100 Hz	10 Hz		± 1.5 dB
100 to 1000 Hz	10% of midband frequency		± 1.5 dB
1000 to 2000 Hz	100 Hz		± 3.0 dB
Overall			± 1.0 dB
Note: Control bandwidths may be combined for tolerance evaluation purposes. The statistical degrees of freedom will be at least 100.			
Sound Pressure Levels			
<u>1/3-Octave Midband Frequencies</u>			
31.5 to 40 Hz			± 5.0 dB
50 to 2000 Hz			± 3.0 dB
2500 to 10000 Hz			± 5.0 dB
Overall			± 1.5 dB
Note: The statistical degrees of freedom will be at least 100.			
Shock Response Spectrum (Peak Absolute Acceleration, Q = 10)			
<u>Natural Frequencies Spaced at 1/6-Octave Intervals</u>			

At or below 3000 Hz	± 6.0 dB
Above 3000 Hz	+ 9.0/-6.0 dB
Note: At least 50% of the spectrum values will be greater than the nominal test specification.	

e. The identification of separate states or modes where the configuration or environmental levels may be different (such as during testing, launch, upper-stage transfer, on-orbit, eclipse, or reentry).

f. The environmental specifications or life-cycle environmental profiles for each of the environmental test zones.

g. Required special test equipment, facilities, interfaces, and downtime requirements.

h. Required test tools and test beds including the qualification testing planned for the test tools and test beds to demonstrate that they represent an operational system environment and verify that simulated interfaces are correct.

i. Standards to be used for the recording of test data on computer compatible electronic media, such as disks or magnetic tape, to facilitate automated accumulation and sorting of data.

j. The review and approval process to be followed for test plans and procedures, and for making changes to approved test plans and procedures.

k. Overall schedule of tests showing conformance with the program schedules including the scheduled availability of test articles, test facilities, special test equipment, and procedures.

4.7.2 Test Procedures. Tests should be conducted using documented test procedures prepared in accordance with the test objectives in the approved test plans. The test objectives, testing criteria, and pass-fail criteria should be stated clearly in the test procedures. The test procedures should cover all operations in enough detail so that there is no doubt as to the execution of any step. Test objectives and criteria should be stated clearly to relate to design or operations specifications. Where appropriate, minimum requirements for valid data and pass-fail criteria should be provided at the procedure step level. Traceability should be provided from the specifications or requirements to the test procedures. Where practicable, the individual procedure step that satisfies the requirement should be identified. The test procedure for each item should include, as a minimum, descriptions of the following:

a. Criteria, objectives, assumptions, and constraints.

b. Test setup.

c. Initialization requirements.

d. Input data.

e. Test instrumentation.

f. Expected intermediate test results.

g. Requirements for recording output data.

h. Expected output data.

i. Minimum requirements for valid data to consider the test successful.

j. Pass-fail criteria for evaluating results.

k. Safety considerations and hazardous conditions.

4.8 RETEST

Whenever the design of hardware is changed, the hardware involved should be retested, as necessary, and all documentation pertinent to the changes should be revised. When retesting a redesigned item, limited testing may be satisfactory as long as it is adequate to verify the redesign, to confirm that the redesign did not negate prior testing, and to show that no new problems have been introduced. However, care must be exercised with this limited retesting concept since even small changes can potentially affect the item in unexpected ways.

Retesting may also be necessary if a test discrepancy (3.5.9) occurs while performing any of the required testing steps. In that case, conducting a proper failure analysis plays an important part in determining the type and degree of retesting. The failure analysis should include the determination of whether a failure occurred, the cause of the failure, the symptoms of the failure, and isolation of the failure to the smallest replaceable item.

4.8.1 Retest During Qualification or Acceptance. If a test discrepancy occurs during qualification or acceptance testing, the test may be continued without corrective action if the discrepant item or software coding does not affect the validity of test data obtained by the continuation of testing. Otherwise the test should be interrupted and the discrepancy verified. To the extent practicable, the test configuration should not be modified until the cause of the discrepancy has been isolated and verified. If the discrepancy is caused by the test setup, test software, or a failure in the test equipment, the test being conducted at the time of the discrepancy may be continued after the cause is removed and repairs are completed, as long as the discrepancy did not overstress the item under test. If the discrepancy is caused by a failure of the item under test, the preliminary failure analysis and appropriate corrective action should normally be completed and properly documented before testing is resumed. Retesting may be required to establish a basis for determining compliance of a test item to a specification or requirement, and may be required to assess the readiness of test items for integrated system testing.

4.8.2 Retest During Prelaunch Validation. If a discrepancy occurs during prelaunch validation testing (integrated system testing), it should be documented for later evaluation. The test director is responsible for assessing the effect of the discrepancy to determine whether the discrepancy has jeopardized the probable success of the remainder of the test. The test director may decide to continue or halt the test. If continued, the test starts at the test procedure step designated by the test director. The integrated system testing should be continued, where practicable, to conserve time-critical operational resources. When the discrepancy has been corrected or explained, retesting may be required.

4.8.3 Retest During Operational Tests and Evaluations. If a discrepancy occurs during operational tests and evaluations, it should be documented for later evaluation. The operating agency is responsible for assessing the effect of the discrepancy to determine whether the discrepancy has jeopardized the probable success of the remainder of the test. The operating agency is also responsible for determining the degree of retesting required.

4.9 DOCUMENTATION

4.9.1 Test Documentation Files. The test plans and procedures (4.7), including a list of test equipment, calibration dates and accuracy, computer software, test data, test log, test results and conclusions, problems or deficiencies, pertinent analyses, and resolutions should be documented and maintained. The test documentation files should be maintained by the applicable contractors for the duration of their contracts.

4.9.2 Test Data. Pertinent test data should be maintained in a quantitative form to permit the evaluation of performance under the various specified test conditions; pass or fail statements alone may be insufficient. The test data should also be compared across major test sequences for trends or evidence of anomalous behavior. To the extent practicable, all relevant test measurements and the environmental conditions imposed on the units should be recorded on computer compatible electronic media, such as disks, magnetic tape, or by other suitable means to facilitate automated accumulation and sorting of data for the critical test parameters. These records are intended to be an accumulation of trend data and critical test parameters that should be examined for out of tolerance values and for characteristic signatures during transient and mode switching. For development and qualification tests, a summary of the test results should be documented in test reports. The test report should detail the degree of success in meeting the test objectives of the approved test plans and should document the test results, deficiencies, problems encountered, and problem resolutions.

4.9.3 Test Log. Formal test conduct will be documented in a test log. The test log should identify the personnel involved and be time-tagged to permit a reconstruction of test events such as start time, stop time, anomalies, and any periods of interruption.

4.10 TEST EVALUATION TEAM. As a cost containment and quality assurance measure, it is strongly recommended that a high level, joint contractor and customer, test evaluation team be established for each of the major vehicle level tests, particularly the mode survey qualification test, the thermal balance qualification test, the subsystem structural static load qualification test, and major separation qualificaation tests. The test conductor would typically be the chairman of the Test Evaluation Team. Other members should be provided by the design organizations that will use the results, by safety, and by quality assurance. The customer should provide a qualified technical representative to the team to perform the usual customer monitoring of the test and to facilitate the timely approval of technically justified deviations from the test requirements. The members of the team would typically change for each test. The purpose of each Test Evaluation Team is to:

a. Evaluate the adequacy of the test configuration, including instrumentation, prior to the start of testing.

b. Provide guidance in resolving technical problems and issues arising during testing.

c. Expedite the disposition of discrepancies and the approval of corrective actions, if required.

d. Verify adequacy of the test results.

e. Recommend tear-down of the test setup.

During the mode survey test, the Test Evaluation Team may deviate from the completeness requirement for modes judged to be unimportant, and from the orthogonality standard for problem modes. Such deviations require adequate technical justification and typically the concurrence of the designated representative of the customer.

SECTION 5.

DEVELOPMENT TESTS

5.1 GENERAL

Development tests, or engineering tests, should be conducted as required to:

a. Validate new design concepts or the application of proven concepts and techniques to a new configuration.

b. Assist in the evolution of designs from the conceptual phase to the operational phase.

c. Reduce the risk involved in committing designs to the fabrication of qualification and flight hardware.

d. Validate qualification and acceptance test procedures.

e. Investigate problems or concerns that arise after successful qualification.

Requirements for development testing therefore depend upon the maturity of the subsystems and units used and upon the operational requirements of the specific program. An objective of development testing is to identify problems early in their design evolution so that any required corrective actions can be taken prior to starting formal qualification testing. Development tests should be used to confirm structural and 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 exceeds the design limits to identify marginal capabilities and marginal design features. Comprehensive development testing is an especially important ingredient to mission success in programs that plan to use qualification items for flight, including those that allow a reduction in the qualification test levels and durations. Development tests may be conducted on breadboard equipment, prototype hardware, or the development test vehicle equipment.

Development tests may be conducted at in-plant test facilities, which may include subcontractor's facilities, at a government approved test bed, or at any other appropriate test facility. However, when performed at a government facility, that facility may require approval of the test plans and procedures. Internal contractor documentation of development test plans, test procedures, and test results are normally used unless stated otherwise by contract.

The development test requirements are necessarily unique to each new launch vehicle, upper-stage vehicle, and space vehicle. The following provide guidelines for conducting appropriate development tests when their need has been established.

5.2 PART, MATERIAL, AND PROCESS DEVELOPMENT TESTS AND EVALUATIONS

Part, material, and process development tests and evaluations are conducted to demonstrate the feasibility of using certain items or processes in the implementation of a design. These development tests and evaluations may be conducted to assess design alternatives, manufacturing alternatives, and to evaluate tradeoffs to best achieve the development objectives. Development tests and evaluations are required for new types of parts, materials, and processes; to assure proper application of parts, materials, and processes in the design; and to develop acceptance criteria for these items to avoid assembling defective units.

Material characterization testing under simulated environmental conditions is normally conducted for composite laminate, insulations, seals, fluid lines, and items not well characterized for their intended use.

5.3 SUBASSEMBLY DEVELOPMENT TESTS, IN-PROCESS TESTS AND INSPECTIONS

Subassemblies are subjected to development tests and evaluations as required to minimize design risk, to demonstrate manufacturing feasibility, and to assess the design and manufacturing alternatives and trade-offs required to best achieve the development objectives. Tests are conducted as required to develop in-process manufacturing tests, inspections, and acceptance criteria for the items to avoid assembling defective hardware items.

5.4 UNIT DEVELOPMENT TESTS

Units are subjected to development tests and evaluations as may be required to minimize design risk, to demonstrate manufacturing feasibility, to establish packaging designs, to demonstrate electrical and mechanical performance, and to demonstrate the capability to withstand environmental stress including storage, transportation, extreme combined environments, and launch base operations. Temperature cycling and random vibration testing at levels beyond the qualification requirements should be conducted to further increase confidence in the design and identify the weakest elements. New designs should be characterized across worst-case voltage, frequency, and temperature variations at the breadboard level. Functional tests of prototype units in thermal and vibration environments are normally conducted. Development tests of deployables, of thrust vector controls, and of the attitude control subsystem are normally conducted. Life tests of critical items that may have a wearout failure mode, such as moving mechanical assemblies, should also be conducted. Vibration resonance searches of a unit should be conducted to correlate with a mathematical model and to support design margin or failure evaluations. Development tests and evaluations of vibration and shock test fixtures should be conducted prior to first use to prevent inadvertent overtesting or undertesting, including avoidance of excessive cross-axis responses. These development tests of fixtures should result in the design of shock and vibration test fixtures that can be used during unit qualification and acceptance tests. When it is not practicable to use fixtures of the same design for unit qualification and acceptance tests, evaluation surveys should be performed on each fixture design to assure that the unit responses are within allowable margins.

5.4.1 Structural Composite Development Tests. Development tests will be conducted on structural components made of advanced composites or bonded materials, such as payload adapters, payload fairings, motor cases, and composite-overwrapped pressure vessels.

If appropriate, testing should include:

a. Static load or burst testing to validate the ultimate structural capabilities.

b. Damage tolerance testing to define acceptance criteria.

c. Acoustic transmission loss test for composite fairings.

5.4.2 Thermal Development Tests. For critical electrical and electronic units designed to operate in a vacuum environment less than 0.133 pascal (0.001 Torr), thermal mapping for known boundary conditions should be performed in the vacuum environment to verify the internal unit thermal analysis, and to provide data for thermal mathematical model correlation. Once correlated, the thermal model is used to demonstrate that critical part temperature limits, consistent with reliability requirements and performance, are not exceeded. When electrical and electronic packaging is not accomplished in accordance with known and accepted techniques relative to the interconnect subsystem, 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. Heat transport capacity tests may be required for constant and variable conductance heat pipes at the unit level to demonstrate compliance with 3.3.1. Thermal conductance tests may be performed to verify conductivity across items such as vibration isolators, thermal isolators, cabling, and any other potentially significant heat conduction path.

5.4.3 Shock and Vibration Isolator Development Tests. When a unit is to be mounted on shock or vibration isolators whose performance is not well known, development testing should be conducted to verify their suitability. The isolators should be exposed to the various induced environments (for example, temperature and chemical environments) to verify retention of isolator performance (especially resonant frequencies and amplifications) and to verify that the isolators have adequate service life (3.5.7). The unit or a rigid simulator with proper mass properties (mass, center of gravity, mass moments of inertia), should be tested on its isolators in each of three orthogonal axes, and, if necessary, in each of three rotational axes. Responses at all corners of the unit should be determined to evaluate isolator effectiveness and, when applicable, to establish the criteria for unit acceptance testing without isolators (7.4.4). When multiple units are supported by a vibration isolated panel, responses at all units should be measured to account for the contribution of panel vibration modes.

5.5 VEHICLE AND SUBSYSTEM DEVELOPMENT TESTS

Vehicles and subsystems are subjected to development tests and evaluations using structural and thermal development models as may be required 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 subsystems. Vehicle level development testing also provides an opportunity to develop handling and operating procedures as well as to characterize interfaces and interactions.

5.5.1 Mechanical Fit Development Tests. For launch, upper-stage, and space vehicles, a mechanical fit, assembly, and operational interface test with the facilities at the launch or test site is recommended. Flight-weight hardware should be used if practicable; 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.

5.5.2 Mode Survey Development Tests. In advance of the qualification mode survey test (6.2.10), a development mode survey test (or modal survey) should be conducted at the vehicle or subsystem level when uncertainty in analytically predicted structural dynamic characteristics is judged to be excessive for purposes of structural or control subsystem design, and an early identification of problem areas is desired. The test article may be full-scale or subscale; for a large vehicle, such as a launch vehicle, a subscale model is often used. Such a development test does not replace a modal survey required for vehicle qualification, unless the test also meets the requirements in 6.2.10.

5.5.3 Structural Development Tests. For structures having redundant load paths, structural tests may be required to verify the stiffness properties and to measure member loads, stress distributions, and deflections. The stiffness data are of particular interest where nonlinear structural behavior exists that is not fully exercised in a mode survey test (5.5.2, 6.2.10). 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 may be used to experimentally verify the loads transformation matrix. Deflection data may be also used to experimentally verify the appropriate deflection transformation matrix. These matrices may be used, in conjunction with the dynamic model, to calculate loads such as axial forces, bending moments, shears, and torsional moments, and various stresses and deflections, which can be converted into design load and clearance margins for the vehicle. This development test does not replace the structural static load test that is required for subsystem qualification (6.3.1); however, the two tests may be incorporated into a single test sequence that encompasses the requirements of both tests, provided that the test article is flight-like, the manufacturing log is up-to-date, and the test plan is prepared according to the qualification requirements.

5.5.4 Acoustic and Shock Development Tests. Since high-frequency vibration and shock responses are difficult to predict by analytical techniques, acoustic and shock development testing of the launch, upper-stage, and space vehicles may be necessary to verify the adequacy of the dynamic design criteria for units. Vehicle units that are not installed at the time of the test should be dynamically simulated with respect to mass, center of gravity, moments of inertia, interface stiffness, and geometric characteristics. For the acoustic test, the vehicle is normally exposed to the qualification acoustic levels in an acoustic chamber. For the shock test, all explosive-ordnance devices and other mechanisms capable of imparting a significant shock to the vehicle should be operated. Where practicable, the shock test should involve physical separation of elements being deployed or released. When a significant shock is expected from subsystems not on board the vehicle under test (such as when a fairing separation causes shock responses on an upper stage under test), the adaptor subsystem or suitable simulation will be attached and appropriate explosive-ordnance devices or other means used to simulate the shock imposed. The pyroshock environment may vary significantly between ordnance activations. Therefore, the statistical basis given in 3.3.2 will be used for estimating maximum expected and extreme spectra. Multiple activations of ordnance devices may be used to provide data for better-substantiated estimates.

5.5.5 Thermal Balance Development Tests. A thermal balance development test may be necessary to verify the analytical modeling of launch, upper-stage, or space vehicles, and to verify the unit thermal design criteria. For vehicles in which thermally induced structural distortions are critical to mission success, the thermal balance test also evaluates alignment concerns. The test vehicle should consist of a thermally equivalent structure with addition of equipment panels, thermal control insulation, finishes, and thermally equivalent models of electrical, electronic, pneumatic, and mechanical units. Testing should be conducted in a space simulation test chamber capable of simulating the ascent, transfer orbit, and orbital thermal-vacuum conditions as may be appropriate.

5.5.6 Transportation and Handling Development Tests. The handling and transport of launch, upper-stage, and space vehicles, or their subtier elements, is normally conducted so as to result in dynamic environments well below those expected for launch and flight. However, since these environments are difficult to predict, it is often necessary to conduct a development test of potentially significant handling and transportation configurations to determine worst-case dynamic inputs. Such a test should use a development model of the item or a simulator which has at least the proper mass properties, instrumented to measure responses of the item. In particular, a drop test representative of a maximum credible operational occurrence should be conducted to demonstrate protection of the item in the handling apparatus and shipping container. The data should be sufficient to determine whether the environments are benign relative to the design requirements, or to provide a basis for an analysis to demonstrate lack of damage, or to augment qualification and acceptance testing, if necessary.

5.5.7 Wind-tunnel Development Tests. Flight vehicle aerodynamic and aerothermal data are needed to establish that the vehicles survive flight, and function properly under the imposed loads. For flight vehicles with a new or significantly changed aerodynamic design, the following wind-tunnel tests ~~will~~ should be conducted:

a. Force and Moment Tests. These tests provide the resultant aerodynamic forces and moments acting on the vehicle during the high-dynamic-pressure region of flight. Data from these tests are used in both structural and control subsystem design and in trajectory analysis.

b. Steady-State Pressure Tests. These tests determine the spatial distribution of the steady-state component of the pressures imposed on the vehicle's external surfaces during the high-dynamic-pressure region of flight. These data are used to obtain the axial airload distributions which are used to evaluate the static-elastic characteristics of the vehicle. These data are also used in compartment venting analyses to determine burst and collapse pressures imposed on the vehicle structure. The design and testing of the payload fairing structure are particularly dependent upon high-quality definition of these pressures.

c. Aerodynamic Heating Tests. These tests determine the heating effects due to fin and fuselage junctures, drag (friction), angle of attack, flow transition, shock wave impingement, proximity effects for multibody vehicles, and surface discontinuities.

d. Base Heating Tests. These tests determine the heating effects due to thermal radiation, multiplume recirculation convection, plume-induced flow separation on the vehicle body, and the base flow field.

e. Thruster Plume-impingement Heating Tests. These tests determine the heating effects due to impingement of the thruster plumes.

f. Transonic and Supersonic Buffet and Aerodynamic Noise Tests. These tests define the spatial distribution of the unsteady or fluctuating component of the pressures imposed on the vehicle external surfaces during the high-dynamic-pressure region of flight. These data are used to obtain the dynamic airloads acting to excite the various structural modes of the vehicle and are used in aeroelastic, flutter, and vibroacoustic analyses.

g. Ground-wind-induced Oscillation Tests. These tests define the resultant forces and moments acting on the vehicle prior to launch when it is exposed to the ground-wind environment. Flexible models or elastically-mounted rigid models are used to simulate at least the first cantilever bending mode of the vehicle. Nearby structures or terrain, which may influence the flow around the vehicle, should also be simulated.

SECTION 6.

QUALIFICATION TESTS

6.1 GENERAL QUALIFICATION TEST REQUIREMENTS

Qualification tests will be conducted to demonstrate that the design, manufacturing process, and acceptance program produce mission items that meet specification requirements. In addition, the qualification tests will validate the planned acceptance program including test techniques, procedures, equipment, instrumentation, and software. The qualification test baseline will be tailored for each program. Each type of flight item that is to be acceptance tested will undergo a corresponding qualification test, except for certain structural items as identified herein.

In general, a single qualification test specimen of a given design will be exposed to all applicable environmental tests. The use of multiple qualification test specimens may be required for one-time-use devices (such as explosive ordnance or solid-propellant rocket motors). Aside from such cases, multiple qualification specimens of a given design may be used to enhance confidence in the qualification process, but are not required by this Handbook.

6.1.1 Qualification Hardware. The hardware subjected to qualification testing will be produced from the same drawings, using the same materials, tooling, manufacturing process, and level of personnel competency as used for flight hardware. Ideally, a qualification item would be randomly selected from a group of production items. A vehicle or subsystem qualification test article should be fabricated using qualification units to the maximum extent practicable. Modifications are permitted if required to accommodate benign changes that may be necessary to conduct the test. These changes include adding instrumentation to record functional parameters, test control limits, or design parameters for engineering evaluation. When structural items are rebuilt or reinforced to meet specific strength or rigidity requirements, all modifications will be structurally identical to the changes incorporated in flight articles. The only testing required prior to the start of qualification testing of an item is the wear-in (7.4.10) to achieve a smooth, consistent, and controlled operation of the item (such as for moving mechanical assemblies, valves, and thrusters).

6.1.2 Qualification Test Levels and Durations. To demonstrate margin, the qualification environmental conditions will stress the qualification hardware to more severe conditions than the maximum conditions that might occur during service life (3.5.7), including not only flight, but also a maximum time or number of cycles that can be accumulated in acceptance testing and retesting. Qualification testing, however, should not create conditions that exceed applicable design safety margins or cause unrealistic modes of failure. If the equipment is to be used by more than one program, or in different vehicle locations, the qualification test conditions should envelope those of the various programs or vehicle locations involved. Typical qualification margins on the flight and acceptance test levels and durations are summarized in Table IV.

Table IV. Typical Qualification Test Level Margins and Durations.

Test	Units	Vehicle
Shock*	6 dB above maximum expected environment, 3 times in both directions of 3 axes	1 activation of all shock-producing events; 2 additional activations of controlling events (6.2.3.3)
Acoustic*	6 dB above acceptance for 3 minutes	6 dB above acceptance for 2 minutes
Vibration*	6 dB above acceptance for 3 minutes, each of 3 axes	6 dB above acceptance for 2 minutes, each of 3 axes
Thermal Vacuum (Tables V, VI)	10°C beyond acceptance temperatures for 6 cycles	10°C beyond acceptance temperatures for 13 cycles
Combined Thermal Vacuum and Thermal Cycle (Tables V, VI)	10°C beyond acceptance temperatures for 25 thermal vacuum cycles and 53 1/2 thermal cycles	10°C beyond acceptance temperatures for 3 thermal vacuum cycles and 10 thermal cycles
Static Load	1.25 times the limit load for unmanned flight or 1.4 times the limit load for manned flight, for a duration close to actual flight loading times	Same as for unit, but only tested at subsystem level

*Accelerated testing per 6.1.4.1 is assumed. Also, durations generally are longer for environments dominated by liquid engine or solid motor operation.

6.1.3 <u>Thermal Vacuum and Thermal Cycle Tests</u>. The required number of qualification thermal cycles is intended to demonstrate a capability for 4 times the thermal fatigue potentially expended in service life (3.5.7). The requirements stated assume that such fatigue is dominated by acceptance testing, and that the flight and other aspects (such as transportation) do not impose significant additional fatigue. It is further assumed that units, due to acceptance retesting, may be subjected to as many as 2 times the number of thermal cycles specified for a basic test. If a different limit on number of cycles is used, the required number of qualification cycles will be changed per note 5 of Table VI. No allowance is made for acceptance retest of vehicles. For both thermal cycle and thermal vacuum tests, the temperature ranges in Table V are the basis for the number of cycles in Table VI for qualification and acceptance testing.

In instances where these baseline requirements are not appropriate due to the temperature range, acceptance retest allowance, or significance of the mission or other service, the qualification number of cycles will be modified per note 5 of Table VI. Also, the maximum allowable number of acceptance thermal cycles can be extended after the original qualification by performing the required additional testing on the qualification test item necessary to meet the requirement in note 5 of Table VI.

Electrical and electronic units, or units containing electrical and electronic elements, are subjected to multiple thermal vacuum cycles and thermal cycles for the purpose of uncovering workmanship deficiencies by a process known as "environmental stress screening." Such screening is intended to identify defects that may result in early failures. Therefore the number of cycles imposed is generally unrelated to mission thermal cycles. For units not containing electrical or electronic elements, only thermal vacuum testing is required and the number of thermal cycles are considerably reduced (Table VI, 6.4.3.4, and 7.4.3.3).

6.1.4 Acoustic and Vibration Qualification. For the acoustic and vibration environments, the qualification tests are designed to demonstrate the ability of the test item to endure both of the following:

a. The acceptance test spectrum (7.1.2 or 7.1.3) for 4 times the maximum allowable duration of acceptance testing of flight items, including any retesting.

b. The extreme expected spectrum (6 dB higher than acceptance, unless a lesser margin can be justified per 3.3.2) for a duration of 4 times the fatigue equivalent duration in flight (3.3.3), but for not less than 1 minute. The maximum allowable duration of acceptance testing can be extended after the original qualification by performing additional testing on the qualification test item. If one or more electrical or electronic units are involved, this additional acoustic or vibration testing will be followed by at least 1.5 thermal cycles or 1.5 thermal vacuum cycles.

Either the approach described in 6.1.4.1 or 6.1.4.2 may be selected for conduct of the qualification testing.

Table V. Temperature Ranges for Thermal Cycle (TC) and Thermal Vacuum (TV) Tests.

Required Testing	Unit	Vehicle	
	TC & TV	TC	TV
Acceptance (DT_A)	105°C[1]	≥ 50°C	note 3
Qualification (DT_Q)	125°C[2]	≥ 70°C[2]	note 4

Notes:
1. Recommended, but reduced if impracticable or increased if necessary to encompass operational temperatures (7.1.1).
2. $DT_Q = DT_A + 20°C$.
3. Governed by the unit that first reaches its hot or cold acceptance temperature limit.
4. Like note 3, but for qualification temperature limit.

Symbols: DT_A = Acceptance temperature range.
DT_Q = Qualification temperature range.

Table VI. Numbers of Cycles[1] for Thermal Cycle (TC) and Thermal Vacuum (TV) Tests.

Required Testing	Unit			Vehicle	
	Acceptance (Table XIII)		Qualification (Table X)	Acceptance (Table XII)	Qualification (Table VIII)
	N_A[3]	N_{AMAX}[4]	N_Q[5]	N_A	N_Q[5,6]
Both: TC[2]	8.5	17	53.5	4	10
TV	4	8	25	1	3
Only TV	1	2	6	4	13
Only TC	12.5	25	78.5		

Notes:
1. Numbers of cycles correspond to temperature ranges in Table V.
2. Tests may be conducted in vacuum to be integrated with TV.
3. For tailoring: $N_A = 10(125/DT_A)^{1.4}$ for TC only and for the sum of TC and TV when both conducted.
4. $N_{AMAX} = 2N_A$, but can be changed to allow for more or less retesting.
5. $N_Q = 4N_{AMAX}(DT_A/DT_Q)^{1.4}$, assuming temperature cycling during missionor other service is insignificant; if significant, additional cycling will be required using the same fatigue equivalence basis.
6. $N_{AMAX} = N_A$, assuming that vehicle-level acceptance retesting will not be conducted.

Symbols:

N_A = Required number of acceptance cycles.

N_{AMAX} = Maximum allowable number of acceptance cycles, including retesting.

N_Q = Required number of qualification cycles.

6.1.4.1 Accelerated Testing. All or any portion of the testing at the acceptance level may be accelerated by replacing it with a reduced duration of testing at the qualification level. Table VII shows time reduction factors, rounded to the nearest integer, for selected combinations of margin and maximum test tolerance on the spectrum at any frequency. For example, when the qualification margin M is 6 dB and the test tolerance on the spectrum T is as high as 3 dB at some frequency, the time reduction factor is 12. Then 24 minutes of acceptance level testing (6.1.4.2) could be accelerated to 2 minutes of testing at the qualification level. With a typical 1 minute test duration required for flight (4 times a typical 15 second fatigue equivalent duration in flight), the qualification test for this example would apply the extreme expected level for a total of 3 minutes per axis.

6.1.4.2 <u>Two-condition Testing</u>. The two-condition approach to acoustic or vibration qualification testing applies the acceptance test condition first (6.1.4a). For example, if the maximum allowable duration of acceptance vibration testing per axis is 6 minutes for any flight item, then 24 minutes of acceptance level vibration per axis would be required to satisfy the acceptance condition part of qualification. This would be followed by a test at the extreme expected spectrum, typically 6 dB higher for 1 minute per axis (6.1.4b)

Table VII. Time Reduction Factors, Acoustic and Random Vibration Tests.

Margin M (dB)	Maximum Test Tolerance on Spectrum, T (dB)	Time Reduction Factor
6.0	±1.5	15
6.0	±3.0	12
4.5	±1.5	7
4.5	±3.0	4
3.0	±1.5	3
3.0	±3.0	1
Note: In general, the time reduction factor is $10^{M/5}\,[1 + (4/3)\sinh^2(T/M)]^{-1}$, where T is the greater of the absolute value of the negative tolerance for the qualification test and the positive tolerance for the acceptance test.		

6.2 VEHICLE QUALIFICATION TESTS

The vehicle-level qualification test baseline will include all the required tests specified in Table VIII. The "other" tests (3.5.5) deemed applicable, and additional special tests that are conducted as acceptance tests for the vehicle element (such as alignments, instrument calibrations, antenna patterns, and mass properties), will also be conducted as part of qualification testing. Vehicle elements controlled by on-board data processing will have the flight version of the computer software resident in the on-board computer. Verification of the operational requirements will be demonstrated to the maximum extent practicable.

Table VIII. Vehicle Qualification Test Baseline.

Test	Reference Paragraph	Suggested Sequence	Launch Vehicle	Upper-stage Vehicle	Space Vehicle
Inspection[1]	4.5	1	R	R	R
Functional[1]	6.2.1	2	R	R	R
Pressure/leakage	6.2.6	3,7,11	R	R	R
EMC	6.2.2	4	R	R	R
Shock	6.2.3	5	R	R	R
Acoustic[2] or Vibration	6.2.4 or 6.2.5	6	O	R	R
Thermal Cycle[3]	6.2.7	8	O	O	O
Thermal Balance[4]	6.2.8	9	—	R	R
Thermal Vacuum	6.2.9	10	O	R	R
Modal Survey	6.2.10	any	R	R	R
Recommended vehicle qualification requirements (3.5.5) R = baseline requirement (high probability of being required) O = "other" (low probability of being required;					

Test	Reference Paragraph	Suggested Sequence	Launch Vehicle	Upper-stage Vehicle	Space Vehicle
— = not required (negligible probability of being required).					
Notes: 1. Required before and after each test as appropriate. Include special tests as applicable (6.2).					
2. Vibration conducted in place of acoustic test for a compact vehicle typically with mass less than 180 kg (400 lb).					
3. Required if thermal cycling acceptance test (7.2.7) conducted.					
4. May be combined with thermal vacuum test.					

6.2.1 Functional Test, Vehicle Qualification

6.2.1.1 Purpose. The functional test verifies that the mechanical and electrical performance of the vehicle meet the specification requirements, including compatibility with ground support equipment, and validates all test techniques and software algorithms used in computer-assisted commanding and data processing. Proper operation of all redundant units or mechanisms should be demonstrated to the maximum extent practicable.

6.2.1.2 Mechanical Functional Test. Mechanical devices, valves, deployables, and separation subsystems will be functionally tested at the vehicle level in the launch, orbital, or recovery configuration appropriate to the function. Alignment checks will be made where appropriate. Fit checks will be made of the vehicle physical interfaces using master gages or interface assemblies. The test should validate that the vehicle performs within maximum and minimum limits under worst-case conditions including environments, time, and other applicable requirements. Tests will demonstrate positive margins of strength, torque, and related kinematics and clearances. Where operation in earth gravity or in an operational temperature environment cannot be performed, a suitable ground test fixture may be used to permit operation and performance evaluation. The pass-fail criteria will be adjusted as appropriate to account for worst-case maximum and minimum limits that have been modified to adjust for ground test conditions.

6.2.1.3 Electrical and Fiber-optic Circuit Functional Test. The vehicle should be in its flight configuration with all units and subsystems connected, except explosive-ordnance elements. The test will verify the integrity of electrical and fiber-optic circuits, including functions, redundancies, end-to-end paths, and at least nominal performance, including radio-frequency and other sensor inputs. End-to-end sensor testing may be accomplished with a self-test or coupled inputs.

The test will be designed to operate all units, primary and redundant, and to exercise all commands and operational modes to the extent practicable. The operation of all thermally controlled units, such as heaters and thermostats, will be verified by test. Where control of such units is implemented by sensors, electrical or electronic devices, coded algorithms, or a computer, end-to-end performance testing should be conducted. The test will demonstrate that all commands having precondition requirements (such as enable, disable, a specific equipment configuration, and a specific command sequence), cannot be executed unless the preconditions are satisfied. Whenever practicable, equipment performance parameters that might affect end-to-end performance (such as power, voltage, gain, frequency, command and data rates) will be varied over specification ranges to demonstrate the performance. Autonomous functions will be verified to occur when the conditions exist for which they are designed. Continuous monitoring of several perceptive parameters, including input and output parameters, and the vehicle main bus by a power transient monitoring device, will be provided to detect intermittent failures.

For at least one functional test in the qualification sequence, the vehicle will be operated through a mission profile with all events occurring in actual flight sequence to the extent practicable. This sequence will include the final countdown, launch, ascent, separation, upper-stage operation, orbital operation, and return from orbit as appropriate. All explosive-ordnance firing circuits will be energized and monitored during these events to verify that the proper energy density is delivered to each device and in the proper sequence. All measurements that are telemetered will also be monitored during appropriate portions of these events to verify proper operations.

6.2.1.4 <u>Supplementary Requirements</u>. Functional tests will be conducted before and after each of the vehicle tests to detect equipment anomalies and to assure that performance meets specification requirements. These tests do not require the mission profile sequence. Sufficient data will be analyzed to verify the adequacy of the testing and the validity of the data before any change is made to an environmental test configuration, so that any required retesting can be readily accomplished. During these tests, the maximum use of telemetry will be employed for data acquisition, problem identification, and problem isolation. Functional tests required during individual vehicle tests are specified in connection with each test.

6.2.2 Electromagnetic Compatibility Test, Vehicle Qualification

6.2.2.1 <u>Purpose</u>. The electromagnetic compatibility test demonstrates electromagnetic compatibility of the vehicle and ensures that adequate margins exist in a simulated launch, orbital, disposal, and return-from-orbit electromagnetic environment.

6.2.2.2 <u>Test Description</u>. The operation of the vehicle and selection of instrumentation will be suitable for determining the margin against malfunctions and unacceptable or undesired responses due to electromagnetic incompatibilities.
The test will demonstrate satisfactory electrical and electronic equipment operation in conjunction with the expected electromagnetic radiation from other subsystems or equipment, such as from other vehicle elements and ground support equipment. The vehicle will be subjected to the required tests while in the launch, orbital, and return-from-orbit configurations and in all possible operational modes, as applicable. Special attention will be given to areas indicated to be marginal by analysis. Potential electromagnetic interference from the test vehicle to other subsystems will be measured. The tests will be conducted according to the requirements of MIL-STD-1541. The tests will include but not be limited to three main segments:

a. Radiated emissions susceptibility.

b. Intersystem radiated susceptibility.

c. External radio frequency interference susceptibility.

Explosive-ordnance devices having bridge wires, but otherwise inert, will be installed in the vehicle and monitored during all tests.

6.2.3 Shock Test, Vehicle Qualification

6.2.3.1 Purpose. The shock test demonstrates the capability of the vehicle to withstand or, if appropriate, to operate in the induced shock environments. The shock test also yields the data to validate the extreme and maximum expected unit shock requirement (3.3.7).

6.2.3.2 Test Description. The vehicle will be supported and configured to allow flight-like dynamic response of the vehicle with respect to amplitude, frequency content, and paths of transmission. Support of the vehicle may vary during the course of a series of shock tests in order to reflect the configuration at the time of each shock event. Test setups will avoid undue influence of test fixtures, and prevent recontact of separated portions.

In the shock test or series of shock tests, the vehicle will be subjected to shock transients that simulate the extreme expected shock environment (3.3.7) to the extent practicable. Shock events to be considered include separations and deployments initiated by explosive ordnance or other devices, as well as impacts and suddenly applied or released loads that may be significant for unit dynamic response (such as due to an engine transient, parachute deployment, and vehicle landing). All devices on the vehicle capable of imparting significant shock excitation to vehicle units will be activated. Those potentially significant shock sources not on the vehicle under test, such as on an adjoining payload fairing or a nearby staging joint, will also be actuated or simulated and applied through appropriate interfacing structures. Dynamic instrumentation will be installed to measure shock responses in 3 orthogonal directions at attachments of selected units.

6.2.3.3 Test Activations. All explosive-ordnance devices and other potentially significant shock-producing devices or events, including those from sources not installed on the vehicle under test, will be activated at least one time or simulated as appropriate. Significant shock sources are those that induce a shock response spectrum (3.3.7) at any unit location that is within 6 dB of the envelope of the shock response spectra from all shock sources. The significant sources will be activated 2 additional times to provide for variability in the vehicle test and to provide data for prediction of maximum and extreme expected shock environments for units (3.3.2). Activation of both primary and redundant devices will be carried out in the same sequence as they are intended to operate in service.

6.2.3.4 Supplementary Requirements. Electrical and electronic units will be operating and monitored to the maximum extent practicable. Continuous monitoring of several perceptive parameters, including input and output parameters, and the vehicle main bus by a power transient monitoring device, will be provided to detect intermittent failures.

6.2.4 Acoustic Test, Vehicle Qualification

6.2.4.1 Purpose. The acoustic test demonstrates the ability of the vehicle to endure acoustic acceptance testing and meet requirements during and after exposure to the extreme expected acoustic environment in flight (3.3.4). Except for items whose environment is dominated by structure-borne vibration, the acoustic test also verifies the adequacy of unit vibration qualification levels and serves as a qualification test for items not tested at a lower level of assembly.

6.2.4.2 Test Description. The vehicle in its ascent configuration will be installed in an acoustic test facility capable of generating sound fields or fluctuating surface pressures that induce vehicle vibration environments sufficient for vehicle qualification. The vehicle will be mounted on a flight-type support structure or reasonable simulation thereof. Significant fluid and pressure conditions will be replicated to the extent practicable. Appropriate dynamic instrumentation will be installed to measure vibration responses at attachment points of critical and representative units. Control microphones will be placed at a minimum of 4 well-separated locations, preferably at one half the distance from the test article to the nearest chamber wall, but no closer than 0.5 meter (20 inches) to both the test article surface and the chamber wall. When test article size exceeds facility capability, the vehicle may be appropriately subdivided and acoustically tested as one or more subsystems or assemblies.

6.2.4.3 Test Level and Duration. The test will be conducted per 6.1.4. The typical version of the test involves accelerated acceptance-level testing per 6.1.4.1 and applies the qualification-level spectrum for a total of 2 minutes. This is based on a qualification margin of 6 dB, a maximum of 3 minutes of accumulated acceptance testing on a flight vehicle, and a fatigue equivalent duration of not greater than 15 seconds. Operating time should be divided approximately equally between redundant functions. Where insufficient test time is available to test redundant units, functions, and modes that are operating during the launch, ascent, or reentry phase, extended testing will be performed at a level no lower than 6 dB below the qualification level.

6.2.4.4 Supplementary Requirements. During the test, all electrical and electronic units, even if not operating during launch, will be electrically energized and sequenced through operational modes to the maximum extent practicable, with the exception of units that may sustain damage if energized. Continuous monitoring of several perceptive parameters, including input and output parameters, and the vehicle main bus by a power transient monitoring device, will be provided to detect intermittent failures.

6.2.5 Vibration Test, Vehicle Qualification. The vibration test may be conducted instead of an acoustic test (6.2.4) for small, compact vehicles which can be excited more effectively via interface vibration than by an acoustic field. Such vehicles typically have a mass under 180 kilograms (400 pounds).

6.2.5.1 Purpose. The vibration test demonstrates the ability of the vehicle to endure vibration acceptance testing and meet requirements during and after exposure to the extreme expected environment in flight (3.3.5). Except for items whose response is dominated by acoustic excitation, the vibration test also verifies the adequacy of unit vibration qualification levels and serves as a qualification test for items that have not been tested at a lower level of assembly.

6.2.5.2 Test Description. The vehicle and a flight-type adapter, in the ascent configuration, will be vibrated using one or more shakers through appropriate vibration fixtures. Vibration will be applied in each of 3 orthogonal axes, one direction being parallel to the vehicle thrust axis. Instrumentation will be installed to measure, in those same 3 axes, the vibration inputs and the vibration responses at attachment points of critical and representative units.

6.2.5.3 Test Levels and Duration. The test will be conducted per 6.1.4 to produce the required spectrum at the input to the vehicle or at attachment points of critical or representative units, as specified. When necessary to prevent unrealistic input forces or unit responses, the spectrum at the vehicle input may be limited or notched, but not below the minimum spectrum for a vehicle (7.1.3). The typical version of the test for each axis involves accelerated acceptance-level testing per 6.1.4.1 and applies the qualification spectrum for 2 minutes (same basis as in 6.2.4.3). Operating time should be divided approximately equally between redundant functions. Where insufficient test time is available to test redundant units, functions, and modes that are operating during the launch, ascent, or reentry phase, extended testing will be performed at a level no lower than 6 dB below the qualification level.

6.2.5.4 Supplementary Requirements. Same as 6.2.4.4, except that the structural response will also be monitored to ensure that no unrealistic test conditions occur.

6.2.6 Pressure and Leakage Tests, Vehicle Qualification

6.2.6.1 Purpose. These tests demonstrate the capability of pressurized subsystems to meet the specified flow, pressure, and leakage rate requirements.

6.2.6.2 Test Description. The vehicle will 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. Preliminary tests will be performed, as necessary, to verify compatibility with the test setup and to ensure proper control of the equipment and test functions. The requirements of the subsystem including flow, leakage, and regulation will be measured while operating applicable valves, pumps, and motors. The flow checks will verify that the plumbing configurations are adequate. Checks for subsystem cleanliness, moisture levels, and pH levels will also be made. Where pressurized subsystems are assembled with other than brazed or welded connections, the specified torque values for these connections will be verified prior to the initial qualification leak check.

In addition to the high-pressure test, propellant tanks and thruster valves will be tested for leakage under propellant servicing conditions. The subsystem will be evacuated to the internal pressure normally used for propellant loading and the pressure monitored for decay as an indication of leakage.

6.2.6.3 Test Levels and Durations.

a. For launch and upper-stage vehicles which contain pressurized structures, the pressurized subsystem will be pressurized to a proof pressure which is 1.1 times the maximum expected operating pressure (MEOP) and held constant for a short dwell time, sufficient to assure that the proper pressure was achieved within the allowed test tolerance. The test pressure will then be reduced to the MEOP for leakage inspection.

b. For space vehicles, unless specified otherwise, the pressurized subsystems will be pressurized to a proof pressure which is 1.25 times the MEOP and held for 5 minutes and then the pressure will be reduced to the MEOP. This sequence will be conducted 3 times, followed by inspection for leakage at the MEOP. The duration of the evacuated propulsion subsystem leakage test will not exceed the time that this condition is normally experienced during propellant loading.

6.2.6.4 Supplementary Requirements. Applicable safety standards will be followed in conducting all tests. Tests for detecting external leakage will be performed at such locations as joints, fittings, plugs, and lines. The acceptable leakage rate to meet mission requirements will be based upon an appropriate analysis. In addition, the measurement technique will account for leakage rate variations with pressure and temperature and have the required threshold, resolution, and accuracy to detect any leakage equal to or greater than the acceptable leak rate. If appropriate, the leakage rate measurement will be performed at the MEOP and at operational temperature, with the representative fluid commodity, to account for dimensional and viscosity changes. Times to achieve thermal and pressure equilibrium, test duration, and temperature sensitivity will be determined by an appropriate combination of analysis and development test, and the results documented. Leakage detection and measurement procedures may require vacuum chambers, bagging of the entire vehicle or localized areas, or other special techniques to achieve the required accuracies.

6.2.7 Thermal Cycle Test, Vehicle Qualification

6.2.7.1 Purpose. The thermal cycle test demonstrates the ability of the vehicle to withstand the stressing associated with flight vehicle thermal cycle acceptance testing, with a qualification margin on temperature range and maximum number of cycles. The thermal cycle test, in combination with a reduced-cycle thermal vacuum test, can be selected as an alternate to the thermal vacuum test (6.2.9 and Table VI).

6.2.7.2 Test Description. The vehicle will be placed in a thermal chamber at ambient pressure, and a functional test will be performed to assure readiness for the test. The vehicle will be operated and monitored during the entire test, except that vehicle power may be turned off if necessary to reach stabilization at the cold temperature. Vehicle operation will be asynchronous with the temperature cycling, and redundant units will be operated for approximately equal times.

When the relative humidity of the inside spaces of the vehicle is below the value at which the cold test temperature would cause condensation, the temperature cycling will begin. One complete thermal cycle is a period beginning at ambient temperature, then cycling to one temperature extreme and stabilizing (3.5.8), then to the other temperature extreme and stabilizing, and then returning to ambient temperature. Strategically placed temperature monitors installed on units will assure attainment and stabilization of the expected temperature extremes for several units. Auxiliary heating and cooling may be employed for selected temperature-sensitive units (e.g., batteries). If it is necessary in order to achieve the required temperature rate of change, parts of the vehicle such as solar arrays and passive thermal equipment may be removed for the test. The last thermal cycle will contain cold and hot soaks during which the vehicle will undergo a functional test, including testing of redundant units.

6.2.7.3 Test Level and Duration. The minimum vehicle temperature range will be 70°C from the hot to the cold condition (Table V). With the 70°C qualification temperature range, the required number of cycles will be 10. For other ranges, see Table VI. The average rate of change of temperature will be as rapid as practicable.

6.2.7.4 Supplementary Requirements. Continuous monitoring of several perceptive parameters, including input and output parameters and the vehicle main bus by a power transient monitoring device, will be provided to detect intermittent failures. Moisture condensation inside of electrical and electronic units will be prevented. Combinations of temperature and humidity which allow moisture deposition either on the exterior surfaces of the vehicle or inside spaces where the humidity is slow to diffuse (for example, multilayer insulation) will be avoided.

6.2.8 Thermal Balance Test, Vehicle Qualification

6.2.8.1 Purpose. The thermal balance test provides the data necessary to verify the analytical thermal model and demonstrates the ability of the vehicle thermal control subsystem to maintain the specified operational temperature limits of the units and throughout the entire vehicle. The thermal balance test also verifies the adequacy of unit thermal design criteria. The thermal balance test can be combined with the thermal vacuum test (6.2.9).

6.2.8.2 Test Description. The qualification vehicle will be tested to simulate the thermal environment experienced by the vehicle during its mission. Tests will be capable of validating the thermal model over the full mission range of seasons, equipment duty cycles, ascent conditions, solar angles, maximum and minimum unit thermal dissipations including effects of bus voltage variations, and eclipse combinations so as to include the worst-case hot and cold temperatures for all vehicle units. As a minimum, two test conditions will be imposed: a worst hot case and a worst cold case. If practicable, 2 additional cases should be imposed: a transient for correlation with the model and a case chosen to check the validity of the correlated model. Special emphasis will be placed on defining the test conditions expected to produce the maximum and minimum temperatures of sensitive units such as batteries. Sufficient measurements will be made on the vehicle internal and external units to verify the vehicle thermal design and analyses. The power requirements of all thermostatically or electronically controlled heaters and coolers will be verified during the test, and appropriate control authority demonstrated.

The test chamber, with the test item installed, will provide a pressure of no higher than 13.3 millipascal (10^{-4} Torr) for space and upper-stage vehicles, or a pressure commensurate with service altitude for launch vehicles. Where appropriate, provisions should be made to prevent the test item from "viewing" warm chamber walls, by using black-coated cryogenic shrouds of sufficient area and shape that are capable of approximating liquid nitrogen temperatures. The vehicle thermal environment may be supplied by one of the following methods:

a. Absorbed Flux. The absorbed solar, albedo, and planetary irradiation is simulated using heater panels or infrared (IR) lamps with their spectrum adjusted for the external thermal coating properties, or using electrical resistance heaters attached to vehicle surfaces.

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

c. Equivalent Radiation Sink Temperature. The equivalent radiation sink temperature is simulated using infrared lamps and calorimeters with optical properties identical to those of the vehicle surface.

d. Combination. The thermal environment is supplied by a combination of the above methods.

The selection of the method and fidelity of the simulation depends upon details of the 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 will be incorporated down to the unit level to evaluate total vehicle performance within operational limits as well as to identify unit problems. The vehicle will be operated and monitored throughout the test. Dynamic flight simulation of the vehicle thermal environment should be provided unless the external vehicle temperature does not vary significantly with time. (See 4.10 regarding formation of a Test Evaluation Team.)

6.2.8.3 Test Levels and Duration. Test conditions and durations for the thermal balance test are dependent upon the vehicle configuration, design, and mission details. Normally, boundary conditions for evaluating thermal design will include both of the following:

a. Maximum external absorbed flux plus maximum internal power dissipation.

b. Minimum external absorbed flux plus minimum internal power dissipation.

The thermal time constant of the subsystems and mission profile both influence the time required for the vehicle to achieve thermal equilibrium and hence the test duration.

6.2.8.4 Supplementary Requirements. Success criteria depend not only on survival and operation of each item within specified temperature limits, but also on correlation of the test data with theoretical thermal models. As a goal, correlation of test results to the thermal model predictions should be within ± 3°C. Lack of correlation with the theoretical models may indicate either a deficiency in the model, test setup, or vehicle hardware. The correlated thermal math model will be used to make the final temperature predictions for the various mission phases (such as prelaunch, ascent, on-orbit, and disposal orbit).

6.2.9 Thermal Vacuum Test, Vehicle Qualification.

6.2.9.1 Purpose. The thermal vacuum test demonstrates the ability of the vehicle to meet qualification requirements under vacuum conditions and temperature extremes which simulate those predicted for flight plus a design margin, and to withstand the thermal stressing environment of the vehicle thermal vacuum acceptance test plus a qualification margin on temperature range and number of cycles.

6.2.9.2 Test Description. The vehicle will be placed in a thermal vacuum chamber and a functional test performed to assure readiness for chamber closure. The vehicle will be divided into separate equipment zones, based on the limits of the temperature-sensitive units and similar unit qualification temperatures within each zone. Units that operate during ascent will be operating and monitored for corona and multipacting, as applicable, as the pressure is reduced to the lowest specified level. The rate of chamber pressure reduction will be no greater than during ascent, and may have to be slower to allow sufficient time to monitor for corona and multipacting. Equipment that does not operate during launch will have electrical power applied after the lowest specified pressure level has been reached. A thermal cycle begins with the vehicle at ambient temperature. The temperature is raised to the specified high level and stabilized (3.5.8). Following the high-temperature soak, the temperature will be reduced to the lowest specified level and stabilized. Following the low-temperature soak, the vehicle will be returned to ambient temperature to complete one thermal cycle. Functional tests will be conducted during the first and last thermal cycle at both the high- and low-temperature limits with functional operation and monitoring of perceptive parameters during all other cycles. If simulation of the ascent environment is desirable at the beginning of the test, the first cycle may begin with a transition to cold thermal environment, rather than a hot thermal environment.

In addition to the thermal cycles for an upper-stage or space vehicle, the chamber may be programmed to simulate various orbital flight operations. Execution of operational sequences will be coordinated with expected environmental conditions, and a complete cycling of all equipment will be performed including the operating and monitoring of redundant units and paths. Vehicle electrical equipment will be operating and monitored throughout the test. Temperature monitors will assure attainment of temperature limits. Strategically placed witness plates, quartz crystal microbalances, or other instrumentation will be installed in the test chamber to measure the outgassing from the vehicle and test equipment.

6.2.9.3 Test Levels and Duration. Temperatures in various equipment areas will be controlled by the external test environment and internal heating resulting from equipment operation. During the hot and cold half cycles, the temperature limit is reached as soon as one unit in each equipment area is at the hot or cold temperature reached during its qualification thermal testing. Unit temperatures will not be allowed to go outside their qualification range at any time during the test. The pressure will be maintained at no higher than 13.3 millipascal (10^{-4} Torr) for space and upper-stage vehicles and, for launch vehicles, at no higher than the pressure commensurate with the highest possible service altitude. When the alternate thermal cycle test (6.2.7) is not performed, the thermal vacuum qualification test will include at least 13 complete hot-cold cycles (Table VI). When thermal cycling is performed, the thermal vacuum qualification test will include at least 3 complete hot-cold cycles (Table VI).

The rate of temperature change will equal or exceed the maximum predicted mission rate of change. The temperature soak (3.5.11) will be at least 8 hours at each temperature extreme during the first and last cycles. For intermediate cycles, the soak duration will be at least 4 hours. Operating time should be divided approximately equally between redundant units.

6.2.9.4 Supplementary Requirements. Continuous monitoring of several perceptive parameters, including input and output parameters, and the vehicle main bus by a power transient monitoring device, will be provided to detect intermittent failures. It may be necessary to achieve temperature limits 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 vehicle will generally be avoided. External baffling, shadowing, or heating will be utilized to the extent feasible. The vehicle will be operated over the qualification temperature range, although performance within specification is not required outside of 10^{m}C beyond the maximum and minimum expected temperatures.

6.2.10 Mode Survey Test, Vehicle Qualification.

6.2.10.1 Purpose. The mode survey test (or modal survey) is conducted to experimentally derive a structural dynamic model of a vehicle or to provide a basis for test-verification of an analytical model. After upgrading analytically to the flight configuration (such as different propellant loading and minor differences between flight and test unit mass properties), this model is used in analytical simulations of flight loading events to define the verification-cycle structural loads environment. These loads are used to determine structural margins and adequacy of the structural static test loading conditions (6.3.1). They are therefore critical for verification of vehicle structural integrity and qualification of the structural subsystem as flight-ready. Where practicable, a modal survey is also performed to define or verify models used in the final preflight evaluation of structural dynamic effects on control subsystem precision and stability.

6.2.10.2 Test Description. The test article will consist of flight-quality structure with assembled units, payloads, and other major subsystems, and will contain actual or simulated liquids at specified fill-levels. For large vehicles, complexity and testing practicability may dictate that tests be performed on separate sections of the vehicle. For large launch vehicles in particular, practicality may also dictate use of an integrated program of ground and flight tests, involving substantial flight data acquisition and analysis, to acquire the necessary data for model verification. Wire harnesses may be installed for the mode survey test, but are not required. Mass simulators may be used to represent a flight item when its attachment-fixed resonances have been demonstrated by test to occur above the frequency range of interest established for the modal survey. Dynamic simulators may be used for items that have resonances within the frequency range of interest if they are accurate dynamic representations of the flight item. Alternatively, mass simulators may be used if flight-quality items are subjected separately to a modal survey meeting qualification requirements. All mass simulators are to include realistic simulation of interface attach structure and artificial stiffening of the test structure will be avoided.

The data obtained in the modal survey will be adequate to define the resonant frequencies and associated mode shapes and damping values, for all modes that occur in the frequency range of interest, generally up to at least 50 Hz. In addition, the primary mode will be acquired in each coordinate direction, even if its frequency lies outside the specified test range. The test modes are considered to have acceptable quality when they are orthogonal, with respect to the analytical mass matrix, to within 10%. (See 4.10 regarding formation of a Test Evaluation Team.)

6.2.10.3 Test Levels. The test is generally conducted at response levels that are low compared to the expected flight levels. Limited testing will be conducted to evaluate nonlinear behavior, with a minimum of 3 levels used when significant nonlinearity is identified.

6.2.10.4 <u>Supplementary Requirements</u>.

6.2.10.4.1 <u>Correlation Requirements.</u> When the modal survey data are used to test-verify an analytical dynamic model for the verification-cycle loads analyses, rather than to define the model directly, adequate model-to-test correlation will be demonstrated quantitatively as follows:

a. Using a cross-orthogonality matrix formed from the analytical mass matrix and the analytical and test modes, corresponding modes are to exhibit at least 95% correlation and dissimilar modes are to be orthogonal to within 10%.

b. Analytical model frequencies are to be within 3% of test frequencies.

With adequate justification, limited exceptions to this standard of correlation are acceptable for problem modes; also, alternative quantitative techniques can be used if their criteria for acceptability are comparable.

6.2.10.4.2 <u>Pretest Requirements.</u> Because of their criticality to achieving a successful test, appropriate pretest analyses and experimentation will be performed to:

a. Establish adequacy of the test instrumentation.

b. Evaluate the test stand and fixturing to preclude any boundary condition uncertainties that could compromise test objectives.

c. Verify that mass simulators have no resonances within the frequency range of interest.

6.3 <u>SUBSYSTEM QUALIFICATION TESTS</u>

Subsystem qualification tests will be conducted on subsystems for any of the following purposes:

a. To verify their design.

b. To qualify those subsystems that are subjected to environmental acceptance tests.

c. When this level of testing provides a more realistic or more practical test simulation than testing at another level of assembly.

For purpose c, included are tests such as the required structural static load test, and environmental tests where the entire flight item is too large for existing facilities. Also, the qualification of certain units such as interconnect tubing or wiring may be more readily completed at the subsystem level rather than at the unit level. In this case, the appropriate unit tests may be conducted at the subsystem level to complete required unit qualification tests. Types of subsystems that are not specifically identified herein may be tested in accordance with the vehicle level test requirements. Subsystem qualification test requirements are listed in Table IX.

6.3.1 <u>Structural Static Load Test, Subsystem Qualification</u>.

6.3.1.1 <u>Purpose</u>. The structural static load test demonstrates the adequacy of the subsystem structures to meet requirements of strength and stiffness, with the desired qualification margin, when subjected to simulated critical environments (such as temperature, humidity, pressure, and loads) predicted to occur during its service life (3.5.7).

6.3.1.2 <u>Test Description</u>. The support and load application fixture will consist of an adequate replication of the adjacent structural section to provide boundary 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 will be included in the test requirements. If more than one design ultimate load condition is to be applied to the same test specimen, a method of sequential load application will 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, and to provide data for any conditions which simulate those existing in the flight article. Static loads representing the design yield load (3.4.5) and the design ultimate load (3.4.4) will be applied to the structure, and measurements of the strain and deformation will be recorded. Strain and deformation will be measured before loading, after removal of the yield loads, and at several intermediate levels up to yield load for post-test diagnostic purposes. The test conditions will encompass the extreme predicted combined effects of acceleration, vibration, pressure, preloads, and temperature. These effects can be simulated in the test conditions as long as the failure modes are covered and the design margins are enveloped by the test. For example, temperature effects, such as material strength degradation and additive thermal stresses, can often be accounted for by increasing mechanical loads. Analysis of flight profiles will be used in subsequent design modification effort, and to provide data for use in any weight reduction programs. Failure at design yield load means material gross yielding or deflections which degrade mission performance. Failure at design ultimate load means rupture or collapse. (See 4.10 regarding formation of a Test Evaluation Team.)

6.3.1.3 <u>Test Levels and Duration</u>

a. <u>Static Loads</u>. Unless otherwise specified, the design ultimate load test will be conducted at 1.4 times the limit load for manned flight, and 1.25 times the limit load for unmanned flight. The design yield load test will be conducted at 1.0 times limit load for both manned and unmanned flight.

b. Temperature. Critical flight temperature and load combinations will be simulated or taken into account.

c. Duration of Loading. Loads will be applied as closely as practicable to actual flight loading times, with a dwell time not longer than necessary to record test data such as stress, strain, deformation, and temperature.

Table IX. Subsystem Qualification Test Baseline.

TEST	Reference Paragraph	Structure	Space Experiment	Launch Vehicle Subsystem	Payload Fairing
Static Load	6.3.1	R	O[4]	O[4]	R
Vibration or Acoustic	6.3.2 6.3.3	O[1]	O[1]	O[1,2]	R[5]
Thermal Vacuum	6.3.4	O	R[3]	O[2]	O
Separation	6.3.5	R	—	—	R
Mechanical Functional	6.2.1.2	O	O	O[4]	R

Recommended vehicle qualification requirements (3.5.5).
R = baseline requirement (high probability of being required)
O = "other" (low probability of being required.)
— = not required (negligible probability of being required).

Notes:
1 Vibration conducted in place of acoustic test for a compact subsystem.
2 Required for subsystems containing critical equipment (for example, guidance equipment). Not required if performed at the vehicle level.
3 Discretionary if performed at the vehicle level.
4 Required if not performed at another level of assembly.
5 Acoustic test required.

6.3.1.4 Supplementary Requirements. Pretest analysis will be conducted to identify the locations of minimum design margins and associated failure modes that correspond to the selected critical test load conditions. This analysis will be used to locate instrumentation, to determine the sequence of loading conditions, and to provide early indications of anomalous occurrences during the test. This analysis will also form the basis for judging the adequacy of the test loads. In cases where a load or other environment has a relieving, stabilizing, or other beneficial effect on the structural capability, the minimum, rather than the maximum, expected value will be used in defining limit-level test conditions. In very complex structures where simulation of the actual flight loads is extremely difficult, or not feasible, multiple load cases may be used to exercise all structural zones to design yield and design ultimate loads.

6.3.2 Vibration Test, Subsystem Qualification

6.3.2.1 Purpose. Same as 6.2.5.1.

6.3.2.2 Test Description. Same as 6.2.5.2.

6.3.2.3 Test Levels and Duration. Same as 6.2.5.3.

6.3.2.4 Supplementary Requirements. Same as 6.2.5.4.

6.3.3 Acoustic Test, Subsystem Qualification

6.3.3.1 Purpose. Same as 6.2.4.1.

6.3.3.2 Test Description. Same as 6.2.4.2.

6.3.3.3 Test Levels and Duration. Same as 6.2.4.3.

6.3.3.4 Supplementary Requirements. Same as 6.2.4.4, as applicable.

6.3.4 Thermal Vacuum Test, Subsystem Qualification

6.3.4.1 Purpose. Same as 6.2.9.1.

6.3.4.2 Test Description. Same as 6.2.9.2.

6.3.4.3 Test Levels and Duration. Same as 6.2.9.3.

6.3.4.4 Supplementary Requirements. Same as 6.2.9.4.

6.3.5 Separation Test, Subsystem Qualification

6.3.5.1 Purpose. The separation test demonstrates the adequacy of the separation subsystem to meet its performance requirements on such parameters as: separation velocity, acceleration, and angular motion; time to clear and clearances between separating hardware; flexible-body distortion and loads; amount of debris; and explosive-ordnance shock levels. For a payload fairing using a high-energy separation subsystem, the test also demonstrates the structural integrity of the fairing and its generic attachments under the separation shock loads environment. The data from the separation test are also used to validate the analytical method and basic assumptions used in the separation analysis. The validated method is then used to verify that requirements are met under worst-case flight conditions.

6.3.5.2 Test Description. The test fixtures will replicate the interfacing structural sections to simulate the separation subsystem boundary conditions existing in the flight article. The remaining boundary conditions for the separating bodies will simulate the conditions in flight at separation, unless the use of other boundary conditions will permit an unambiguous demonstration that subsystem requirements can be met. The test article will include all attached flight hardware that could pose a debris threat if detached. When ambient atmospheric pressure may adversely affect the test results, such as for large fairings, the test will be conducted in a vacuum chamber duplicating the altitude condition encountered in flight at the time of separation. Critical conditions of temperature, pressure, or loading due to acceleration will be simulated or taken into account. As a minimum, instrumentation will include high-speed cameras to record the motion of specially marked target locations, accelerometers to measure the structural response, and strain gages to verify load levels in structurally critical attachments. (See 4.10 regarding formation of a Test Evaluation Team.)

6.3.5.3 Test Activations. A separation test will be conducted to demonstrate that requirements on separation performance parameters are met under nominal conditions. When critical off-nominal conditions cannot be modeled with confidence, at least one additional separation test will be conducted to determine the effect on the separation process. When force or torque margin requirements are appropriate, a separate test will be conducted to demonstrate that the margin is at least 100%; for separation subsystems involving fracture of structural elements, however, the margin demonstrated will be at least 50%. In addition, debris risk will be evaluated by conducting a test encompassing the most severe conditions that can occur in flight, or by including loads scaled from those measured in tests under nominal conditions.

6.3.5.4 Supplementary Requirements. A post-test inspection for debris will be conducted on the test article and in the test chamber.

6.4 UNIT QUALIFICATION TESTS

The unit qualification test baseline will include all the required tests specified in Table X. The "other" tests (3.5.5) deemed applicable, and additional special tests that are conducted as acceptance tests on the unit, will also be conducted as part of qualification testing. Unit qualification tests will normally be accomplished entirely at the unit level. However, in certain circumstances, the required unit qualification tests may be conducted partially or entirely at the subsystem or vehicle levels of assembly. Tests of units 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. If moving mechanical assemblies or other units have static or dynamic fluid interfaces or are pressurized during operation, those conditions should be replicated during unit qualification testing. Unit performance will meet the applicable mission requirements over the entire qualification environmental test range, to the maximum extent practicable. At the end of all required qualification tests, the qualification unit should be disassembled and inspected (4.5).

Where units fall into two or more categories of Table X, the required tests specified for each category will be applied. For example, a star sensor may be considered to fit both "Electrical and Electronic" and "Optical" categories. A thruster with integrated valves would be considered to fit both "Thruster" and "Valve" categories.

6.4.1 Functional Test, Unit Qualification

6.4.1.1 Purpose. The functional test verifies that the electrical, optical, and mechanical performance of the unit meets the specified operational requirements of the unit.

6.4.1.2 Test Description. Electrical tests will include application of expected voltages, impedance, frequencies, pulses, and waveforms at the electrical interfaces of the unit, including all redundant circuits. These parameters will be varied throughout their specification ranges and the sequences expected in flight operation. The unit output will be measured to verify that the unit performs to specification requirements. Functional performance will also include electrical continuity, stability, response time, alignment, pressure, leakage, or other special tests that relate to a particular unit configuration. Moving mechanical assemblies will be tested in the configuration corresponding to the environment being simulated and will be passive or operating corresponding to their state during the corresponding environmental exposure. Torque versus angle and time versus angle, or equivalent linear measurements for linear devices, will be made. Functional tests should include stiffness, damping, friction and breakaway characteristics, where appropriate. Moving mechanical assemblies that contain redundancy in their design will demonstrate required performance in each redundant mode of operation during the test.

Table X. Unit Qualification Test Baseline.

Test	Reference Paragraph	Suggested Sequence	Electrical and Electronic	Antenna	MMA	Solar Array	Battery	Valve or Propulsion Component	Pressure Vessel or Component	Thruster	Thermal	Optical	Structural Component
Inspection[1]	4.5	1	R	R	R	R	R	R	R	R	R	R	R
Functional[1]	6.4.1	2	R	R	R	R	R	R	R	R	R	R	—
Leakage[2]	6.4.7	3,6,12	R	—	R	—	R	R	R	O	O	—	—
Shock	6.4.6	4	R	O[4]	O[4]	O[4]	O[4]	O[4]	O	O[4]	O[4]	O[4]	O
Vibration	6.4.4	5	R	R[5]	R	R[5]	R	R	R	R	R	R[5]	O[7]
Acoustic	6.4.5	5	O	R[5]	—	R[5]	—	—	—	—	—	R[5]	—
Acceleration	6.4.9	7	O	R	O	O	O	—	O	—	—	R	—
Thermal Cycle	6.4.2	8	R	—	—	—	—	—	—	—	—	—	—
Thermal Vac	6.4.3	9	R	R	R	R	R	R	O	R	R	R	O
Climatic	6.4.12	10	O	O	O	O	O	O	O	O	O	O	—
Proof Pressure[3]	6.4.8	11	O	—	O	—	O	R	R	R	O	—	—
EMC	6.4.11	13	R	O	O	—	O	—	—	—	—	—	—
Life	6.4.10	14	O	O	O	O	R	O	R[6]	R	O	O	O[8]
Burst [3]	6.4.8	15	O	—	—	—	O	O	R	O	O	—	—

Recommended unit qualification requirements (3.5.5).

R = baseline requirement (high probability of being required).
O = "other" (low probability of being required.
— = not required (negligible probability of being required).

Notes:

1 Required before and following each test as appropriate. Include special tests as applicable (6.2).
2 Required when component is sealed or pressurized.
3 Required when component is pressurized.
4 Required when maximum expected shock spectrum in g's exceeds 0.8 times the frequency in Hz.
5 Either vibration or acoustic test required, whichever is more appropriate, with the other discretionary.
6 For pressure vessels, test per MIL-STD-1522. For pressure components, other than bellows and other flexible fluid devices or lines, life tests are discretionary.
7 Test required if the structural component has a low margin for fatigue, or is not subjected to a static strength qualification test (6.4.4.6).
8 For pressurized structures, the pressure cycle test (6.4.8.2b and 6.4.8.3c) will be required.

Table X. (Continued)
Unit Qualification Test Baselines

6.4.1.3 Supplementary Requirements. Functional or monitoring tests will be conducted before, during, and after each of the unit tests to detect equipment anomalies and to assure that performance meets specification requirements.

6.4.2 Thermal Cycle Test, Electrical and Electronic Unit Qualification

6.4.2.1 Purpose. The thermal cycle test demonstrates the ability of electrical and electronic units to operate over the qualification temperature range and to endure the thermal cycle testing imposed during acceptance testing.

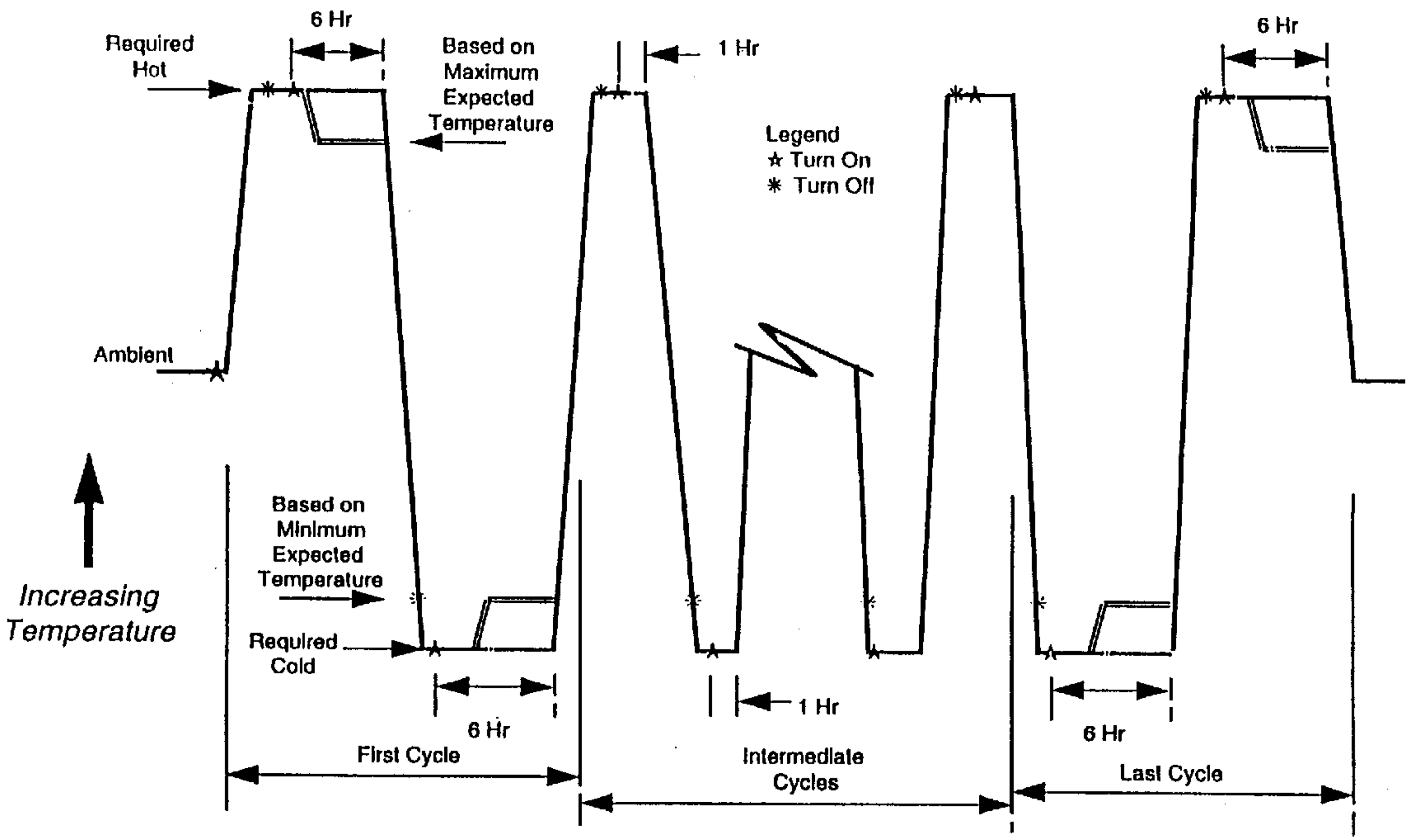
6.4.2.2 Test Description. With the unit operating (power on) and while perceptive parameters are being monitored, the test will follow the temperature profile in Figure 1. The test control temperature will be measured at a representative location on the unit, such as at the mounting point on the baseplate. Each time the control temperature has stabilized (3.5.8) at the hot temperature, the unit will be turned off and then hot started. Then, with the unit operating, the control temperature will be reduced to the cold temperature and the unit turned off. To aid in reaching the cold temperature, the unit may be powered off when the temperature of the unit is at least 10^{m}C colder than its minimum expected temperature (3.3.1). After the unit has stabilized at the cold temperature, the unit will be cold started. Temperature change from ambient to hot, to cold, and return to ambient constitutes one thermal cycle.

6.4.2.3 Test Levels and Duration

a. Pressure and Humidity. Ambient pressure is normally used; however, the thermal cycle test may be conducted at reduced pressure, including vacuum conditions. When unsealed units are being tested, provisions will be taken to preclude condensation on and within the unit at low temperature. For example, the chamber may be flooded with dry air or nitrogen. Also, the last half cycle will be hot

b. Temperature. The unit temperature will reach the qualification hot temperature, 10°C above the acceptance hot temperature (7.1.1), during the hot half cycle; the qualification cold temperature, 10°C below the acceptance cold temperature, during the cold half cycle (Table V). For units exposed to cryogenic temperatures in service, qualification margins will be prescribed on an individual basis. The transitions between hot and cold should be at an average rate of 3 to 5°C per minute, and will not be slower than 1°C per minute.

c. Duration. Table VI shows the number of qualification thermal cycles required for various situations. The last 4 thermal cycles will be failure free. Thermal soak durations (3.5.11) will be a minimum of 6 hours at the hot and 6 hours at the cold temperature during the first and last cycle (Figure 1). Intermediate cycles will have at least 1-hour soaks at the hot and cold temperatures. During thermal soaks, the unit will be turned off until the temperature stabilizes (3.5.8) and then turned on, remaining on until the next soak period off-on sequence. Measurement of thermal soak durations will begin at the time of unit turn-on (Figure 1).



Notes: 1. Operation is continuous except the component is turned off prior to cold and hot start.
2. To facilitate cool down, the component may also be turned off during cool down at temperatures below the minimum expected temperature (3.3.1).

Figure 1. Typical Component Thermal Cycle Profile

6.4.2.4 Supplementary Requirements. The requirements of the thermal cycle test may be satisfied by extending the thermal vacuum test of 6.4.3, to achieve the number of cycles required to meet the requirements of Table VI. Selection of such an alternative requires that the applicable acceptance test be carried out in the same fashion. Functional tests will be conducted after the unit temperatures have stabilized at the hot and cold temperatures during the first and last thermal cycle, and after return to ambient. During the remainder of the test, electrical and electronic units, including all redundant circuits and paths, will be cycled through various operational modes. Perceptive parameters will be monitored for failures and intermittents to the maximum extent practicable. Units will meet their performance requirements within specification over the maximum expected temperature range (3.3.1) extended at both temperature extremes by margins indicated in Figure 3. For digital units, such as computers, the final thermal cycle should employ a sufficiently slow temperature transition to permit a complete functional check to be repeated at essentially all temperatures.

Moisture condensation inside of electrical or electronic units will be prevented. Condensation is also minimized by requiring the first and last half cycle to be hot (Figure 1).

6.4.3 Thermal Vacuum Test, Unit Qualification

6.4.3.1 Purpose. The thermal vacuum test demonstrates the ability of the unit to perform in the qualification thermal vacuum environment and to endure the thermal vacuum testing imposed on flight units during acceptance testing. It also serves to verify the unit thermal design.

6.4.3.2 Test Description. The unit will be mounted in a vacuum chamber on a thermally controlled heat sink or in a manner similar to its actual installation in the vehicle. The unit surface finishes, which affect radiative heat transfer or contact conductance, will be thermally equivalent to those on the flight units. For units designed to reject their waste heat through the baseplate, a control temperature sensor will be attached either to the unit baseplate or the heat sink. The location will be chosen to correspond as closely as possible to the temperature limits used in the vehicle thermal design analysis or applicable unit-to-vehicle interface criteria. For components cooled primarily by radiation, a representative location on the unit case will similarly be chosen. The unit heat transfer to the thermally controlled heat sink and the radiation heat transfer to the environment will be controlled to the same proportions as calculated for the flight environment. During testing of radio-frequency (rf) equipment susceptible to multipaction, a space nuclear radiation environment will be simulated by a gamma-ray or x-ray source at 4 rads per hour.

The chamber pressure will be reduced to the required vacuum conditions. Units that are required to operate during ascent will be operating and monitored for arcing and corona during the reduction of pressure to the specified lowest levels and during the early phase of vacuum operation. At vacuum pressures below 133 millipascals (10^{-3} Torr), units will be monitored as appropriate to also assure that multipacting does not occur. Units that do not operate during launch will have electrical power applied after the test pressure level has been reached.

A thermal cycle begins with the conductive or radiant sources and sinks at ambient temperature. With the unit operating and while perceptive parameters are being monitored, the unit temperature is raised to the specified hot temperature and maintained. All electrical and electronic units that operate in orbit will be turned off, then hot started after a duration sufficient to ensure the unit internal temperature has stabilized (3.5.8), and then functionally tested. With the unit operating, the component temperature will be reduced to the specified cold temperature. To aid in reaching the cold temperature, the unit may be powered off when the temperature of the unit is at least 10°C colder than its minimum expected temperature (3.3.1). After the unit temperature has reached the specified cold temperature, the unit will be turned off (if not previously turned off during the transition) until the internal temperature stabilizes (3.5.8) and then cold started and functionally tested, continuing to maintain the unit at the specified temperature until the end of the soak. The temperature of the sinks will then be raised to ambient conditions. This constitutes one complete thermal cycle.

6.4.3.3 Test Levels and Duration

a. Pressure. For units required to operate during ascent, the time for reduction of chamber pressure from ambient to 20 pascals (0.15 Torr) will be at least 10 minutes to allow sufficient time in the region of critical pressure. The pressure will be further reduced from 20 pascals for operating equipment, or from atmospheric for equipment which does not operate during ascent, to 13.3 millipascals (10^{-4} Torr) at a rate that simulates the ascent profile to the extent practicable. For launch vehicle units, the vacuum pressure will be modified to reflect an altitude consistent with the maximum service altitude.

b. Temperature. The unit hot and cold temperatures will be the same as those specified in 6.4.2.3b. An exception is made for a propulsion unit in contact with propellant for which the cold temperature will be limited to 3°C above the propellant freezing temperature. The transitions between hot and cold should be at an average rate simulating flight conditions.

c. Duration. The number of thermal cycles will be as given in Table VI. Thermal soak durations (3.5.11) will be a minimum of 6 hours at the hot and 6 hours at the cold temperature during the first and last cycle. Intermediate cycles will have at least 1-hour soaks at the hot and cold temperatures with power turned on. Measurement of thermal soak durations will begin at the time of unit turn-on (Figure 1).

6.4.3.4 Supplementary Requirements. The 25-cycle test is applicable to units containing electrical or electronic elements where environmental stress screening is imposed for acceptance testing. For nonelectrical and nonelectronic units, the 6-cycle test applies (Table VI).

Functional tests will be conducted after unit temperatures have stabilized at the hot and cold temperatures during the first and last cycle, and after return of the unit to ambient temperature in vacuum. During the remainder of the test, electrical and electronic units, including all redundant circuits and paths, will be cycled through various operational modes. Perceptive parameters will be monitored for failures and intermittents to the maximum extent practicable. Units will meet their performance requirements within specifications over the maximum expected temperature range extended by 10°C at the hot and cold limits.

For moving mechanical assemblies, performance parameters (such as current draw, resistance torque or force, actuation time, velocity or acceleration) will be monitored. Where practicable, force or torque margins will be determined on moving mechanical assemblies at the temperature extremes. Where this is not practicable, minimum acceptable force or torque margin will be demonstrated. Compatibility with operational fluids will be verified at test temperature extremes for valves, propulsion units, and other units as appropriate.

6.4.4 Vibration Test, Unit Qualification

6.4.4.1 Purpose. The vibration test demonstrates the ability of the unit to endure a maximum duration of corresponding acceptance testing and then meet requirements during and after exposure to the extreme expected dynamic environment in flight (3.3.5).

6.4.4.2 Test Description. The unit will be mounted to a fixture through the normal mounting points of the unit. The same test fixture should be used in the qualification and acceptance vibration tests. Attached wiring harnesses and hydraulic and pneumatic lines up to the first attachment point, instrumentation, and other connecting items should be included as in the flight configuration. Such a configuration will be required when units that employ shock or vibration isolators are tested on their isolators. The suitability of the fixture and test control means will have been established prior to the qualification testing (6.4.4.5). The unit will be tested in each of 3 orthogonal axes. Units required to operate under pressure during ascent will be pressurized to simulate flight conditions, from structural and leakage standpoints, and monitored for pressure decay. Units designed for operation during ascent, and whose maximum or minimum expected temperatures fall outside the normal temperature range (7.1.1), are candidates for combined vibration and temperature testing. When such testing is employed, units will be conditioned to be as close to the worst-case flight temperature as is practicable and monitored for temperature during vibration exposure.

Units mounted on shock or vibration isolators will typically require vibration testing at qualification levels in two configurations. A first configuration is with the unit hard-mounted to qualify for the acceptance-level testing if, as is typical, the acceptance testing is performed without the isolators present. The second configuration is with the unit mounted on the isolators to qualify for the flight environment. The unit will be mounted on isolators of the same lot as those used in service, if practicable. Units mounted on isolators will be controlled at the locations where the isolators are attached to the structure. Hard-mounted units will be controlled at the unit mounting attachments.

6.4.4.3 Test Level and Duration. The test will be conducted per 6.1.4. For hard-mounted units, a typical version of the test involves accelerated acceptance-level testing per 6.1.4.1 and applies the qualification level spectrum for 3 minutes per axis. This is based on a qualification margin of 6 dB, a maximum of 6 minutes of accumulated acceptance testing on a flight unit, and a fatigue equivalent duration in flight (3.3.3) of not greater than 15 seconds. Operating time should be divided approximately equally between redundant functions. When insufficient test time is available at the full test level to test redundant circuits, functions, and modes, extended testing using a spectrum no lower than 6 dB below the qualification spectrum will be conducted as necessary to complete functional testing.

6.4.4.4 Supplementary Requirements. During the test, all electrical and electronic units will be electrically energized and functionally sequenced through various operational modes to the maximum extent practicable. This includes all redundant circuits, and all circuits that do not operate during launch. Several perceptive parameters will be monitored for failures or intermittents during the test. Continuous monitoring of the unit, including the main bus by a power transient monitoring device, will be provided to detect intermittent failures. When necessary to prevent unrealistic input forces or unit responses for units whose mass exceeds 23 kilograms (50 pounds), the spectrum may be limited or notched, but not below the minimum test spectrum for a unit (7.1.3). The vibration test does not apply to a unit having a large surface causing its vibration response to be due predominantly to direct acoustic excitation (6.4.5).

6.4.4.5 Fixture Evaluation. The vibration fixture will be verified by test to uniformly impart motion to the unit under test and to limit the energy transfer from the test axis to the other two orthogonal axes (crosstalk). The crosstalk levels should be lower than the input for the respective axis. In 1/6-octave bands above 1000 Hz, exceedances of up to 3 dB are allowed provided that the sum of their bandwidths does not exceed 300 Hz in a cross axis. The dynamic test configuration (fixture and unit) will be evaluated for crosstalk before testing to qualification levels.

6.4.4.6 Special Considerations for Structural Units. Vibration acceptance tests of structural units are normally not conducted because the process controls, inspections, and proof testing that are implemented are sufficient to assure performance and quality. However, to demonstrate structural integrity of structural units having critical fatigue-type modes of failure, with a low fatigue margin, a vibration qualification test will be conducted. The test duration will be 4 times the fatigue equivalent duration in flight at the extreme expected level (3.3.5). When a structural unit is not subjected to a static strength qualification test, a brief random vibration qualification test will be conducted with an exposure to 3 dB above the extreme expected level. The duration will be that necessary to achieve a steady-state response, but not less than 10 seconds, to demonstrate that ultimate strength requirements are satisfied.

6.4.5 Acoustic Test, Unit Qualification

6.4.5.1 Purpose. The acoustic test demonstrates the ability of a unit having large surfaces, whose vibration response is due predominantly to direct acoustic excitations, to endure a maximum duration of acoustic acceptance testing and then meet requirements during and after exposure to the extreme expected dynamic environment in flight (3.3.4). For such units, the acoustic test will be conducted and the vibration test (6.4.4) is discretionary.

6.4.5.2 Test Description. The unit in its ascent configuration will be installed in an acoustic test facility capable of generating sound fields or fluctuating surface pressures that induce unit vibration environments sufficient for unit qualification. The unit should be mounted on a flight-type support structure or reasonable simulation thereof. Significant fluid and pressure conditions will be replicated to the extent practicable. Appropriate dynamic instrumentation will be installed to measure vibration responses. Control microphones will be placed at a minimum of 4 well-separated locations at one half the distance from the test article to the nearest chamber wall, but no closer than 0.5 meter (20 inches) to both the test article surface and the chamber wall.

6.4.5.3 Test Level and Duration. Same as 6.2.4.3 except the qualification test duration will be 3 minutes based on a maximum of 6 minutes of accumulated acceptance testing on a flight unit.

6.4.5.4 Supplementary Requirements. Same as 6.2.4.4.

6.4.6 Shock Test, Unit Qualification

6.4.6.1 Purpose. The shock test demonstrates the capability of the unit to meet requirements during and after exposure to the extreme expected shock environment in flight (3.3.7).

6.4.6.2 Test Description. The unit will be mounted to a fixture through the normal mounting points of the unit. The same test fixture should be used in the qualification and acceptance shock tests. If shock isolators are to be used in service, they will be installed. The selected test method will be capable of meeting the required shock spectrum with a transient that has a duration comparable to the duration of the expected shock in flight. A mounting of the unit 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. Sufficient prior development of the test mechanism will have been carried out to validate the proposed test method before testing qualification hardware. The test environment will comply with the following conditions:

a. A transient having 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 will be a goal for the duration of the shock transient to approximate the duration of the service shock event. In general, the duration of the shock employed for the shock spectrum analysis will not exceed 20 milliseconds.

6.4.6.3 Test Level and Exposure. The shock spectrum in each direction along each of the 3 orthogonal axes will be at least the qualification level for that direction. For vibration or shock isolated units, the lower frequency limit of the response spectrum will be below 0.7 times the natural frequency of the isolated unit. A sufficient number of shocks will be imposed to meet the amplitude criteria in both directions of each of the 3 orthogonal axes at least 3 times the number of significant events at that unit location. A significant event for the unit being qualified is one that produces a maximum expected shock spectrum within 6 dB of the envelope of maximum expected spectra (3.3.7) from all events.

6.4.6.4 Supplementary Requirements. Electrical and electronic units, including redundant circuits, will be energized and monitored to the maximum extent practicable, including those that are not normally operating during the service shock. A functional test will 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 will not transfer and will not chatter in excess of specification limits during the shock test.

A shock qualification test is not required along any axis for which both the following are satisfied:

a. The qualification random vibration test spectrum when converted to an equivalent shock response spectrum (3-sigma response for Q = 10) exceeds the qualification shock spectrum requirement at all frequencies below 2000 Hz.

b. The maximum expected shock spectrum above 2000 Hz does not exceed g values equal to 0.8 times the frequency in Hz at all frequencies above 2000 Hz, corresponding to a velocity of 1.27 meters/second (50 inches/second).

6.4.7 Leakage Test, Unit Qualification

6.4.7.1 Purpose. The leakage test demonstrates the capability of pressurized components and hermetically sealed units to meet the specified design leakage rate requirements.

6.4.7.2 Test Description. An acceptable leak rate to meet mission requirements is based upon development tests and appropriate analyses. An acceptable measurement technique is one that accounts for leak rate variations with differential pressure and hot and cold temperatures and has the required threshold, resolution, and accuracy to detect any leakage equal to or greater than the maximum acceptable leak rate. Consideration should be given to testing units at differential pressures greater or less than the maximum or minimum operating differential pressure to provide some assurance of a qualification margin for leakage. If appropriate, the leak rate test will be made at qualification hot and cold temperatures with the representative fluid to account for geometry alterations and viscosity changes.

6.4.7.3 Test Level and Duration. Unless otherwise specified, the leakage tests will be performed with the unit pressurized at the maximum differential operating pressure, as well as at the minimum differential operating pressure if the seals are dependent upon pressure for proper sealing. The test duration will be sufficient to detect any significant leakage.

6.4.8 Pressure Test, Unit Qualification

6.4.8.1 Purpose. The pressure test demonstrates adequate margin, so that structural failure does not occur before the design burst pressure is reached, or excessive deformation does not occur at the maximum expected operating pressure (MEOP).

6.4.8.2 Test Description

a. Proof Pressure Test. For items such as pressurized structures and pressure components, a proof test with a minimum of 1 cycle of proof pressure will be conducted. Evidence of either leakage, a permanent set or distortion that exceeds a drawing tolerance, or failure of any kind will constitute failure to pass the test.

b. Pressure Cycle Test. For pressurized structures and pressure vessels, a pressure cycle test will be conducted. Requirements for application of external loads in combination with internal pressures during testing will be evaluated based on the relative magnitude and on the destabilizing effect of stresses due to the external load. If limit combined tensile stresses are enveloped by the test pressure stress, the application of external load is not required.

c. Burst Test. The pressure will be increased to the design burst pressure, while simultaneously applying the ultimate external load(s), if appropriate. The internal pressure will be applied at a sufficiently slow rate that dynamic stresses are negligible. For pressure vessels, after demonstrating no burst at the design burst pressure, the pressure will be increased to actual burst of the vessel, and the actual burst pressure will be recorded.

6.4.8.3 Test Levels and Durations.

a. Temperature and Humidity. The test temperature and humidity conditions will be consistent with the critical-use temperature and humidity. As an alternative, tests may be conducted at ambient conditions if the test pressures are suitably adjusted to account for temperature and humidity effects on material strength and fracture toughness.

b. Proof Pressure. Unless otherwise specified, the minimum proof pressure for pressurized structures will be 1.1 times the MEOP. For pressure vessels, and other pressure components such as lines and fittings, the minimum proof pressure will comply with the requirements specified in MIL-STD-1522. The pressure will be maintained for a time just sufficient to assure that the proper pressure was achieved. Except that for pressure vessels, the hold time will be a minimum of 5 minutes unless otherwise specified.

c. Pressure Cycle. Unless otherwise specified, the peak pressure for pressurized structures will equal the MEOP during each cycle, and the number of cycles will be 4 times the predicted number of operating cycles or 50 cycles, whichever is greater. For pressure vessels, the test will comply with the requirements specified in MIL-STD-1522.

d. Burst Pressure. Unless otherwise specified, the minimum design burst pressure for pressurized structures will be 1.25 times the MEOP. For pressure vessels and pressure components, the minimum design burst pressure will comply with MIL-STD-1522.

The design burst pressure will be maintained for a period of time just sufficient to assure that the proper pressure was achieved.

6.4.8.4 Supplementary Requirements. Applicable safety standards will be followed in conducting all tests. Unless otherwise specified, the qualification testing of pressure vessels will include a demonstration of a leak-before-burst (LBB) failure mode using pre-flawed specimens as specified in MIL-STD-1522. The LBB pressure test may be omitted if available material data are directly applicable to be used for an analytical demonstration of the leak-before-burst failure mode.

6.4.9 Acceleration Test, Unit Qualification

6.4.9.1 Purpose. The acceleration test demonstrates the capability of the unit to withstand or, if appropriate, to operate in the qualification level acceleration environment.

6.4.9.2 Test Description. The unit will be attached, as it is during flight, to a test fixture and subjected to acceleration in appropriate directions. The specified accelerations apply to the center of gravity of the test item. If a centrifuge is used, the arm (measured to the geometric center of the test item) should be at least 5 times the dimension of the test item measured along the arm. The acceleration gradient across the test item should not result in accelerations that fall below the qualification level on any critical member of the test item. In addition, any over-test condition should be minimized to prevent unnecessary risk to the test article. Inertial units such as gyros and platforms may require counter-rotating fixtures on the centrifuge arm.

6.4.9.3 Test Levels and Duration

a. Acceleration Level. The test acceleration level will be at least 1.25 times the maximum predicted acceleration (3.4.8) for each direction of test. The factor will be 1.4 for manned flight.

b. Duration. Unless otherwise specified, the test duration will be at least 5 minutes for each direction of test.

6.4.9.4 Supplementary Requirements. If the unit is to be mounted on shock or vibration isolators in the vehicle, the unit should be mounted on these isolators during the qualification test.

6.4.10 Life Test, Unit Qualification

6.4.10.1 Purpose. The life test applies to units that may have a wearout, drift, or fatigue-type failure mode, or a performance degradation, such as batteries. The test demonstrates that the units have the capability to perform within specification limits for the maximum duration or cycles of operation during repeated ground testing and in flight.

6.4.10.2 Test Description. One or more units will be operated under conditions that simulate their service conditions. These conditions will be selected for consistency with end-use requirements and the significant life characteristics of the particular unit. Typical environments are ambient, thermal, and thermal vacuum to evaluate wearout and drift failure modes; and pressure, thermal, and vibration to evaluate fatigue-type failure modes. The test will be designed to demonstrate the ability of the unit to withstand the maximum operating time and the maximum number of operational cycles predicted during its service life (3.5.7) with a suitable margin.

6.4.10.3 Test Levels and Durations.

a. Pressure. For pressurized structures and pressure vessels, the pressure level will be that specified in 6.4.8.3c. For other units, ambient pressure will be used except where degradation due to a vacuum environment may be anticipated, such as for some unsealed units. In those cases, a pressure of 13.3 millipascals (10^{-4} Torr) or less will be used.

b. Environmental Levels. The extreme expected environmental levels will be used. Higher levels may be used to accelerate the life testing, provided that the resulting increase in the rate of degradation is well established and that unrealistic failure modes are not introduced.

c. Duration. For pressurized structures and pressure vessels, the duration will be that specified in 6.4.8.3c. For other units, the total operating time or number of operational cycles will be at least 2 times that predicted during the service life (3.5.7), including ground testing, in order to demonstrate an adequate margin. For a structural component having a fatigue-type failure mode that has not been subjected to a vibration qualification test, the test duration will be at least 4 times the specified service life.

d. Functional Duty Cycle. Complete functional tests will be conducted before the test begins and after completion of the test. During the life test, functional tests will be conducted in sufficient detail, and at sufficiently short intervals, so as to establish trends.

6.4.10.4 Supplementary Requirements. For statistically-based life tests, the duration is dependent upon the number of samples, confidence, and reliability to be demonstrated. The mechanisms in each unit that are subjected to wearout should be separately tested. For these mechanisms, the duration of the life test should assure with high confidence that the mechanisms will not wear out during their service life. At the end of the life test, mechanisms and moving mechanical assemblies will be disassembled and inspected for anomalous conditions. The hardware may be disassembled and inspected earlier if warranted. The critical areas of parts that may be subject to fatigue failure will be inspected to determine their integrity.

6.4.11 Electromagnetic Compatibility (EMC) Test, Unit Qualification

6.4.11.1 Purpose. The electromagnetic compatibility test will demonstrate that the electromagnetic interference characteristics (emission and susceptibility) of the unit, under normal operating conditions, do not result in malfunction of the unit. It also demonstrates that the unit does not emit, radiate, or conduct interference which could result in malfunction of other units.

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

6.4.12 Climatic Tests, Unit Qualification

6.4.12.1 Purpose. These tests demonstrate that the unit is capable of surviving exposure to various climatic conditions without excessive degradation, or operating during exposure, as applicable. Exposure conditions include those imposed upon the unit during fabrication, test, shipment, storage, preparation for launch, launch itself, and reentry if applicable. These can include such conditions as humidity, sand and dust, rain, salt fog, and explosive atmosphere. Degradation due to fungus, ozone, and sunshine will be verified by design and material selection.

It is the intent that environmental design of flight hardware not be unnecessarily driven by terrestrial natural environments. To the greatest extent feasible, the flight hardware will be protected from the potentially degrading effects of extreme terrestrial natural environments by procedural controls and special support equipment. Only those environments that cannot be controlled need be considered in the design and testing.

6.4.12.2 Humidity Test, Unit Qualification

6.4.12.2.1 Purpose. The humidity test demonstrates that the unit is capable of surviving or operating in, if applicable, warm humid environments. In the cases where exposure is controlled throughout the life cycle to conditions with less than 55% relative humidity, and the temperature changes do not create conditions where condensation occurs on the hardware, then verification by test is not required.

6.4.12.2.2 Test Description and Levels. For units exposed to unprotected ambient conditions, the humidity test will conform to the method given in MIL-STD-810. For units located in protected, but uncontrolled environments, the unit will be installed in a humidity chamber and subjected to the following conditions (time line illustrated in Figure 2):

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

b. **Cycle 1.** The temperature will be increased to +35^{m}C over a 1-hour period; then the humidity will be increased to not less than 95% over a 1-hour period with the temperature maintained at +35^{m}C. These conditions will be maintained for 2 hours. The temperature will then be reduced to +2^{m}C over a 2-hour period with the relative humidity stabilized at not less than 95%. These conditions will be maintained for 2 hours.

c. **Cycle 2.** Cycle 1 will be repeated except that the temperature will be increased from +2°C to +35°C over a 2-hour period; moisture is not added to the chamber until +35°C is reached.

d. **Cycle 3.** The chamber temperature will be increased to +35°C over a 2-hour period without adding any moisture to the chamber. The test unit will then be dried with air at room temperature and 50% maximum relative humidity by blowing air through the chamber for 6 hours. The volume of air used per minute will be equal to 1 to 3 times the test chamber volume. A suitable container may be used in place of the test chamber for drying the test unit.

e. **Cycle 4.** If it had been removed, the unit will be placed back in the test chamber and the temperature increased to +35°C and the relative humidity increased to 90% over a 1-hour period; and these conditions will be maintained for at least 1 hour. The temperature will then be reduced to +2°C over a 1-hour period with the relative humidity stabilized at 90%; and these conditions will be maintained for at least 1 hour. A drying cycle should follow (see Cycle 3).

6.4.12.2.3 Supplementary Requirements. The unit will 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 unit will be functionally tested during the Cycle 4 periods of stability, after the 1-hour period to reach +35°C and 90% relative humidity, and again after the 1-hour period to reach the +2°C and 90% relative humidity.

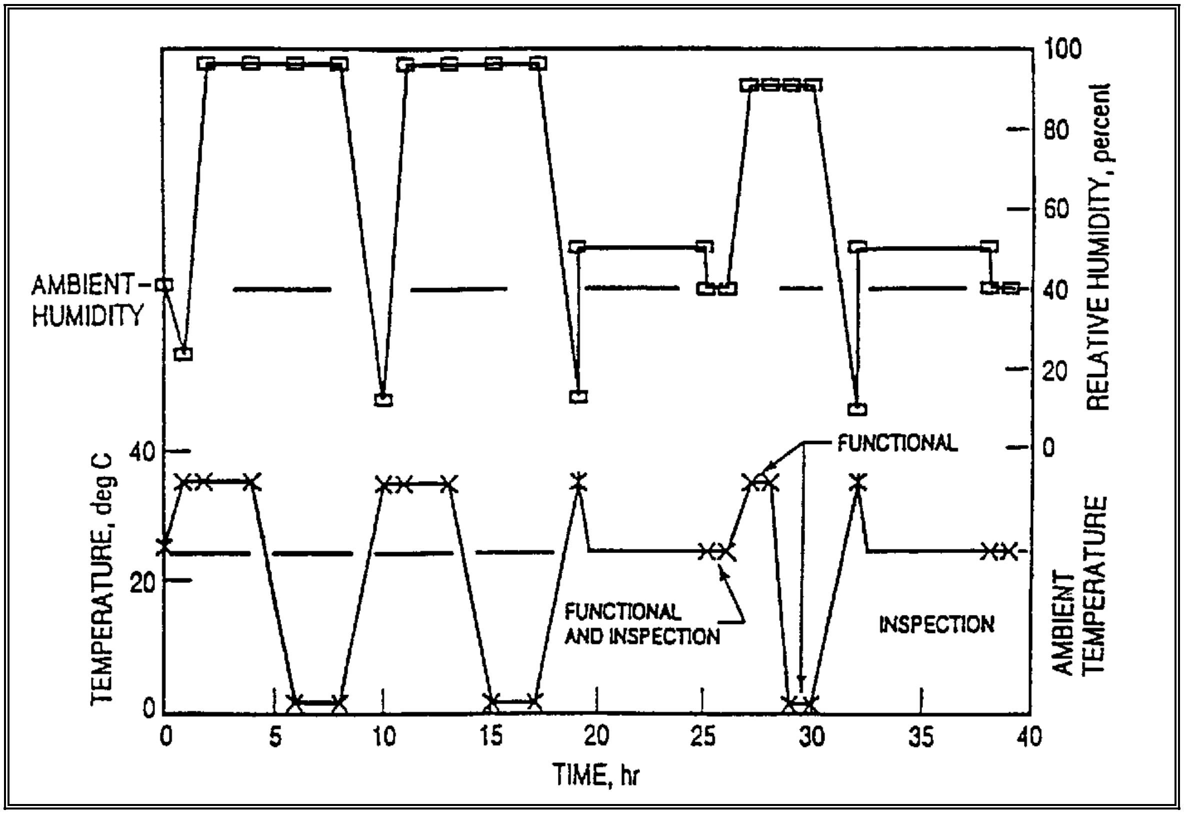


Figure 2. Humidity Test Time Line.

6.4.12.3 Sand and Dust Test, Unit Qualification

6.4.12.3.1 Purpose. The sand and dust test is conducted to determine the resistance of units to blowing fine sand and dust particles. This test will not be required for units protected from sand and dust by contamination control, protective shipping and storage containers, or covers. However, in those cases, rain testing demonstrating the adequacy of the protective shelters, shipping and storage containers, or covers, as applicable, may be required instead of a test of the unit itself.

6.4.12.3.2 Test Description. The test requirements for the sand and dust test will conform to the method given in MIL-STD-810.

6.4.12.4 Rain Test, Unit Qualification

6.4.12.4.1 Purpose. The rain test will be conducted to determine the resistance of units to rain. Units protected from rain by protective shelters, shipping and storage containers, or covers, will not require verification by test.

6.4.12.4.2 Test Description. Buildup of the unit, shelter, container, or the cover being tested will be representative of the actual fielded configuration without any duct tape or temporary sealants. The initial temperature difference between the test item and the spray water will be a minimum of 1°C. For temperature-controlled containers, the temperature difference between the test item and the spray water will at least be that between the maximum control temperature and the coldest rain condition in the field. Nozzles used will produce a square spray pattern or other overlapping pattern (for maximum surface coverage) and droplet size predominantly in the 2 to 4.5 millimeter range at approximately 375 kilopascals gage pressure (40 psig). At least one nozzle will be used for each approximately 0.5 square meter (6 ft^2) of surface area and each nozzle will be positioned at 0.5 meter (20 inches) from the test surface. All exposed faces will be sprayed for at least 40 minutes. The interior will be inspected for water penetration at the end of each 40-minute exposure. Evidence of water penetration will constitute a failure.

6.4.12.5 Salt Fog Test, Unit Qualification

6.4.12.5.1 Purpose. The salt fog test is used to demonstrate the resistance of the unit to the effects of a salt spray atmosphere. The salt fog test is not required if the flight hardware is protected against the salt fog environment by suitable preservation means and protective shipping and storage containers.

6.4.12.5.2 Test Description. The requirements for the salt fog test will conform to the method given in MIL-STD-810.

6.4.12.6 Explosive Atmosphere Test, Unit Qualification

6.4.12.6.1 Purpose. The explosive atmosphere test is conducted to demonstrate unit operability in an ignitable fuel-air mixture without igniting the mixture.

6.4.12.6.2 Test Description. The test requirements for the explosive atmosphere test will conform to the method given in MIL-STD-810.

SECTION 7.

ACCEPTANCE TESTS

7.1 GENERAL ACCEPTANCE TEST REQUIREMENTS

Acceptance tests will be conducted as required to demonstrate the acceptability of each deliverable item. The tests will demonstrate conformance to specification requirements and provide quality-control assurance against workmanship or material deficiencies. Acceptance testing is intended to stress screen items to precipitate incipient failures due to latent defects in parts, materials, and workmanship. However, the testing will not create conditions that exceed appropriate design safety margins or cause unrealistic modes of failure. If the equipment is to be used by more than one program or in different vehicle locations, the acceptance test conditions should envelope those of the various programs or vehicle locations involved. Typical acceptance test levels and durations are summarized in Table XI, and are detailed in subsequent paragraphs.

The test baseline will be tailored for each program, giving consideration to both the required and other tests (3.5.5). For special items, such as some tape recorders and certain batteries, the specified acceptance test environments would result in physical deterioration of materials or other damage. In those cases, less severe acceptance test environments that still satisfy the system operational requirements will be used.

7.1.1 Temperature Range and Number of Thermal Cycles, Acceptance Tests. Two requirements on the unit acceptance temperature range (Figure 3) are:

a. The range will encompass the maximum and minimum expected temperatures (3.3.1).

b. The range should be as large as practicable to meet environmental stress screening purposes. A range of 105°C is recommended, and is the basis used in Tables V and VI.

For units, the range from -44 to +61°C is recommended if requirement "a" is satisfied. The number of cycles will be in compliance with Table VI. If this 105°C temperature range, plus the 1-°C hot and cold extension for qualification, gives rise to unrealistic failure modes or unrealistic design requirements, the range may be shifted or reduced to the extent necessary. To compensate for a reduced range, the number of thermal cycles for acceptance tests will then be increased per note 3 of Table VI. For units exposed to cryogenic temperatures, acceptance temperature limits will encompass the highest and lowest temperatures with appropriate uncertainty margins (Table II). For units which do not contain electrical or electronic elements, the minimum acceptance test will be 1 thermal vacuum cycle in accordance with 7.4.3.

For vehicle thermal vacuum tests, at least one unit will reach its acceptance hot temperature during hot soaks. During cold soaks at least one unit will reach its acceptance cold temperature. If the ambient pressure thermal cycle alternative test is selected, the minimum temperature range will be 50^{m}C. The number of thermal vacuum and thermal cycles are specified in Table VI

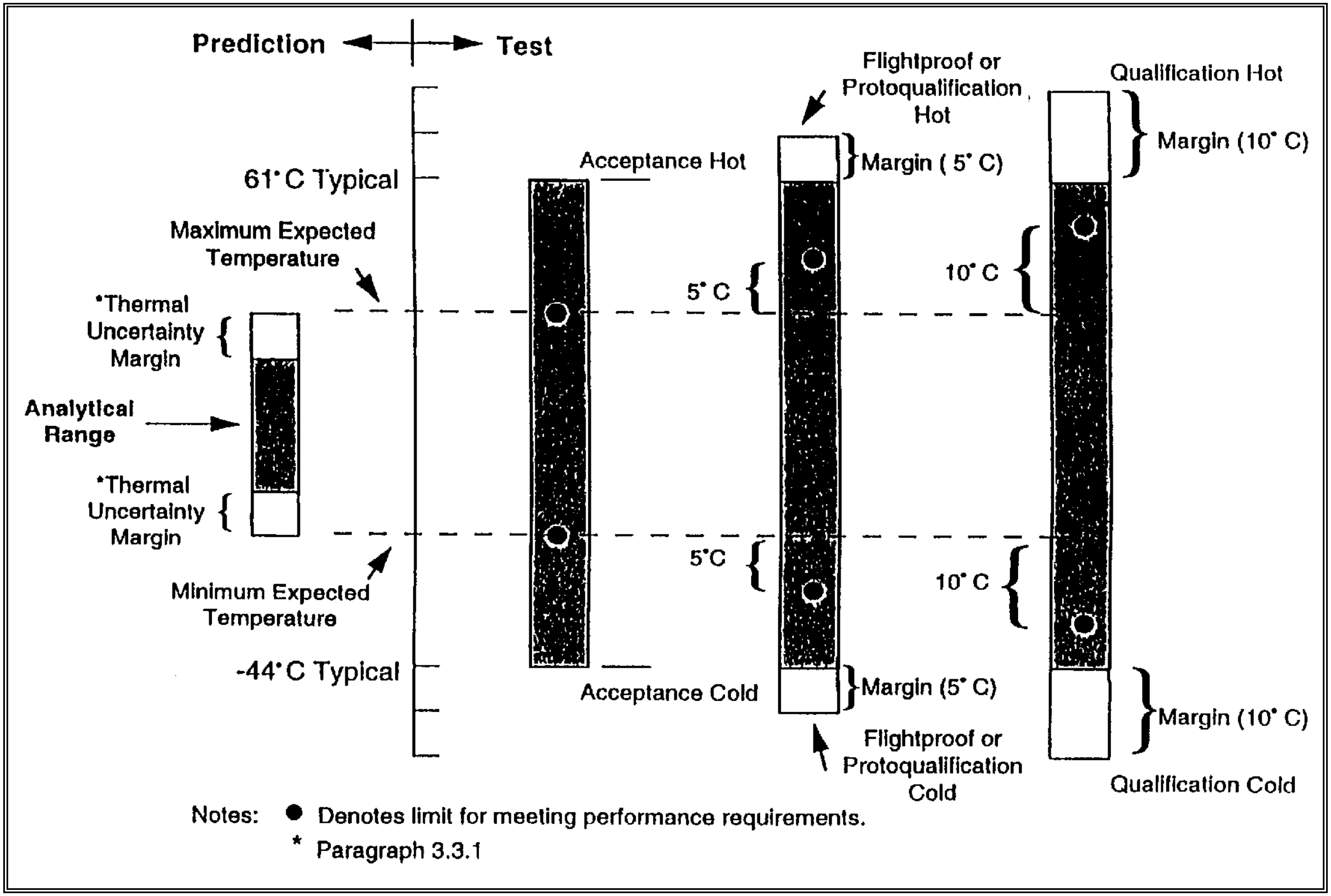


Figure 3. Unit Predicted and Test Temperature Ranges.

Table XI. Typical Acceptance Test Levels and Durations.

Test	Units	Vehicles
Shock	Maximum expected spectrum (3.3.7), achieved once in both directions of 3 axes. Discretionary if spectrum is low (7.4.6.4).	1 activation of significant shock-producing events (7.2.3.3).
Acoustic	Same as for vehicles.	Envelope of maximum expected spectrum (3.3.4) and minimum spectrum (Figure 4), 1 minute.
Vibration	Envelope of maximum expected spectrum (3.3.5) and minimum spectrum (Figure 5), 1 minute in each of 3 axes.	Same as for units, except minimum spectrum in Figure 6.
Thermal Vacuum*	1 cycle, -44 to +61°C (7.1.1). Vacuum at 13.3 millipascals (10^{-4} Torr).	4 cycles, -44 to +61°C (7.2.8). Same pressure as for units.
Thermal Cycle*	12.5 cycles, -44 to +61°C.	See 7.2.7.
Combined Thermal Vacuum and Cycle*	8.5 thermal cycles and 4 thermal vacuum cycles, -44 to +61°C.	See 7.2.7.
Proof Load	For bonded structures and structures made of composite material, or having sandwich construction: 1.1 times limit load.	Same as for units, but only tested at subsystem level.
Proof Pressure	For pressurized structures, 1.1 times the MEOP. For pressure vessels and other pressure components, comply with MIL-STD-1522.	Same as for units.

*See Tables V and VI.

7.1.2 <u>Acoustic Environment, Acceptance Tests</u>. The acceptance test acoustic spectrum will be the maximum expected environment (3.3.4), but not less than the minimum free-field spectrum in Figure 4. The minimum duration of the acceptance acoustic test is 1 minute.

7.1.3 Vibration Environment, Acceptance Tests. The acceptance test random vibration spectrum will be the maximum expected environment (3.3.5), but not below the minimum spectrum in Figure 5 for a unit or below the minimum spectrum in Figure 6 for a vehicle. The minimum spectrum for a unit whose mass exceeds 23 kilograms (50 pounds) should be evaluated on an individual basis. The acceptance sinusoidal vibration amplitude, if significant, will be that of the maximum expected sinusoidal vibration environment (3.3.6). When concurrent random and sinusoidal vibration during service life (3.5.7) can be more severe than either considered separately, an appropriate combination of the two types of vibration should be used for the test. The minimum duration of the acceptance random vibration test will be 1 minute for each of 3 orthogonal axes.

7.1.4 Storage Tests: Vehicle, Subsystem, or Unit Acceptance. Storage test requirements consist of appropriate testing after storage (such as vibration, thermal, and static load or pressure) based on the vehicle design, and the duration and conditions of storage. Items having age-sensitive material may require periodic retesting and those having rotating elements may require periodic operation.

7.2 VEHICLE ACCEPTANCE TESTS

The vehicle acceptance test baseline will include all the required tests specified in Table XII. The "other" tests (3.5.5) deemed applicable, and any special tests for the vehicle element (such as alignments, instrument calibrations, antenna patterns, and mass properties) will also be conducted as part of acceptance testing. If the vehicle is controlled by on-board data processing, the flight version of the computer software will be resident in the vehicle computer for these tests. The verification of the operational requirements will be demonstrated in these tests to the extent practicable.

Table XII. Vehicle Acceptance Test Baseline.

TEST	REFERENCE PARAGRAPH	SUGGESTED SEQUENCE	LAUNCH VEHICLE	UPPER STAGE	SPACE VEHICLE
Inspection[1]	4.5	1	R	R	R
Functional[1]	7.2.1	2	R	R	R
Pressure/Leak	7.2.6	3,7,10	R	R	R
EMC	7.2.2	4	—	O	O
Shock	7.2.3	5	O	O	O
Acoustic[2] or Vibration	7.2.4 or 7.2.5	6	O	R	R
Thermal Cycle	7.2.7	8	O	O	O
Thermal Vac[3]	7.2.8	9	O	R	R
Storage	7.1.4	any	O	O	O

Recommended vehicle acceptance requirements. (3.5.5)
R = baseline requirement (high probability of being required)

O = "other" (low probability of being required)
— = not required (negligible probability of being required).

Notes:
1 Required before and following each test as appropriate. Include special tests as applicable (7.2).
2 Vibration conducted in place of acoustic test for a compact vehicle, typically with mass less than 180 kg (400 lb).
3 Requirements modified if thermal cycle test (7.2.7) conducted.

7.2.1 Functional Test, Vehicle Acceptance

7.2.1.1 Purpose. The functional test verifies that the electrical and mechanical performance of the vehicle meets the performance requirements of the specifications and detects any anomalous condition.

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

7.2.1.3 Electrical and Fiber-optic Circuit Functional Test. Same as the electrical functional test for vehicle qualification (6.2.1.3), except that tests are limited to critical functions and are only necessary at nominal operational conditions. The final ambient functional test conducted prior to shipment of the 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 its critical features can be duplicated, as nearly as practicable, at the launch base. The results of all factory functional tests, and of those conducted at the launch base, will be used for trend analysis.

7.2.1.4 Supplementary Requirements. Same as 6.2.1.4.

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

7.2.3 Shock Test, Vehicle Acceptance

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

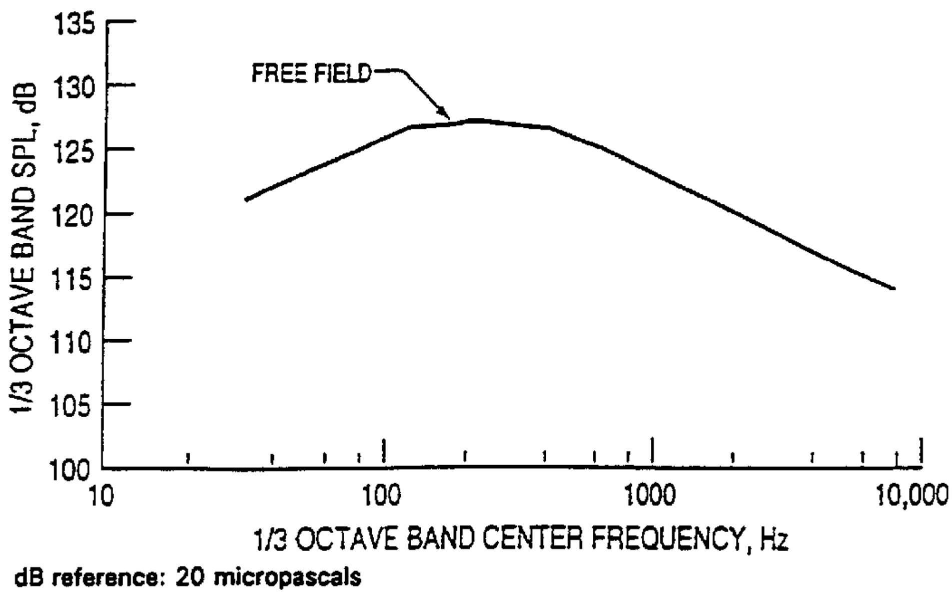
7.2.3.2 Test Description. Same as 6.2.3.2, except that the dynamic instrumentation may be reduced.

7.2.3.3 Test Activations. Shock acceptance testing of vehicles should be performed in those instances deemed advisable due to severity of the environment or susceptibility of the design. One activation of those events causing significant shocks to critical and shock sensitive units should be conducted. Firing of both primary and redundant explosive-ordnance devices is required in the same relationship as they are to be used in flight. However, when the structure is explosively severed, as in the case of a shaped charge, such testing is discretionary. To aid in fault detection, the shock test should be conducted with subsystems operating and monitored to the greatest extent practicable.

7.2.4 Acoustic Test, Vehicle Acceptance

7.2.4.1 Purpose. The acoustic test simulates the flight or minimum workmanship-screen acoustic environment and the induced vibration on units in order to expose material and workmanship defects that might not be detected in a static test condition. It also serves as an acceptance test for functional subsystems, units, and interconnection items that have not been previously acceptance tested.

7.2.4.2 Test Description. Same as 6.2.4.2, except that the dynamic instrumentation may be reduced.

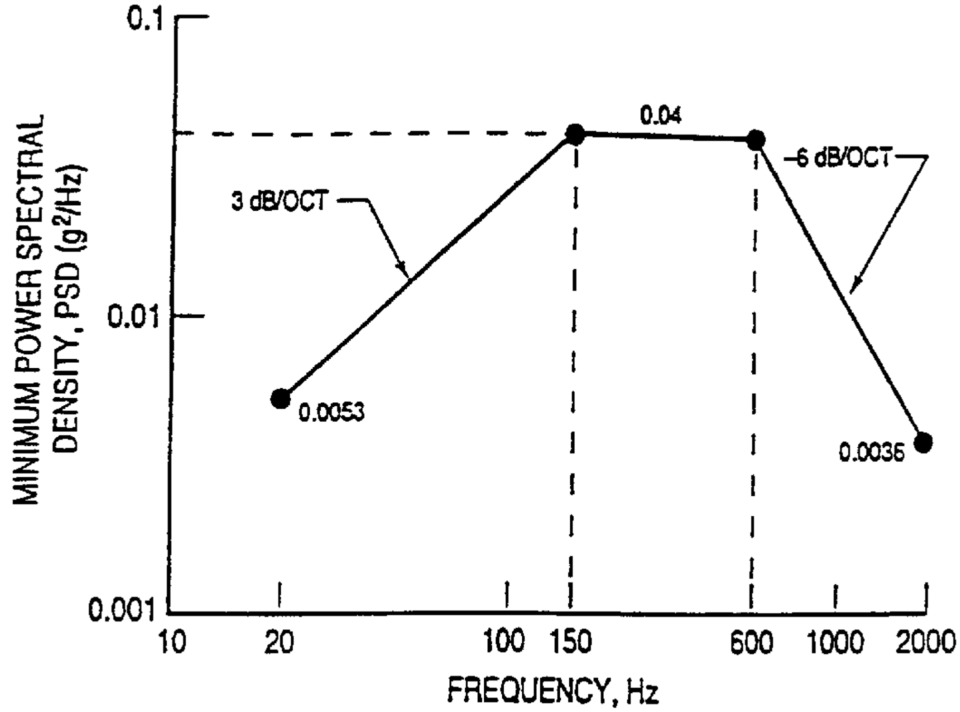


dB reference: 20 micropascals

Curve Values

1/3-Octave-Band Center Frequency (Hz)	Minimum Sound Pressure Level (dB)	1/3-Octave-Band Center Frequency (Hz)	Minimum Sound Pressure Level (dB)
31	121	630	125
40	122	800	124
50	123	1000	123
63	124	1250	122
80	125	1600	121
100	125.7	2000	120
125	126.5	2500	119
160	126.7	3150	118
200	127	4000	117
250	127	5000	116
315	126.7	6300	115
400	126.5	8000	114
500	125.7	10000	113
		Overall	138

Figure 4. Minimum Free-field Acoustic Spectrum,Vehicle and Unit Acceptance Tests.

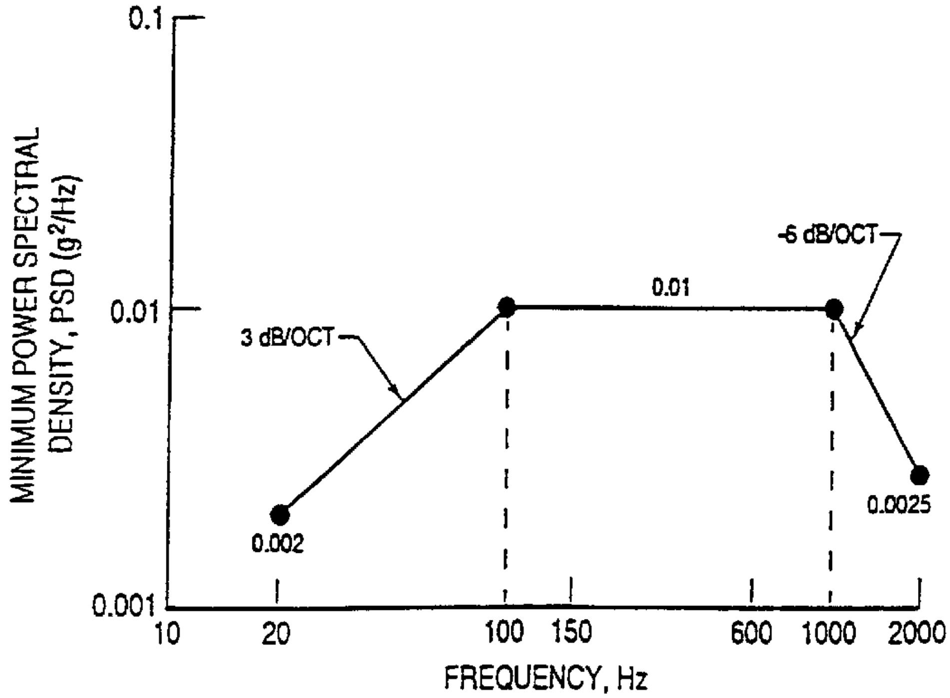


Curve Values	
Frequency (Hz)	Minimum PSD (g^2/Hz)
20	0.0053
20 to 150	+3 dB per octave slope
150 to 600	0.04
600 to 2000	–6 dB per octave slope
2000	0.0036

The overall acceleration level is 6.1 grms.

Note: This spectrum applies only to <u>electrical and electronic</u> units whose mass does not exceed 23 kilograms (50 pounds).

Figure 5. Minimum Random Vibration Spectrum, Unit Acceptance Tests.



Curve Values	
Frequency (Hz)	Minimum PSD (g^2/Hz)
20	0.002
20 to 100	+3 dB per octave slope
100 to 1000	0.01
1000 to 2000	–6 dB per octave slope
2000	0.0025

The overall acceleration level is 3.8 grms.

Figure 6. Minimum Random Vibration Spectrum, Vehicle Acceptance Tests.

7.2.4.3 Test Level and Duration. The acoustic environment will be as defined in 7.1.2. Operating time for launch operating elements should be divided approximately equally between redundant units. Where insufficient time is available to test redundant units, functions, and modes that are operating during the launch, ascent, or reentry phase, extended testing will be at a level no lower than 6 dB below the acceptance level.

7.2.4.4 Supplementary Requirements. Same as 6.2.4.4, except only units that are operating or pressurized during launch, ascent, or reentry phase need be energized and sequenced through operational modes.

7.2.5 Vibration Test, Vehicle Acceptance

7.2.5.1 Purpose. Same as 7.2.4.1. The vibration test may be conducted in lieu of an acoustic test (7.2.4) for a compact vehicle which can be excited more effectively via interface vibration than by an acoustic field. Such vehicles typically have a mass below 180 kilograms (400 pounds).

7.2.5.2 Test Description. Same as 6.2.5.2, except that dynamic instrumentation may be reduced.

7.2.5.3 Test Level and Duration. The random vibration environment will be as defined in 7.1.3. When necessary to prevent excessive input forces or unit responses, the spectrum at the vehicle input may be limited or notched, but not below the minimum spectrum in Figure 6. Vibration will be applied in each of the 3 orthogonal axes as tested for qualification. Where insufficient time is available to test redundant circuits, functions, and modes that are operating during the launch, ascent, or reentry phase, extended testing will be at a level no lower than 6 dB below the acceptance level.

7.2.5.4 Supplementary Requirements. Same as 6.2.5.4, except only units that are operating or pressurized during the launch, ascent, or reentry phase need be energized and sequenced through operational modes.

7.2.6 Pressure and Leakage Tests, Vehicle Acceptance

7.2.6.1 Purpose. The pressure and leakage test demonstrates the capability of fluid subsystems to meet the specified flow, pressure, and leakage requirements.

7.2.6.2 Test Description. Same as 6.2.6.2.

7.2.6.3 Test Levels and Durations.

a. Same as 6.2.6.3a.

b. Same as 6.2.6.3b, except only 1 pressure cycle.

7.2.6.4 Supplementary Requirements. Same as 6.2.6.4.

7.2.7 Thermal Cycle Test, Vehicle Acceptance

7.2.7.1 Purpose. The thermal cycle test detects material, process, and workmanship defects by subjecting the vehicle to a thermal cycle environment.

7.2.7.2 Test Description. Same as 6.2.7.2.

7.2.7.3 Test Level and Duration. The minimum temperature range will be 50°C. The average rate of change of temperature from one extreme to the other will be as rapid as practicable. Operating time should be divided approximately equally between redundant circuits. The minimum number of thermal cycles will be 4 (Tables V and VI).

7.2.7.4 Supplementary Requirements. Same as 6.2.7.4. If the thermal cycle test is implemented, only one thermal cycle is required in the thermal vacuum acceptance test specified in 7.2.8.

7.2.8 Thermal Vacuum Test, Vehicle Acceptance

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

7.2.8.2 Test Description. Same as 6.2.9.2.

7.2.8.3 Test Levels and Duration. Temperatures in various equipment areas will 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 unit in each equipment area equals the acceptance test temperature as defined in 7.1.1. For space and upper-stage vehicles, the pressure will be maintained at or below 13.3 millipascals (10^{-4} Torr). For launch vehicles, the pressure will be maintained at equal to or less than the pressure commensurate with the highest possible service altitude.

Operating time should be divided approximately equally between redundant circuits. The thermal vacuum acceptance test will include at least 4 complete hot-cold cycles at the maximum predicted orbital rate of temperature change and have at least an 8-hour soak at the hot and cold temperatures during the first and last cycles. For intermediate cycles, the soak duration at each temperature extreme will be 4 hours minimum. The soak duration will be extended as necessary to test flight operational conditions including redundancy. If the alternate thermal cycle test (7.2.7) is conducted, then only 1 hot-cold thermal vacuum cycle will be conducted with an 8-hour minimum soak duration at hot and cold temperatures (Tables V and VI).

During one cycle, thermal equilibrium will be achieved at both hot and cold temperatures to allow collection of sufficient data to verify the function of any thermostats, louvers, heat pipes, electric heaters, and to assess the control authority of active thermal subsystems.

7.2.8.4 Supplementary Requirements. Same as 6.2.9.4, except that the acceptance temperature limits apply. Performance within specification is not required at temperatures beyond the maximum and minimum expected temperatures.

7.3 SUBSYSTEM ACCEPTANCE TESTS

Except for pressurized subsystems, subsystem-level acceptance tests are considered discretionary. These tests can be effective since failures detected at this level usually are much less costly to correct than are those detected at the vehicle level. Also, certain acceptance tests should be conducted at the subsystem level where this level provides a more perceptive test than would be possible at either the unit or vehicle level. The desirability of conducting these subsystem acceptance tests should be evaluated considering such factors as

a. The relative accessibility of the subsystem and its units.

b. The retest time at the vehicle level.

c. The cost and availability of a subsystem for testing of spare units.

When subsystem level tests are performed, the test requirements are usually based on vehicle-level test requirements.

7.3.1 Proof Load Test, Structural Subsystem Acceptance

7.3.1.1 Purpose. The proof load test will be required for all bonded structures, and structures made of composite material or having sandwich construction. It detects material, process, and workmanship defects that would respond to structural proof loading. The proof load test is not required if a proven nondestructive evaluation method, with well established accept and reject criteria, is used.

7.3.1.2 Test Descriptions. Same as 6.3.1.2, except that every structural element will be subjected to its proof load and not to higher loading.

7.3.1.3 Test Level and Duration

a. Static Load. Unless otherwise specified, the proof load for flight items will be 1.1 times the limit load (3.4.6).

b. Duration. Loads will be applied as closely as practicable to actual flight loading times, with a minimum dwell time sufficient to record test data.

7.3.2 Proof Pressure Test, Pressurized Subsystem Acceptance

7.3.2.1 Purpose. The proof pressure test detects material and workmanship defects that could result in failure of the pressurized subsystem.

7.3.2.2 Test Description. Same as 6.4.8.2a.

7.3.2.3 Test Levels and Duration. Same as 6.4.8.3b.

7.4 UNIT ACCEPTANCE TESTS

The unit acceptance test baseline consists of all the required tests specified in Table XIII. Any special tests, and the "other" tests (3.5.5) deemed applicable, will also be conducted as part of acceptance testing.

Unit acceptance tests will normally be accomplished entirely at the unit level. Acceptance tests of certain units (such as solar arrays, interconnect tubing, radio-frequency circuits, and wiring harnesses) may be partially accomplished at higher levels of assembly.

Where units fall into two or more categories of Table XIII, the required tests specified for each category will be applied. For example, a star sensor may be considered to fit both "Electrical and Electronic Equipment" and "Optical Equipment" categories. In this example, a thermal cycle test would be conducted since it is required for electronic equipment, even though there is no requirement for thermal cycling of optics. Similarly, an electric motor-driven-actuator fits both "Electrical and Electrical Equipment" and "Moving Mechanical Assembly" categories. The former makes thermal cycling a required test, even though this is an "other" test (3.5.5) for the moving mechanical assembly category.

7.4.1 Functional Test, Unit Acceptance

7.4.1.1 Purpose. The functional test verifies that the electrical and mechanical performance of the unit meets the specified operational requirements of the unit.

7.4.1.2 Test Description. Same as 6.4.1.2.

7.4.1.3 Supplementary Requirements. Same as 6.4.1.3.

7.4.2 Thermal Cycle Test, Electrical and Electronic Unit Acceptance. If qualification thermal cycle testing (6.4.2) was conducted in vacuum, the thermal cycle acceptance test will be performed in vacuum and combined with the test of 7.4.3. The combined number of cycles will meet the requirements of Table

7.4.2.1 Purpose. The thermal cycle test detects material and workmanship defects prior to installation of the unit into a vehicle, by subjecting the unit to thermal cycling.

7.4.2.2 Test Description. Same as 6.4.2.2 except, to aid in reaching the cold temperature, the unit may be powered off when the temperature of the unit is at or below its minimum expected temperature (3.3.1).

7.4.2.3 Test Levels and Duration

a. Pressure and Humidity. Same as 6.4.2.3a.

b. Temperature. The hot and cold temperatures will be the acceptance temperature limits (7.1.1).

c. Duration. The minimum number of thermal cycles will be 12.5, the last two of which will be failure free. For units subjected to the thermal vacuum test of 7.4.3, the number of cycles is reduced by the number of thermal vacuum cycles imposed (Table VI). Temperature soak durations (3.5.11) will be a minimum of 6 hours at the hot and 6 hours at the cold temperature during the first and last cycle. For the intermediate cycles, the soaks will be at least 1 hour long. During soak periods, the unit will be turned off until the temperature stabilizes (3.5.8) and then turned on. Measurement of each temperature soak duration will begin at the time of unit start (Figure 1). The transitions between cold and hot temperatures should be at an average rate of 3 to 5°C per minute and will not be slower than 1°C per minute. Additional operation at the hot acceptance temperature will be accumulated so that the combined duration of thermal cycling, thermal vacuum (7.4.3), and the additional hot operation is at least 200 hours. If desired, the added hot operation can be accomplished by extending hot soak durations during thermal or thermal vacuum cycling. The last 100 hours of operation will be failure free. For internally redundant units, the operating hours will consist of at least 150 hours of primary operation and at least 50 hours of redundant operation, The last 50 hours of each will be failure free.

Table XIII. Unit Acceptance Test Baseline.

Test	Reference Paragraph	Suggested Sequence	Electrical and Electronic	Antenna	MMA	Solar Array	Battery	Valve or Propulsion Component	Pressure Vessel or Component	Thruster	Thermal	Optical	Structural Component
Inspection[1]	4.5	1	R	R	R	R	R	R	R	R	R	R	R
Functional[1]	7.4.1	3	R	R	R	R	R	R	R	R	R	R	—
Leakage[3]	7.4.9	4,7,12	R	—	R	—	R	R	R	O	O	—	—
Shock	7.4.6	5	O[4]	—	—	—	—	—	—	—	—	O	—
Vibration	7.4.4	6	R	R[5]	R	R[5]	R[8]	R	O	R	R	R[5]	—
Acoustic	7.4.5	6	O	R[5]	—	R[5]	—	—	—	—	—	R[5]	—
Thermal Cycle	7.4.2	8	R	—	—	—	—	—	—	—	—	—	—
Thermal Vac	7.4.3	9	R[2]	O	R[7]	O	R[8]	R	O	R	R	R	O
Wear-in	7.4.10	2	—	—	R	—	—	R	—	R	—	—	—
Proof Pressure	7.4.8	10	—	—	O	—	O	R	R	O	—	—	—
Proof Load	7.4.7	11	—	—	—	—	—	—	—	—	—	—	O[6]
EMC	7.4.11	13	O	—	—	—	—	—	—	—	—	—	—

Recommended vehicle qualification requirements.

R = baseline requirement (high probability of being required)
O = "other" (low probability of being required; 3.5.5)
— = not required (negligible probability of being required).

Notes:
1 Required before and after each test as appropriate. Include special tests as applicable (6.2).
2 Discretionary for sealed or low-powered components.
3 Applicable only to sealed or pressurized components.
4 Required when shock levels are high (7.4.6.4).
5 Either vibration or acoustic, whichever is more appropriate, with the other discretionary.
6 Test required if composite materials are used. The test may be omitted if proven nondestructive evaluation methods are used with well-established acceptance and reject criteria.
7 Excluding hydraulic components for launch vehicles.
8 Not required for batteries that cannot be recharged after testing.

7.4.2.4 Supplementary Requirements. Same as 6.4.2.4, except that units are only required to meet their performance requirements within specification over the maximum expected temperature range.

7.4.3 Thermal Vacuum Test, Unit Acceptance

7.4.3.1 Purpose. The thermal vacuum test detects material and workmanship defects by subjecting the unit to a thermal vacuum environment.

7.4.3.2 Test Description. Same as 6.4.3.2, except that the space nuclear radiation environment need not be simulated.

7.4.3.3 Test Levels and Duration

a. Pressure. The pressure will be reduced from atmospheric to 13.3 millipascals (10^{-4} Torr) for on-orbit simulation, or to the functionally appropriate reduced pressure, at a rate that simulates the ascent profile, to the extent practicable. For launch vehicle units, the vacuum pressure will be modified to reflect an altitude consistent with the maximum service altitude. For units that are proven to be free of vacuum related failure modes, the thermal vacuum acceptance test may be conducted at ambient pressure.

b. Temperature. The hot and cold temperatures will be the acceptance temperature limits (7.1.1).

c. Duration. The basic requirement, except for electrical and electronic units, is a single cycle with 6-hour hot and cold soaks (Table VI). For electrical and electronic units, a minimum of 4 thermal vacuum cycles will be used (Table VI). Temperature soak durations will be at least 6 hours at the hot temperature and 6 hours at the cold temperature during the first and last cycle. During the two intermediate cycles, the soaks will be 1 hour long. During each soak period, the unit will be turned off until the temperature has stabilized and then turned on. Measurement of temperature soak durations (3.5.11) will begin at the time of unit turn-on (Figure 1).

7.4.3.4 Supplementary Requirements. Functional tests will be conducted at the hot and cold temperatures during the first and last cycle, and after return of the unit to ambient temperature in vacuum. During the remainder of the test, electrical and electronic units, including all redundant circuits and paths, will be cycled through various operational modes. Perceptive parameters will be monitored for failures and intermittents to the maximum extent practicable. Units will meet their performance requirements over the maximum expected temperature range. Units will be operated over the entire acceptance temperature range, although performance within specification is not required if the acceptance test temperatures extend beyond the minimum or maximum expected temperatures.

For moving mechanical assemblies, performance parameters, such as current draw, resistance torque or force, actuation time, velocity or acceleration, will be monitored. Compatibility of thrusters with their operational fluids will be verified at test temperature extremes.

7.4.4 Vibration Test, Unit Acceptance

7.4.4.1 Purpose. The vibration test detects material and workmanship defects by subjecting the unit to a vibration environment.

7.4.4.2 Test Description. Same as 6.4.4.2, except that attached hydraulic and pneumatic lines are not required. Electrical and electronic units mounted on shock or vibration isolators will normally be tested hard mounted to assure that the minimum spectrum shown in Figure 5 is input to the test item.

7.4.4.3 Test Level and Duration. The vibration environment will be as defined in 7.1.3. The minimum spectrum is shown in Figure 5. Where insufficient time is available to test all modes of operation, extended testing at a level no lower than 6 dB below the acceptance test level will be conducted as necessary to complete functional testing.

7.4.4.4 Supplementary Requirements. Same as 6.4.4.4 and if the dynamic test configuration (unit and fixture) changes from the qualification configuration, then the fixture evaluation (6.4.4.5) will be repeated before testing to acceptance levels.

7.4.4.5 Special Considerations for Isolators. All isolators will be lot tested in at least one axis, with rated supported mass, to verify that dynamic amplification and resonant frequency are within allowable limits. Test inputs may either be the maximum expected random vibration level applied for at least 15 seconds, or be a reference sinusoidal input having a frequency sweep rate not greater than 1 octave per minute.

7.4.5 Acoustic Test, Unit Acceptance

7.4.5.1 Purpose. The acoustic test detects material and workmanship defects by subjecting the unit to an acoustic environment.

7.4.5.2 Test Description. Same as 6.4.5.2.

7.4.5.3 Test Level and Duration. The unit acoustic environment will be as defined in 7.1.2. Where insufficient time is available during the 1-minute to check redundant circuits, functions, and modes that are operating during the launch, ascent, or reentry phase, extended testing at a level no lower than 6 dB below the acceptance level will be conducted as necessary to complete functional testing.

7.4.5.4 Supplementary Requirements. Same as 6.2.4.4.

7.4.6 Shock Test, Unit Acceptance

7.4.6.1 Purpose. The shock test is intended to reveal material and workmanship defects in units subject to high-level shock environments in flight.

7.4.6.2 Test Description. The unit will be attached at its normal points to the same fixture or structure used for its shock qualification test (6.4.6.2). The unit will be electrically energized and monitored. The test technique employed will be identical to that selected for its qualification, differing only in level and the number of repetitions. A functional test of the unit will be performed before and after the shock test. The unit will be electrically energized during the testing. Circuits should be monitored for intermittents to the maximum extent practicable.

7.4.6.3 Test Level and Exposure. The shock response spectrum in both directions of each of 3 orthogonal axes will be at least the maximum expected level for that direction. A sufficient number of shocks will be imposed to meet the required level in each of these 6 directions at least once.

7.4.6.4 Supplementary Requirements. A shock acceptance test becomes a required test (3.5.5) if the maximum expected shock response spectrum in g's exceeds 1.6 times the frequency in Hz (corresponding to a velocity of 2.54 meters/second or 100 inches/second). For example, if the maximum expected shock response spectrum value at 2000 Hz exceeds 3200g, the test is required.

7.4.7 Proof Load Test, Structural Unit Acceptance

7.4.7.1 Purpose. The proof load test will be conducted for all structural units made from composite material or having adhesively bonded parts. The proof load test detects material, process, and workmanship defects that would respond to structural proof loading. The requirement for the proof load test is waived if a proven nondestructive evaluation method, with well established accept and reject criteria, is used instead.

7.4.7.2 Test Descriptions. Same as 7.3.1.2.

7.4.7.3 Test Level and Duration. Same as 7.3.1.3.

7.4.8 Proof Pressure Test, Unit Acceptance

7.4.8.1 Purpose. The proof pressure test detects material and workmanship defects that could result in failure of the pressure vessel or other units in usage.

7.4.8.2 Test Description. Same as described in 6.4.8.2a.

7.4.8.3 Test Level and Duration. Same as 6.4.8.3a and b.

7.4.8.4 Supplementary Requirements. MIL-STD-1522 and applicable safety standards will be followed.

7.4.9 Leakage Test, Unit Acceptance

7.4.9.1 Purpose. The leakage test demonstrates the capability of units to meet the specified leakage requirements.

7.4.9.2 Test Description. The unit leak checks will be made using the same method as used for qualification.

7.4.9.3 Test Level and Duration. Same as 6.4.7.3.

7.4.10 Wear-in Test, Unit Acceptance

7.4.10.1 Purpose. The wear-in test detects material and workmanship defects that occur early in the unit life, and to wear-in or run-in of mechanical units so that they perform in a smooth, consistent, and controlled manner.

7.4.10.2 Test Description. While the unit is operating under conditions representative of operational loads, speed, and environments and while perceptive parameters are being monitored, the unit will be operated for the specified time period. 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 will be conducted at ambient temperature. For thrusters, a cycle is a hot firing that includes a start, steady-state operation, and shutdown. For hot firings of thrusters utilizing hydrazine propellants, action will 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 wear-in test requirements.

7.4.10.3 Test Levels and Duration.

a. Pressure. Ambient pressure should normally be used.

b. Temperature. Ambient temperature will be used for operations if the test objectives can be met. Otherwise, temperatures representative of the operational environment will be used.

c. Duration. The number of cycles will be either 15 or 5% of the total number of expected cycles during service life (3.5.7), whichever is greater.

7.4.10.4 Supplementary Requirements. Perceptive parameters will be monitored during the wear-in test to detect evidence of degradation.

7.4.11 EMC Test, Unit Acceptance. Limited EMC acceptance testing will be accomplished on units that exhibit emission or susceptibility characteristics, which may adversely affect vehicle performance, to verify that these characteristics have not deteriorated from the qualification test levels. The tests should be restricted to only those necessary to evaluate these critical characteristics.

SECTION 8.

ALTERNATIVE STRATEGIES

The qualification testing in Section 6 provides a demonstration that the design, manufacturing, and acceptance testing produces flight items that meet specification requirements. In a minimum-risk program, the hardware items subjected to qualification tests are themselves not eligible for flight, since there has been no demonstration of remaining life from fatigue and wear standpoints. Yet, programmatic realities of limited production, tight schedules, and budgetary limits do not always provide for dedicated nonflight qualification items. In response, strategies have evolved to minimize the risk engendered by this situation. The three strategies or combinations thereof, described in this section, may be used at the vehicle, subsystem, and unit levels. It should be recognized that these strategies present a higher risk than the use of standard acceptance tested items for flight that have margins demonstrated by testing of a dedicated qualification item. The higher risk of these alternate strategies may be partially mitigated by enhanced development testing and by increasing the design factors of safety.

The strategies are intended for use in space vehicle programs that have a very limited number of vehicles.

8.1 SPARES STRATEGY

This strategy does not alter the qualification and acceptance test requirements presented in Sections 6 and 7. Yet, in some cases, qualification hardware may be used for flight if the risk is minimized. In a typical case, the qualification test program results in a qualification test vehicle that was built using units that had been qualification tested at the unit level. After completing the qualification tests, the critical units can be removed from the vehicle and the qualification vehicle can then be refurbished, as necessary. Usually a new set of critical units would be installed that had only been acceptance tested. This refurbished qualification vehicle would then be certified for flight when it satisfactorily completes the vehicle acceptance tests in 7.2. In vehicles where redundant units are provided, only one of the redundant units would have been qualification tested at the unit level, so only it would be removed and replaced. The qualification units that were removed would be refurbished, as necessary, and would typically be used as flight spares. However, qualification units that are mission or safety critical (3.2.2) should never be used for flight.

8.2 FLIGHTPROOF STRATEGY

With a flightproof strategy, all flight items are subjected to enhanced acceptance testing, and there is no qualification item. The risk taken is that there has been no formal demonstration of remaining life for the flight items. This risk is alleviated to some degree by the fact that each flight item has met requirements under acceptance testing at higher than normal levels. The test levels are mostly less than those specified in Section 6 for qualification, but are never less than those specified in Section 7 for acceptance. The test durations for the flightproof test strategy are the same as those specified for acceptance. It is recommended that development testing be used to gain confidence that adequate margin, especially in a fatigue or wear sense, remains after the maximum allowed accumulated acceptance testing at the enhanced levels.

8.2.1 Vehicle Flightproof Tests. The vehicle flightproof tests will be conducted as in 7.2 (Table XII), with the following modifications:

a. The vehicle shock test will be conducted as in 6.2.3 for the first flight vehicle. For subsequent vehicles, only 1 activation of significant events is required (7.2.3).

b. The vehicle acoustic and random vibration tests will be conducted as in 7.2.4 and 7.2.5, except that the test level will be 3 dB above the acceptance test environment (7.1.2 and 7.1.3). For the first flight vehicle, the tests will be conducted with power on, to the extent practicable.

c. The vehicle thermal vacuum tests will be conducted as in 7.2.8, except that the hot and cold temperatures will be 5°C beyond the acceptance temperatures for units (7.1.1).

d. The vehicle thermal balance test will be conducted on the first flight vehicle as in 6.2.8.

e. If a thermal cycle test is conducted as in 7.2.7, then the minimum vehicle temperature range will be 60°C.

f. EMC tests will be conducted as in 6.2.2 for the first flight vehicle. For subsequent vehicles, the EMC test of 7.2.2 will be required.

g. The modal survey will be conducted as in 6.2.10 on the first flight vehicle.

8.2.2 Subsystem Flightproof Tests. The subsystem flightproof tests will be conducted as in 7.3. In addition, a proof load test will be conducted on all structures in the structural subsystem. The proof load will be equal to 1.1 times the limit load.

8.2.3 Unit Flightproof Tests. The unit flightproof tests will be conducted as in 7.4 (Table XIII), with the following modifications:

a. For the first flight unit only, the shock test will be conducted as in 6.4.6, except that the shock level will be 3 dB above the acceptance test level, achieved once in both directions of 3 axes. For subsequent units, the shock test will be conducted if required as described in 7.4.6, except that the shock test level will be 3 dB above the acceptance test level.

b. Vibration and acoustic tests will be conducted as in 7.4.4 and 7.4.5, except that the test level will be 3 dB greater than the acceptance test level (7.1.2 and 7.1.3).

c. The unit thermal vacuum tests will be conducted as in 7.4.3, except that the hot and cold temperatures will be 5°C beyond the acceptance test temperatures (7.1.1). For the first flight antenna and solar array units, this thermal vacuum test will be required.

d. The unit thermal cycle tests will be conducted as in 7.4.2, except that the hot and cold temperatures will be 5°C beyond the acceptance test temperatures (7.1.1).

e. The unit EMC test will be conducted on the first unit as in 6.4.11.

The unit flightproof test approach will not be allowed for pressure vessels, pressure components, structural components with a low fatigue margin, and nonrechargeable batteries. These units will follow a normal qualification and acceptance program as specified in Sections 6 and 7.

8.3 PROTOQUALIFICATION STRATEGY

With a protoqualification strategy, a modified qualification (protoqualification) is conducted on a single item and that test item is considered to be available for flight. The normal acceptance program in Section 7 is then conducted on all other flight items.

8.3.1 Vehicle Protoqualification Tests. The protoqualification tests will be conducted as in 6.2 (Table VIII), with the following modifications:

a. The shock test will be conducted as in 6.2.3, except that only 2 repetitions of activated events are required.

b. The acoustic or random vibration tests will be conducted as in 6.2.4 and 6.2.5, except that the duration factors will be 2 (instead of 4) and the level margin for the flight environment will be 3 dB (instead of 6 dB typically) in place of the requirements in 6.1.4. If the test is accelerated (6.1.4.2), the time reduction factor will be based on the reduced level margin per Table VII.

c. The thermal vacuum test will be conducted as in 6.2.9, except that the hot and cold temperatures will be 5°C beyond the acceptance temperatures for units (7.1.1) and the number of cycles will be half of those in Table VI.

d. If the alternate thermal cycle test is conducted as in 6.2.7, then the minimum vehicle temperature range will be 60°C and the number of cycles will be half of those in Table VI.

8.3.2 Subsystem Protoqualification Tests. The subsystem protoqualification tests will be conducted as in 8.3.1, except that the structural subsystem tests will be conducted as in 6.3 (Table IX) with an ultimate load test factor of 1.25. No detrimental deformation will be allowed during the test. In addition, the design safety factor for ultimate will be 1.4 and the design safety factor for yield will be 1.25.

8.3.3 Unit Protoqualification Tests. The protoqualification unit tests will be conducted as in 6.4 (Table X), with the following modifications:

a. The shock test will be conducted as in 6.4.6, except that only 2 repetitions and only a 3 dB level margin for the flight environment (instead of 6 dB typically, Table IV) will be required.

b. The random vibration or acoustic tests will be conducted as in 6.4.4 and 6.4.5, except that the duration factors will be 2 (instead of 4) and the level margin for the flight environment will be 3 dB (instead of 6 dB typically). If the test is accelerated (6.1.4.2), the time reduction factor will be based on the reduced level margin per Table VII.

c. The thermal vacuum tests will be conducted as in 6.4.3, except that the hot and cold temperatures will be 5°C beyond the acceptance temperatures for units (7.1.1) and the number of cycles will be half of those in Table VI.

d. The thermal cycle tests will be conducted as in 6.4.2, except that the hot and cold temperatures will be 5°C beyond the acceptance temperatures for units (7.1.1) and the number of cycles will be half of those in Table VI.

8.4 COMBINATION TEST STRATEGIES

Various combinations of strategy may be considered depending on specific program considerations and the degree of risk deemed acceptable. For example, the protoqualification strategy for units (8.3.3) may be combined with the flightproof strategy for the vehicle (8.2.1). In other cases, the flightproof strategy would be applied to some units (8.2.3) peculiar to a single mission, while the protoqualification strategy may be applied to multi-mission units (8.3.3). In such cases, the provisions of each method would apply and the resultant risk would be increased correspondingly.

SECTION 9.

PRELAUNCH VALIDATION AND OPERATIONAL TESTS

9.1 PRELAUNCH VALIDATION TESTS, GENERAL REQUIREMENTS

Prelaunch validation testing is accomplished at the factory and at the launch base, with the objective of demonstrating launch system and on-orbit system readiness. Prelaunch validation testing is usually divided into two phases:

Phase a. Integrated system tests (Step 3 tests).
Phase b. Initial operational tests and evaluations (Step 4 tests).

During Phase a, the test series establishes the vehicle baseline data in the factory preshipment acceptance tests. All factory test acceptance data should accompany delivered flight hardware. When the launch vehicle(s), upper-stage vehicle(s), and space vehicle(s) are first delivered to the launch site, tests will be conducted as required to assure vehicle readiness for integration with the other vehicles. These tests also verify that no changes have occurred in vehicle parameters as a result of handling and transportation to the launch base. The launch vehicle(s), upper-stage vehicle(s), and space vehicle(s) may each be delivered as a complete vehicle or they may be delivered as separate stages and first assembled at the launch site as a complete launch system. 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 that ship a complete vehicle to the launch site, these tests primarily confirm vehicle performance, check for transportation damage, and demonstrate interface compatibility.

During Phase b, initial operational tests and evaluations (Step 4 tests) are conducted following the integrated system tests to demonstrate successful integration of the vehicles with the launch facility, and that compatibility exists between the vehicle hardware, ground equipment, computer software, and within the entire launch system and on-orbit system. The point at which the integrated system tests end and the initial operational tests and evaluations begin is somewhat arbitrary since the tests may be scheduled to overlap in time. To the greatest extent practicable, the initial operational tests and evaluations are to exercise all vehicles and subsystems through every operational mode in order to ensure that all mission requirements are satisfied. These Step 4 tests will be conducted in an operational environment, with the equipment in its operational configuration, by the operating personnel in order to test and evaluate the effectiveness and suitability of the hardware and software. These tests should emphasize reliability, contingency plans, maintainability, supportability, and logistics. These tests should assure compatibility with scheduled range operations including range instrumentation.

9.2 PRELAUNCH VALIDATION TEST FLOW

Step 4 testing of new or modified ground facilities, ground equipment, or software should be completed prior to starting the prelaunch validation testing of the vehicles at the launch base. The prelaunch validation test flow will follow a progressive growth pattern to ensure proper operation of each 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 vehicles or subsystems are verified, assembly proceeds to the next level of assembly. Following testing of the vehicles and their interfaces, the vehicles are electrically and mechanically mated and integrated into the launch system. Upper-stage vehicles and space vehicles employing a recoverable flight vehicle will utilize a flight vehicle simulator to perform mechanical and electrical interface tests prior to integration with the flight vehicle. Following integration of the launch vehicle(s), upper-stage vehicle(s), and space vehicle(s), functional tests of each of the vehicles will be conducted to ensure its proper operation following the handling operations involved in mating. Vehicle cleanliness will be monitored by use of witness plates. In general, the Step 4 testing of the launch system is conducted first, then the Step 4 testing of the on-orbit space system is conducted.

9.3 PRELAUNCH VALIDATION TEST CONFIGURATION

During each test, the applicable vehicle(s) should be in their flight configuration to the maximum extent practicable, consistent with safety, control, and monitoring requirements. For programs utilizing a recoverable flight vehicle, the test configuration will include any airborne support equipment required for the launch, ascent, and space vehicle deployment phases. This equipment will be mechanically and electrically mated to the space vehicle in its launch configuration. Whenever practicable, ground support equipment should have a floating-point-ground scheme that is connected to the flight vehicle single-point ground. Isolation resistance tests will 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 will 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 will 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 will exercise and demonstrate satisfactory operation of each of the vehicles through all of their mission phases, to the maximum extent practicable. Test data will be compared to corresponding data obtained in factory tests to identify trends in performance parameters. Each test procedure used should include test limits and success criteria sufficient to permit a rapid determination as to whether or not processing and integration of the launch system 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 Tests. Electrical functional tests will be conducted that duplicate, as nearly as practicable, the factory functional tests performed for vehicle acceptance. Mechanical tests for leakage, valve and mechanism operability, and fairing clearance will be conducted.

9.4.1.1 Simulators. Simulation devices will be carefully controlled and will 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, the interfaces disconnected in the subsequent replacement of the simulators with flight hardware will be revalidated. Simulators will be used for the validation of ground support equipment prior to connecting it to flight hardware.

9.4.1.2 Explosive-ordnance Firing Circuits. If not performed at an earlier point in the factory test cycle, validation that proper ignition energy levels are present at each electro-explosive device (EED) will be performed prior to final connection of the firing circuit to the EEDs. A simulation of the EED characteristics will be used during these tests. The circuits will be commanded through power-on, arm, and fire cycles. The circuits are to be monitored during the tests to detect energy densities exceeding ignition threshold during power-on and arm cycles, and to validate that proper ignition energy density is transmitted to the conducting pins of the EED at the fire command. Circuit continuity and stray energy checks will be made prior to connection of a firing circuit to ordnance devices and this check will be repeated whenever that connection is opened and prior to reconnection.

9.4.1.3 Transportation and Handling Monitoring. Monitoring for shock and vibration should be performed at a minimum of the forward and aft interfaces between the shipping container transporter and the article being shipped, and on the top of the article. Measurements should be on the article side of the interface in all three axes at each location. The monitoring requires a sensing and recording subsystem capable of providing complete time histories of the most severe events, as well as condensed summaries of the events, including their time of occurrence. A frequency response up to 300 Hz is required. Monitoring should cover the entire shipment period and the data evaluated as part of the receiving process. Exposure to shock or vibration having a spectrum above the acceptance spectrum may require additional testing or analysis.

9.4.2 Propulsion Subsystem Leakage and Functional Tests. Functional tests of the vehicle propulsion subsystem(s) will be conducted to verify the proper operation of all units, to the maximum extent practicable. Propulsion subsystem leakage rates will be verified to be within allowable limits.

9.4.3 Launch-critical Ground Support Equipment Tests. Hardware associated with ground subsystems that are flight critical and nonredundant (such as umbilicals) will have been subjected to appropriate functional tests under simulated functional and environmental conditions of launch. These tests will include an evaluation of radio-frequency (rf) interference between system elements, electrical power interfaces, and the command and control subsystems. On a new vehicle design or a significant design change to the telemetry, tracking, or receiving subsystem of an existing vehicle, a test will be run on the first vehicle to ensure nominal operation and that explosive-ordnance devices do not fire when the vehicle is subjected to the worst-case electromagnetic interference environment.

9.4.4 Compatibility Test, On-orbit System.

9.4.4.1 Purpose. The compatibility test validates the compatibility of the upper-stage vehicle, the space vehicle, the on-orbit command and control network, and other elements of the space system. For the purpose of establishing the compatibility testing baseline, it is assumed that the on-orbit command and control network is (or operationally interfaces with) the Air Force Satellite Control Network (AFSCN). The compatibility test demonstrates the ability of the upper-stage vehicle and space vehicle, when in orbit, to properly respond to the AFSCN hardware, software, and operations team as specified in the AFSCN Program Support Plan. For programs that have a dedicated ground station, compatibility tests will 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 Range (WR) and the Eastern Range (ER). At both locations, there are facilities that can command the launch, upper-stage, and space vehicles, process telemetry from the vehicles, as well as perform tracking and ranging, thus verifying the system compatibility, the command software, the telemetry processing software, and the telemetry modes. The required tests include the following:

a. Verification of the compatibility of the radio frequencies and signal waveforms used by the flight unit's command, telemetry, and tracking links.

b. Verification of the ability of the flight units to accept commands from the command and control network(s).

c. Verification of the command and control network(s) capability to receive, process, display, and record the vehicle(s) telemetry link(s) required to monitor the flight units during launch, ascent, and on-orbit mission phases.

d. Verification of the ability of the flight units to support on-orbit tracking as required for launch, ascent, and on-orbit mission phases.

9.4.4.3 Supplementary Requirements. The compatibility test should be run as soon as feasible after the vehicles arrives at the launch base. The test is made with every vehicle to verify system interface compatibility. The test will be run using the software model versions that are integrated into the operational on-orbit software of the vehicle under test. A preliminary compatibility test may be run prior to the arrival of the vehicle at the launch base by the use of prototype subsystems, units, 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 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 will be checked by the compatibility tests conducted just prior to launch. Following the completion of the compatibility test, the on-orbit command and control network configuration of software, hardware, and procedures should be frozen until the space vehicle is in orbit and initialized.

9.5 FOLLOW-ON OPERATIONAL TESTS

9.5.1 Follow-on Operational Tests and Evaluations. Follow-on Operational Tests and Evaluations will be conducted at the launch site in an operational environment, with the equipment in its operational configuration. The assigned operating personnel will identify operational system deficiencies.

9.5.2 On-orbit Testing. On-orbit testing should be conducted to verify the functional integrity of the space vehicle following launch and orbital maneuvering. Other on-orbit testing requirements are an important consideration in the design of any space vehicle. For example, there may be a need to calibrate on-line equipment or to verify the operational status of off-line equipment while in orbit. However, on-orbit testing is dependent on the built-in design features, and if testing provisions were not provided, the desired tests cannot be accomplished. On-orbit tests are, therefore, so program peculiar that specific requirements are not addressed in this Handbook.

9.5.3 Tests of Reusable Flight Hardware. Tests of reusable flight hardware will be conducted as required to achieve a successful space mission. Reusable hardware consists of the vehicles and units intended for repeated missions. Airborne support equipment, that performs its mission while attached to a recoverable launch vehicle, is an example of a candidate for reuse. The reusable equipment would be subjected to repeated exposure to test, launch, flight, and recovery environments throughout its service life. The accumulated exposure time of equipment retained in a recoverable vehicle and of airborne support equipment is a function of the planned number of missions involving this equipment and the retest requirements between missions. The environmental exposure time of airborne support equipment is further dependent on whether or not its use is required during the acceptance testing of other nonrecoverable flight equipment. 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 Handbook. Similarly, orbiting space vehicles that have completed their useful life spans may be retrieved by means of a recoverable flight vehicle, refurbished, and reused. 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.

CONCLUDING MATERIAL

Custodians:
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Preparing Activity:
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STANDARDIZATION DOCUMENT IMPROVEMENT PROPOSAL

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3. DOCUMENT TITLE **TEST REQUIREMENTS FOR LAUNCH, UPPER-STAGE, AND SPACE VEHICLES, VOL I : BASELINES**

4. NATURE OF CHANGE *(Identify paragraph number and include proposed rewrite, if possible. Attach extra sheets as needed.)*

5. REASON FOR RECOMMENDATION

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MIL-HDBK-340A (USAF)
01 April 1999

SUPERSEDING
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01 July 1985

DEPARTMENT OF DEFENSE
HANDBOOK

TEST REQUIREMENTS
FOR
LAUNCH, UPPER-STAGE, AND SPACE VEHICLES
Vol II : Applications Guidelines

This handbook is for guidance only.

Do not cite this document as a requirement

AMSC N/A FSC 1810

FOREWORD

1. This Military Handbook is approved for use by all Departments and Agencies of the Department of Defense.

2. This Vol. II of MIL-HDBK-340A is intended to document additional facets of engineering technical information pertinent to the requirements stated in Vol. I; “Test Requirements for Launch, Upper Stage and Space Vehicles.” As a technical reference, this handbook provides the basis for achieving a consistent technical approach for tailoring test requirements, where appropriate, and may also provide the basis to develop alternative approaches where they are appropriate. The information included herein is for general guidance; it need not be followed if it does not accommodate the requirements of the program.

3. The handbook has the same organization as a military standard, i.e., the first three sections are: Section 1, Scope, Section 2 Referenced Documents, and Section 3, Definitions. Section 4 through 7 provide technical information and guidance material for topics contained in Vol I. The section, subsection, and paragraph numbers of this handbook do not correspond to the paragraph numbers of Vol I. However, exact references are given in this Vol. II of the handbook to the corresponding paragraph numbers of Vol I.

4. Each major subsection of this handbook addresses a subject area of interest. Each subject area is organized into three major paragraphs. The first paragraph is titled “Standard Criteria,” and it references the paragraph of Vol I where the topics are discussed. The second major paragraph is titled “Rationale for ..., and it contains background information such as the purpose or reasons for the subject area requirements in Vol I. The third major paragraphs is titled “Guidance for Use of...,” and it contains information intended to aid the reader in the detailed application of the requirements in Vol I for that subject area

5. Beneficial comments (recommendations, additions, or deletions) and any pertinent data which may be of use in improving this document should be provided by letter addressed to: Space and Missile Systems Center, SMC/AXMP, 160 Skynet Street, Suite 2315, Los Angeles AFB, El Segundo, CA 90245-4683 by using the Standardization Document Improvement Proposal (DD Form 1426) appearing at the end of this document or by letter.

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SECTION 1.

1. SCOPE

This Volume II of the handbook provides additional information pertaining to the test requirements of the handbook, "test requirements for launch, upper stage and space vehicles”.

1.1 PURPOSE

This Vol II of the Handbook was written to provide explanations and guidance to the users of VOL I of the Handbook. The information presented herein is intended to aid in the formulation and review of detailed test requirements for launch, upper stage and space vehicles including the tailoring of VOL I of the Handbook requirements for specific program specifications or contracts.

1.2 ORGANIZATION OF HANDBOOK

The organization of this handbook differs from that in VOL I of the Handbook where test requirements for specific levels of hardware assembly are individually discussed for each test category such as acoustics, vibration, thermal vacuum and thermal cycling. The organization of Vol II is structured to allow for a discussion of each test category covering the same type of test at all levels of hardware assembly. For example, the discussion of thermal vacuum testing contains an integrated discussion of thermal vacuum testing and the considerations when these tests are performed at the unit, subsystem and system levels of assembly. The same is true for all other test disciplines addressed in VOL I of the Handbook such as vibration, acoustics, shock, static loads, EMC, pressure, and acceleration.

2.0 APPLICABLE DOCUMENTS

2.1 General. The documents below are not necessarily all of the referenced herein, but are the ones that are needed in order to fully understand the information provided by this handbook.

2.2 Government Documents

2.2.1 SPECIFICATIONS, STANDARDS, AND HANDBOOKS. The following standards and specifications form a part of this document to the extent specified herein. Unless otherwise specified, the issues of these documents are those listed in the Department of Defense Index of Specifications and Standards (DoDISS) and supplement thereto, cited in the solictation. When this handbook is used by acquisition, the application issue of the DoDISS must be cited in the soliciation.

Military Standards

MIL-STD-810	Environmental Test Methods and Engineering Guidelines
MIL-STD-1522 (USAF)	Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems
MIL-STD-1540D	Product Verification Requirements for Launch, Upper Stage and Space Vehicles

Handbooks

MIL-HDBK-340A Vol I	Test Requirements for Launch, Upper Stage, and Space Vehicles : Baselines
MIL-HDBK-343	Design, Construction, and Testing Requirements for One-of-a-Kind Space Equipment
MIL-HDBK-83575	General Handbook for Space Vehicle Wiring Harness Design and Testing
MIL-HDBK-83578	Criteria for Explosives Systems and Devices Used on Space Vehicles

(Unless otherwise indicated, copies of federal and military specifications, standards and handbooks are available from the Standardization Document Order Desk, 700 Robbins Avenue, Building 4D, Philadelphia, PA 19111-5094.)

2.3 TECHNICAL REFERENCES

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2.4 **Order of Precedence** In the event of a conflict between the text of this document and the references sited herein, the text of this document takes precedence. Nothing in this document, however, supercedes applicable laws and regulations unless a specific exemption has been obtained.

SECTION 3.

DEFINITIONS

The definitions of terms used in this handbook are the same as in VOL I of the Handbook.

SECTION 4.

APPLICATION OF TEST REQUIREMENTS AND TAILORING

4.1 STANDARD CRITERIA

Paragraphs 1., 4.2, and 8, of the Handbook VOL I provide information to be considered in tailoring and establishing general test requirements for a given program.

4.2 RATIONALE FOR TAILORING

For most military systems, testing and maintenance costs represent major elements of the life cycle cost. Unlike aircraft programs where testing and maintenance costs are primarily incurred during operational use, the testing costs for launch vehicles, upper stages and spacecraft are primarily incurred prior to operational deployment since maintenance after launch is seldom possible. Because testing represents such a large expense, good management requires tailoring of the test program to assure that a cost effective program is achieved. On one hand, any excessive testing clearly represents a waste of money and time. On the other hand, an undetected deficiency or failure can result in an unsuccessful launch or shortened orbital life. Because a single failure can result in a loss of several 100 million dollars (not including the loss of scientific or operational data) a considerable budget for quality control, and for testing that will ensure success, is usually cost-effective. Some space vehicles have achieved success even with limited testing and quality assurance provisions. Conversely, programs can be found where extensive inspections and tests at every step of the acquisition process still resulted in unsuccessful missions. However, the preponderance of evidence is, as expected, that the use of extensive testing and other quality assurance provisions that are based upon those used for previously successful programs is the most cost-effective approach. For high reliability space programs, testing costs may represent as much as 35 percent of the cost of each vehicle.

VOL I of the Handbook establishes uniform definitions and a baseline set of testing requirements for launch, upper stage and space vehicles. The baseline test requirements are a composite of those tests currently used in achieving successful space missions for high priority programs. The baseline requirements consist of those acceptance and full qualification tests deemed to be required for low risk space programs. It is intended that these baseline requirements will be tailored to fit each program's requirements while recognizing the desire to meet the minimum requirements of VOL I of the Handbook where practicable. VOL I of the Handbook contains provisions for self tailoring of many individual test requirements. VOL I of the Handbook provides a common framework from which program managers can identify and evaluate deviations in their testing and quality assurance plans. The extent of acceptable deviations is a tradeoff among program requirements, acceptable risk, and testing costs including schedule delays. Because the cost-effectiveness of these tradeoffs is difficult to evaluate statistically due to the small sample size of each program, an evaluation of the deviations from VOL I of the Handbook baseline requirements should be included in all program reviews.

4.3 GUIDANCE FOR TEST PROGRAM DESIGN

4.3.1 Tradeoffs Between Unit, Subsystem and Vehicle Testing. The standard test baseline for a low risk program includes acceptance and qualification testing at the unit, the subsystem, and again at the vehicle level of assembly. Higher risk programs, for example, a class D program as defined by MIL-HDBK-343, may defer some or all of the unit or subsystem testing to the vehicle level of assembly. Deferring testing to higher levels of assembly usually offers an improvement in test simulation but can add substantial economic and performance risk. For example, a unit with a manufacturing defect or marginal design, untested at a unit level of assembly, may fail during testing at a higher level of assembly, and delay the whole program while the individual unit problem is being corrected. In this example, the resulting cost can be more than an order of magnitude greater than if the problem had been detected by testing at the unit level of assembly. This situation is illustrated in Figure 1 which shows relative costs of recovering from failures at various levels of assembly. On the other hand, most vehicles will contain types of hardware where particular environmental tests at lower levels of assembly are not meaningful or cost effective. Examples include propulsion tubing and wiring harnesses where most environmental tests are effectively performed at higher levels of assembly. In general the lowest overall economic and performance risks are achieved by judicious testing at all levels of assembly.

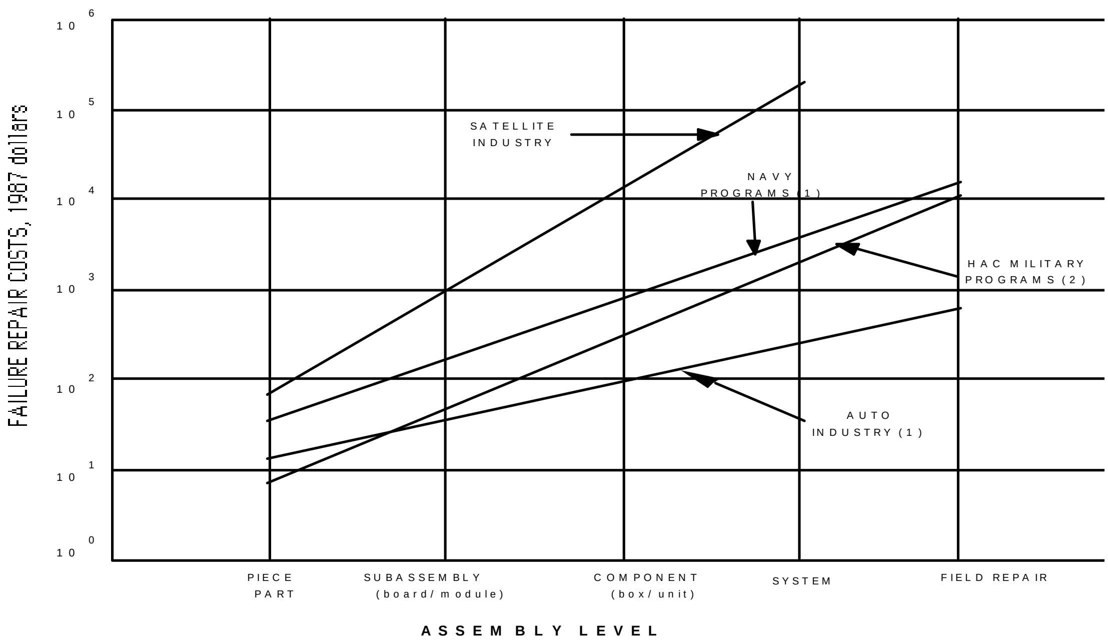

Figure 1 Hardware Failure Costs vs. Assembly Level

1. W. J. Willoughby, IES National Conference, 1979
2. A. Searl, RADC Report TR-82-87, 1982

4.3.2 Development vs. Qualification and Acceptance Testing. Development tests, in general, cannot substitute for qualification tests. However, there can be exceptions in cases when the development tested article meets the same configuration and test documentation requirements applicable to the qualification article. Development tests should not be substituted for acceptance tests of flight articles.

4.3.3 Baseline Qualification and Acceptance Testing. The baseline test program consists of all of the required qualification and acceptance tests defined respectively in Sections 6 and 7 of VOL I of the Handbook. For programs that desire low risk or involve anticipated moderate to large production quantities (ten or more for a space program) the baseline test program will likely be the most advantageous based on both economic and technical risk considerations. There are several major advantages of using the baseline test approach. These advantages are sometimes overlooked when minimum cost test programs are being established. For example, performing qualification tests on non-flight hardware allows for a more thorough evaluation of design capability without the concern of causing damage or inducing latent fatigue in flight hardware. Experience has shown that a thorough qualification test program will reduce the number of problems and discrepancies occurring during subsequent acceptance testing of flight hardware as well as reducing the risk of problems occurring in flight. In addition, retention of the qualification hardware for testing of design modifications and trouble shooting can provide significant program benefits over the long term.

For programs involving smaller production quantities (less than ten vehicles), one or a combination of the alternative strategies discussed in paragraph 4.3.4 of Vol II may provide an economic advantage with an acceptable increase in risk. Generally, launch vehicle programs will not apply these alternative strategies because there is less design redundancy and production quantities are usually sufficiently large such that the risks incurred outweigh the economic advantages.

4.3.3.1 Unit Test Baseline

4.3.3.1.1 Standard Criteria for Unit Test Baseline. The contents of paragraphs 5.4, 6.4 and 7.4 of VOL I of the Handbook provide the baseline requirements for development, qualification and acceptance testing of units. The baseline requirements are summarized in Tables X and XIII of VOL I of the Handbook.

4.3.3.1.2 Rationale for Unit Test Baseline. Qualification tests are a formal demonstration that a production unit (or prototype) is adequate to successfully withstand specified tests. These tests are mainly performed to determine if there are factors that may have been overlooked during design, analysis, or manufacturing. Additionally, the environments used during these tests are more severe than those expected to occur during flight in order to account for factors such as variabilities among production articles, uncertainties associated with testing, the effects of combined environments, and the inability to fully replicate boundary conditions for dynamic and thermal tests. Qualification test requirements, therefore, incorporate margins which are added to the range of environmental levels and stresses expected to occur in flight and during environmental stress screening tests used for acceptance. The maximum expected and extremes of the flight environments are defined in paragraphs 3.3 and 3.4 of VOL I of the Handbook. The environmental test margins specified are intended to incorporate the allowable test condition tolerances. The environmental test margins assure qualification test levels that are more severe than the maximum conditions that can occur in flight and help assure against performance degradation and fatigue failures due to repeated acceptance testing and operational use. For example, the 10 deg C environmental margins specified in VOL I of the Handbook result in the baseline thermal design range for units from -54 deg C to +71 deg C. This baseline design range for units is similar to that used for aircraft subsystems and therefore should not impose unusual design problems in most cases. In addition, this baseline design range encourages the development of standard hardware, provides a very revealing test screen for defective units, allows units to be moved to other locations or changes in orientation on a vehicle without affecting qualification, and may allow the use of a qualified unit on other vehicles without requalification.

Before qualification testing, the units to be qualified should have been subjected to the same controls, inspections, and alignments, imposed on flight units. VOL I of the Handbook however does not require that acceptance tests be performed on the qualification units.

Environmental acceptance tests are conducted on units to demonstrate flightworthiness and to disclose quality deficiencies in the flight article. Acceptance tests are intended to satisfy these goals by subjecting the unit to the maximum environmental exposures expected in service or the environmental stress screen level whichever is more severe. The test program is comprised of a series of required tests, augmented by additional tests on a case-by-case basis depending upon the application and sensitivity of the unit to the test environment.

The suggested test sequences require functional tests before and after each environmental test. Additionally, certain functional tests are required to be performed during some of the environmental tests. The recommended sequencing is based on rationale consisting of a combination of the order in which the environments are encountered during flight and the desire to perceive defects as early in the test sequence as possible (see Subsection 7.2). The categorization of tests into "required" and "other" was guided by the sensitivity of the type of unit to the specific environment and by the probability of encountering the environment. As an example, leak tests are required only on sealed or pressurized equipment, since such equipment may be sensitive to loss of pressure, vacuum, or a purge mechanism.

4.3.3.1.3 **<u>Guidance for Use of Unit Test Baseline</u>.** The sequencing and categorization of the tests should be tailored to each specific unit on a program. This tailoring should consider both increasing and decreasing the severity of the tests. For example, while random vibration tests for electronic units are normally much more revealing than acceleration tests, some electronic units may require both types of tests.

The climatic qualification tests are designated as "other" for all units; however, if units are not fully environmentally protected on the ground, such tests should become required. This is also the case for such tests as explosion-proofing, and radiation which are not specified in VOL I of the Handbook, but each should become mandatory when such requirements exist in operational situations.

The mechanical and electrical functional tests are extremely important elements in the test baselines. The functional tests are conducted prior to and after each of the environmental tests. They should be designed to verify that performance of the units meets the specification requirements, that the units are compatible with ground support equipment, and that all software used is validated. The electrical functional tests should involve applying electrical inputs to interfaces including redundant circuits and the verification of unit performance. The mechanical functional tests should apply mechanical inputs including torques, loads, and motions, and should measure performance. The electrical and mechanical inputs should be varied through their specification ranges to verify the unit performance throughout the range. In addition, the electrical functional tests should include negative logic testing to verify lockout, to assure that no function other than the intended function was performed, and to verify that the signal was not present other than when programmed. To the extent practicable, the functional tests should also be designed so that a database of critical parameters can be established for trend analysis. This is accomplished by measuring the same critical parameters in all of the functional tests conducted before, during, and after each of the baseline environmental tests.

Functional tests provide the criteria for judging successful survival of the unit in a given test environment. It is also important to perform functional tests of the unit while the environment is being imposed. Many defects, which otherwise escape detection by pre- and post-test functional checks, reveal themselves during environmental tests. For example, intermittents may be caused by foreign bodies, contaminants, inadequate clearances, cracks, debonds, and damaged connectors that often are only revealed during environmental tests. Therefore, regardless of the functional mode of the unit during launch and ascent, the unit should be functionally operated and monitored during dynamic as well as thermal tests to increase overall test effectiveness. Practical limitations frequently restrict the extent of operation of the unit during the relatively brief acoustic or vibration tests. Recognizing this problem, VOL I of the Handbook permits extended functional testing with the unit operating and monitored, but conducted at a level 6 dB lower than the required acoustic or vibration test level, after the required environmental exposure has been satisfied.

4.3.3.2 Vehicle Test Baseline.

4.3.3.2.1 Standard Criteria. The contents of paragraphs 5.5, 6.2 and 7.2 in VOL I of the Handbook provide the baseline requirements for development, qualification and acceptance testing of vehicles. The baseline requirements are summarized in Tables VIII and XII of VOL I of the Handbook.

4.3.3.2.2 Rationale for Vehicle Test Baseline. Environmental qualification tests are a formal demonstration that a production vehicle (or prototype) is adequate to successfully sustain specified environmental design levels. These tests are mainly performed to determine if there are factors that may have been overlooked during design, analysis, or manufacturing. Additionally, the environments used during these tests are more severe than those predicted to occur during flight in order to account for variabilities in subsequent production articles and other uncertainties. Qualification test requirements, therefore, incorporate margins which are added to the range of environmental extremes and stresses expected to occur in service. Before qualification testing, the test vehicle should have been subjected to the same controls, inspections, alignments, and tests imposed on flight vehicles. Environmental acceptance tests of the qualification vehicle are not part of the vehicle test baseline.

Environmental acceptance tests are conducted on vehicles to demonstrate flightworthiness and to disclose quality deficiencies in the flight article. Acceptance tests are intended to satisfy these goals by subjecting each flight vehicle to the maximum environmental exposures expected in service. The test program is comprised of a series of tests; some are required tests, and some defined as "other" tests. The "other" tests augment the required tests and these are selected in accordance with the goals and characteristics of a given vehicle program.

4.3.3.2.3 Guidance for Use of Vehicle Test Baseline. The suggested sequence of environmental tests is based on two considerations: preserving the sequence or concurrent nature of the service environments, and assuring that potential failures will be detected as early in the test sequence as possible. Therefore, dynamic tests, which simulate the launch and ascent environment and are generally of short duration with limited performance testing, should precede thermal vacuum and thermal cycling tests, which simulate long duration orbital environments where greater opportunity is afforded for more extensive diagnostic testing. However, in recognition of program-peculiar requirements, such as the buildup sequence and logistic considerations, the order of testing in VOL I of the Handbook is only a suggested sequence. However, the sequencing used should recognize that the thermal vacuum test is an orbital performance check that should be run towards the end of the sequence.

It is extremely important that mechanical and electrical functional tests be conducted before and after each environmental test. These functional tests provide the criteria for judging successful survival of the vehicle in a given test environment. They should be designed to verify that performance of units and of the vehicle meets specification requirements, that the units and the vehicle are compatible with ground support equipment, and that all software used is validated. In addition, electrical functional tests should include negative logic testing to verify lockout, to assure that no function other than the intended function was performed, and to verify that the signal was not present other than when programmed. To the extent practicable, functional tests should also be designed so that a database of critical parameters can be established for trend analysis. This is accomplished by measuring the same critical parameters in all functional tests conducted before, during, and after each of the baseline environmental tests.

The trend data and the final ambient functional test conducted prior to shipment of the vehicle to the launch base provide the data to be used as success criteria during launch base testing. For this reason, vehicle level functional tests should be designed so that they can be duplicated, as nearly as possible, at the launch base.

It is also important to perform functional tests of vehicle subsystems while the environment is being imposed. This is especially important for thermal vacuum tests of upper stage and space vehicles, since these vehicles are expected to be fully operational under these conditions. It is usually considered appropriate during acoustic or vibration tests to have the vehicle in an operating mode representative of launch and ascent. For space and upper stage vehicles the launch and ascent time period usually involves a minimum level of functional performance, with many subsystems inoperative. When possible, however, dynamic tests should be performed on fully functional vehicles with their performance monitored for intermittents. Defects such as improper mounting or intermittents, which otherwise escape detection by pre- and post-test functional checks, can reveal themselves during environmental tests. For example, intermittents may be caused by foreign bodies, contaminants, inadequate clearances, cracks, debonds, and damaged connectors that might only be revealed during environmental tests

4.3.4 Alternative Test Strategies. VOL I of the Handbook provides several alternatives to the baseline test requirements recommended for a low risk program. Generally, the alternative strategies provide cost benefits for space vehicle programs planning to build a small number of satellites (1 to on the order of 10 vehicles). Cost savings are realized by using some or all of the qualification hardware for flight. Some increase in risk may occur and this may be technical or economic or both.

4.3.4.1 Spares Strategy. Of the alternative strategies, the Spares Strategy provides the least increase in risk with modest cost savings. This strategy may be a reasonable choice for a low risk program anticipating the procurement of 5 to 10 vehicles. Additional risk may be incurred because fatigue margins of the qualification hardware used for flight are not known.

4.3.4.2 Flightproof Strategy. The flightproof strategy may provide cost savings with some increase in risk. This strategy may be the most reasonable for a program anticipating the procurement of fewer than five vehicles and where the design is expected to evolve such that each of the vehicles may be somewhat different than its predecessor(s). Additional risk may be incurred because a) reduced test margins allow design deficiencies to remain undetected and b) fatigue margins of flight hardware are not demonstrated.

4.3.4.3 Protoqualification Strategy. The protoqualification strategy can provide cost savings for programs anticipating the procurement of a single vehicle or a small number of vehicles where each vehicle is expected to be identical in design and construction. Cost savings are realized by not having to procure a dedicated set of qualification hardware since the protoqualification hardware is less rigorously tested than would be in normal qualification and then used for flight. Additional risk may be incurred because a) reduced test margins allow possible design deficiencies to remain undetected and b) the protoqualification hardware used for flight has unknown fatigue margins.

4.3.4.4 Combination Test Strategies. Combinations of strategies may provide the best economic versus technical risk tradeoffs for programs planning to use hardware with substantial differences in heritage, complexity and design state-of-the-art. For example, a vehicle incorporating flight proven units with state-of-the-art new design units may apply the protoqualification strategy to the flight proven units, and the baseline full qualification and acceptance testing is applied to the new design units. At the vehicle level, a protoqualification or flightproof strategy then may be employed depending on considerations such as those noted in paragraphs 4.3.4.2 and 4.3.4.3.

4.3.5 Test Thoroughness vs. Risk. The acceptable risk is probably the most difficult factor to evaluate. Risk is not only somewhat subjective, but may vary greatly depending upon who is at risk, what might fail, what is the criticality of the failure should it occur, and who is making the estimate. The problem for the test planner is to ensure that the hardware, software, and procedures are validated prior to launch without conducting tests that are not cost effective. Studies (technical references 1. 2. 3 and 4) related to examining the correlation between test thoroughness and flight performance or risk, have shown that the early orbit failure rate decreases with more thorough ground testing. For example, Figures 2 and 3 reproduced from technical reference 4, illustrate that past programs that utilized the MIL-STD-1540B test baseline of full qualification and acceptance testing tended to have fewer orbital failures than those that performed substantially less testing. The Test Thoroughness Index (TTI), is a measure of the percent compliance with the aggregate of unit through system test requirements contained in the baseline test requirements of VOL I of the Handbook. A TTI of 100 percent indicates a test program that contains all of the required acceptance and qualification tests of VOL I from the unit through the system level of assembly. In general the alternative strategies, Section 8 of VOL I, could be expected to have a TTI in the range of 70 to 85 percent. With present information it is not possible to quantify the increase in risk that may be associated with a test program designed around the alternative strategies vs. the VOL I baseline test program. For any given program the optimum balance between test thoroughness and life cycle cost is dependent on many factors such as hardware design maturity, complexity, production quantities, service environment, and acceptable risk.

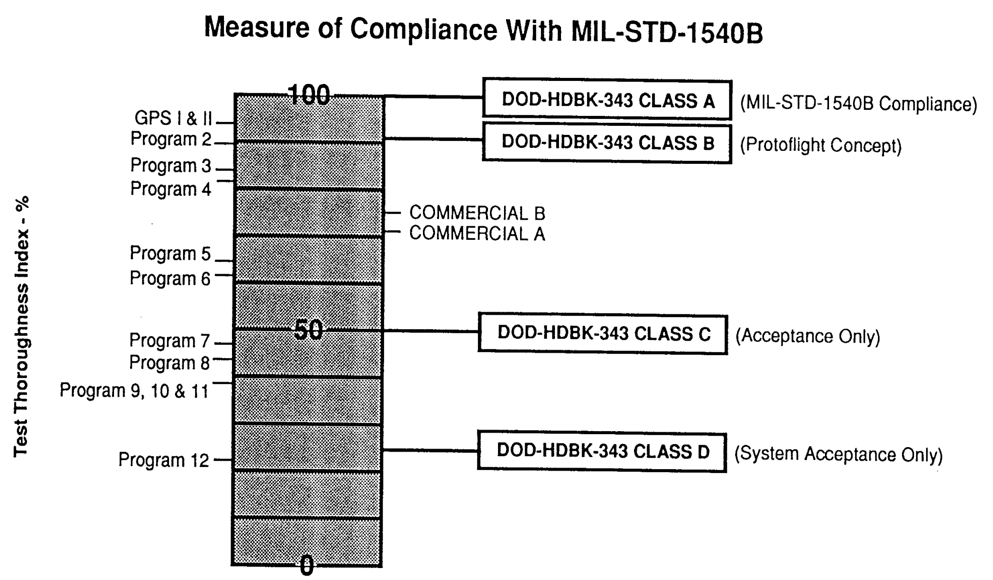

Figure 2. Test Thoroughness Index of Various USAF and Commercial Satellites

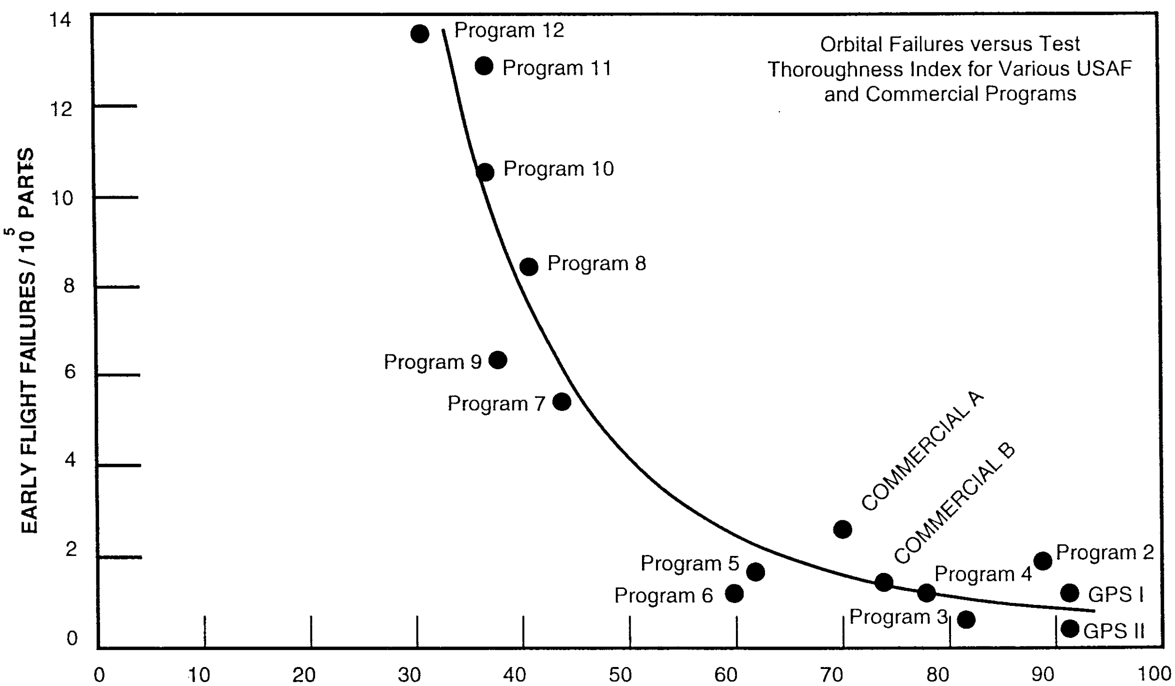

Figure 3. Orbital Failures Versus Test Thoroughness Index of Various USAF and Commercial Programs

4.3.6 Environmental Stress Screening. VOL I of the Handbook incorporates environmental stress screening (ESS) concepts and philosophy into requirements for vibration, acoustic, and thermal cycle testing at the unit through system level of assembly. Fundamentally the ESS concept establishes minimum test levels for acceptance testing which are irrespective of the service environments, if service environments are benign. Qualification test criteria may also be affected since they must encompass, with margin, both the acceptance test conditions and the service environments. Further discussion of the ESS requirements can be found in sections of this Handbook, related to vibration, (Paragraph 5.3.2.1) acoustic (Paragraph 5.4.2.1) and thermal cycle testing (Paragraph 5.6.2).

4.4 QUALIFICATION BY SIMILARITY

4.4.1 Standard Criteria for Qualification by Similarity. VOL I of the Handbook does not directly address criteria for the qualification of items by similarity; however, it does provide the standard test baselines for comparison.

4.4.2 Rationale for Qualification by Similarity. The continued production and use of items designed for launch vehicles, upper stages and space vehicles of one program is of interest to every program office. If existing qualified hardware can be used, not only are the design, tooling, and qualification costs eliminated for subsequent programs, but the continuing usage of the same item increases confidence in the item's reliability. Of course, to accommodate specific requirements of another program, it may not be possible to use the same exact item, so there may be changes required in the item or in its testing. If those changes are within reasonable bounds, then qualification of the revised item by similarity should be considered.

4.4.3 Guidance for Qualification by Similarity

4.4.3.1 Unit Criteria. If unit "A" is to be considered a candidate for qualification by similarity to unit "B" that has already been qualified for launch vehicle, upper stage or space vehicle use, then all of the following conditions should apply:

a. Unit "B" was not qualified by similarity or analysis.

b. Unit "B" was a representative flight article

c. The environments, both amplitude and duration, encountered by unit "B" during its qualification or flight history have been equal to or more severe than the qualification environments intended for unit "A."

d. Units "A" and "B" were produced by the same manufacturer using identical tools, manufacturing processes, and quality control procedures.

e. Unit "B" should have successfully passed a post-environmental functional test series, without need for waivers associated with performance indicating survival of the qualification stresses.

f. Units "A" and "B" should perform similar functions, with "B" having equivalent or greater operating life with variations only in terms of performance such as accuracy, sensitivity, formatting, and input-output characteristics.

g. Supporting documentation for unit B is available and includes specifications, drawings, qualification test procedures, qualification and acceptance test reports, problem failure reports with closure history, test waivers, and flight history summary.

h. Unit "A" should be a minor variation of unit "B." Dissimilarities of safety, reliability, maintainability, weight, mechanical configuration, thermal effects, dynamic response, and structural, mechanical and electrical configurations require that unit A characteristics be enveloped by the characteristics of unit B. Minor design changes involving substitution of piece parts and materials with equivalent reliability items can generally be tolerated. Design dissimilarities resulting from addition or subtraction of piece parts and particularly moving parts, ceramic or glass parts, crystals, magnetic devices, and power conversion or distribution equipment usually compromise qualification based on similarity.

4.4.3.2 Criteria for Other Items. In some cases, the item to be qualified by similarity is not a unit but is some other level of assembly, such as a subsystem. In that case, the criteria for the item to be qualified by similarity would be the same as though the item were a unit (see Paragraph 4.4.3.1).

4.4.3.3 Partial Testing. It is recognized that in some cases, where all the criteria in paragraph 4.4.3.1 are not satisfied, qualification based on engineering analysis plus partial testing may be permissible. In this case, negotiation between the contracting officer and the contractor may result in an abbreviated testing program satisfactory for qualification of the unit or item in question. The acceptability of qualification by similarity should be documented by test reports, drawings, and analyses. This justification or proof of qualification should be documented, and the burden of proof of qualification is the responsibility of the contractor.

4.5 PROPULSION EQUIPMENT TESTING

VOL I of the Handbook does not address performance testing of propulsion equipment. Generally, however, the test requirements of VOL I of the Handbook should be applied to propulsion equipment in accordance with the self tailoring instructions contained within VOL I of the Handbook.

4.6 TEST REQUIRMENTS FOR SPECIFIC PROGRAMS.

To make the requirements clear for a particular program, and to assist in the tailoring process, either Table I or Table II, or an adaptation thereof, should be completed. Table I can be used when primarily broad, general tailoring of the requirements is desired, with only a few specific test or test items to be specially treated differently. Table II can be used when detailed tailoring of the requirements is desired. These tables provide a recommended format for stating changes to the stringency or applicability of the baseline requirements appearing in the Handbook, Vol I relating to the use of “shall” versus “should” and to “required” versus “other” categories of vehicles, subsystems, and units. A sample of a completed Table I and II appears in Tables III and IV, respectively.

4.7 IN-PROCESS CONTROLS.

In-process controls are almost always a more cost-effective way of avoiding defects than the imposition of tests and inspections on completed units. Therefore, appropriate in-process controls and other quality management steps should be imposed to achieve the high-quality and reliability goals of space and launch systems. The acceptance testing requirements are intended to be the last step in assuring the quality of each production item. When it has been thoroughly demonstrated that the purpose of an acceptance testing requirement has been met by the in-process controls or other quality management steps implemented by the manufacturer, consideration should be given to reduce the test to a sampling test, or if appropriate, for deletion of the test.

TABLE I. Requirements Applicability Matrix, General Form.

The matrix designators are as follows:

A	=	Applicable as written	-	"Shall" defines minimum requirements.
			-	"Should" and may" language denotes guidance.
				"Other" test denotes conduct to be evaluated.
F	=	Fully Applicable	-	All "should" or "may" language replaced with "shall".
			-	All "other" tests changed to "required" tests.
G	-	Guidance only	-	All information provided as good practice.
N	-	Not Applicable	-	Requirements are not applicable.

Level of Assembly or Specific Item	Section							
					8. Alternative Strategies			
	4. General Requireme	5. Development Te	6. Qualification Tes	7. Acceptance Tes	8.1 Spare	8.2 Flightproo	8.3 Protoqualificat	9. Prelaunch Vehicle Operational Tests
Units								
Subsystems								
Vehicles								
Integrated System								
Other Items:								

TABLE II Requirements Applicability Matrix, Detailed Form,
(first of 5 pages)

The matrix designators are as follows:

A = Applicable as written - "Shall" defines minimum requirements.

- "Should" and may" language denotes guidance.

"Other" denotes test that may be required subject to an evaluation.

F = Fully Applicable - All "should" or "may" language replaced with "shall".

- All "other" tests changed to "required" tests.

G - Guidance only - All information provided as good practice.

N - Not Applicable - Requirements are not applicable.

Section		Units	Sub-systems	Vehicles	Integrated Systems
3.	DEFINITIONS				
4.2	TESTING PHILOSOPHY				
4.3	PROPULSION EQUIPMENT TESTS				
4.3.1	Engine LRU Acceptance Testing				
4.3.2	Engine LRU Qualification Testing				
4.4	FIRMWARE TESTS				
4.5	INSPECTIONS				
4.6	TEST CONDITION TOLERANCES				
4.7	TEST PLANS AND PROCEDURES				
4.7.1	Test Plans				
4.7.2	Test Procedures				
4.8	RETEST				
4.8.1	During Qualification or Acceptance				
4.8.2	During Prelaunch Validation				
4.8.3	During Operational Tests and Evaluations				
4.9	DOCUMENTATION				
4.9.1	Test Documentation Files				
4.9.2	Test Data				
4.9.3	Test Log				

(table continued next page)

TABLE II Requirements Applicability Matrix, Detailed Form (continued).

(Second of 5 pages)

	Section	Units	Sub-systems	Vehicles	Integrated System
5.1	GENERAL DEVELOPMENT TESTS				
5.2	PMP DEVELOPMENT TESTS AND EVALUATIONS				
5.3	SUBASSEMBLY DEVELOPMENT TESTS, IN-PROCESS TESTS AND INSPECTIONS				
5.4	UNIT DEVELOPMENT TESTS				
5.4.1	Structural Composite Development Tests				
5.4.2	Thermal Development Tests				
5.4.3	Shock & Vibration Isolator Development				
5.5	VEHICLE AND SUBSYSTEM DEVELOPMENT TESTS				
5.5.1	Mechanical Fit Development Tests				
5.5.2	Mode Survey Development Tests				
5.5.3	Structural Development Tests				
5.5.4	Acoustic and Shock Development Tests				
5.5.5	Thermal Balance Cevelopment Tests				
5.5.6	Transport & Handling Development Tests				
5.5.7	Wind-tunnel Development Tests				

(table continued next page)

TABLE II Requirements Applicability Matrix, Detailed Form (Continued). (third of 5 pages)

Section		Units	Sub-systems	Vehicles	Integrated Systems
6.1	GENERAL QUALIFICATIONS TESTS				
6.1.1	Qualification Hardware				
6.1.2	Test Levels and Durations				
6.1.3	Thermal Vacuum and Cycle Tests				
6.1.4	Acoustic & Vibration Qualification Tests				
6.2	VEHICLE QUALIFICATION TESTS - Baseline				
6.2.1	Functional Test, Vehicle Qualification				
6.2.2	EMC, Vehicle Qualification				
6.2.3	Shock Test, Vehicle Qualification				
6.2.4	Acoustic Test, Vehicle Qualification				
6.2.5	Vibtration Test, Vehicle Qualification				
6.2.6	Pressure and Leakage, Vehicle Qualification				
6.2.7	Thermal Cycle Test, Vehicle Qualification				
6.2.8	Thermal Balance Test, Vehicle Qualification				
6.2.9	Thermal Vacuum Test, Vehicle Qualification				
6.2.10	Mode Survey Test, Vehicle Qualification				
6.3	SUBSYSTEM QUALIFICATION TESTS - Baseline				
6.3.1	Structural Static Load Test				
6.3.2	Vibration Test				
6.3.3	Acoustic Test				
6.3.4	Thermal Vacuum Test				
6.3.5	Separation Test				
6.4	UNIT QUALIFICATION TESTS - Baseline				
6.4.1	Functional Test				
6.4.2	Thermal Cycle Test				
6.4.3	Thermal Vacuum Test				
6.4.4	Vibration Test				
6.4.5	Acoustic Test				
6.4.6	Shock Test				
6.4.7	Leakage Test				
6.4.8	Pressure Test				
6.4.9	Acceleration Test				
6.4.10	Life Test				
6.4.11	EMC Test				
6.4.12	Climatic Test				

(table continued next page)

TABLE II Requirements Applicablity Matrix, Detailed Form
(Continued). (fourth of 5 pages)

Section		Units	Sub-systems	Vehicles	Integrated Systems
7.1	GENERAL ACCEPTANCE TESTS				
7.1.1	Temperature Range & No. of Thermal Cycles				
7.1.2	Acoustic Environment				
7.1.3	Vibdration Environment				
7.1.4	Storage Tests				
7.2	VEHICLE ACCEPTANCE TESTS - Baseline				
7.2.1	Functional Test				
7.2.2	EMC Test				
7.2.3	Shock Test				
7.2.4	Acoustic Test				
7.2.5	Vibration Test				
7.2.6	Pressure and Leakage Test				
7.2.7	Thermal Cycle Test				
7.2.8	Thermal Vacuum Test				
7.3	SYBSYSTEM ACCEPTANCE TESTS - Baseline				
7.3.1	Proof Load Test				
7.3.2	Proof Pressure				
7.4	UNIT QUALIFICATION TESTS - Baseline				
7.4.1	Functional Test				
7.4.2	Thermal Cycle Test				
7.4.3	Thermal Vacuum Test				
7.4.4	Vibration Test				
7.4.5	Acoustic Test				
7.4.6	Shock Test				
7.4.7	Leakage Test				
7.4.8	Proof Pressure Test				
7.4.9	Proof Load Test				
7.4.10	Wear-in Test				
7.4.11	EMC Test				

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TABLE II Requirements Applicablity Matrix, Detailed Form (Continued).
(last of 5 pages)

	Section	Units	Sub-systems	Vehicles	Integrated Systems
8.1	SPARES STRATEGY	☐			
8.2	FLIGHTPROOF STRATEGY	☐	☐	☐	
8.2.1	Vehicle Tests			☐	
8.2.2	Subsystem Tests		☐	☐	
8.2.3	Unit Tests	☐	☐	☐	
8.3	PROTOQUALIFICATION STRATEGY	☐	☐	☐	
8.3.1	Vehicle Tests			☐	
8.3.2	Subsystem Tests		☐		
8.3.3	Unit Tests	☐			
8.4	COMBINATION TEST STRATEGIES	☐	☐	☐	
9	PRELAUNCH TEST STRATEGIES			☐	☐
9.1	GENERAL REQUIREMENTS			☐	☐
9.2	TEST FLOW			☐	☐
9.3	TEST CONFIGURATION			☐	☐
9.4	TEST DESCRIPTIONS			☐	☐
9.4.1	Functional Test			☐	☐
9.4.2	Propulsion Leakage & Function Tests			☐	☐
9.4.3	Critical Ground Support Tests			☐	☐
9.4.4	Compatibility Test, On-orbit System			☐	☐
9.5	FOLLOW-ON OPERATIONAL TESTS			☐	☐
9.5.1	Operational Tests and Evaluations				☐
9.5.2	On-orbit Testing				☐
9.5.3	Tests of Reusable Flight Hardware	☐	☐	☐	

TABLE III. Sample of Table I, Requirements Applicability Matrix, General Form. The matrix designators are as follows:

A =	Applicable as written	-	"Shall" defines minimum requirements.
		-	"Should" and may" language denotes guidance.
			"Other" test denotes conduct to be evaluated.
F =	Fully Applicable	-	All "should" or "may" language replaced with "shall".
		-	All "other" tests changed to "required" tests.
G -	Guidance only	-	All information provided as good practice.
N -	Not Applicable	-	Requirements are not applicable.

Level of Assembly or Specific Item	Section							
					8. Alternative Strategies			
	4. General Requirements	5. Development Tests	6. Qualification Tests	7. Acceptance Tests	8.1 Spares	8.2 Flightproof	8.3 Protoqualification	9. Prelaunch Vehicles & Operational Tests
Units	F	F	F	F	N	N	N	F
Subsystems	F	F	F	F	N	N	N	F
Vehicles	F	F	A	G	N	N	N	F
Integrated System	F	F	G	G	N	N	N	F
Other Items:								
Space Environment	G	G	G	G	G	G	G	G

TABLE IV. Sample of Table II, Requirements Applicability Matrix, Detailed Form. (first of 5 pages)

The matrix designators are as follows:

A = Applicable as written
- "Shall" defines minimum requirements.
- "Should" and may" language denotes guidance.
- "Other" test denotes conduct to be evaluated.

F = Fully Applicable
- All "should" or "may" language replaced with "shall".
- All "other" tests changed to "required" tests.

G - Guidance only - All information provided as good practice.

N - Not Applicable - Requirements are not applicable.

Section		Units	Sub-systems	Vehicles	Integrated Systems
3.	DEFINITIONS	A	A	A	A
4.2	TESTING PHILOSOPHY	A	A	A	A
4.3	PROPULSION EQUIPMENT TESTS	A			
4.3.1	Engine LRU Acceptance Testing	A			
4.3.2	Engine LRU Qualification Testing	F			
4.4	FIRMWARE TESTS	G			
4.5	INSPECTIONS	A	A	A	A
4.6	TEST CONDITION TOLERANCES	A	A	A	A
4.7	TEST PLANS AND PROCEDURES	A	A	A	A
4.7.1	Test Plans	F	F	F	A
4.7.2	Test Procedures	F	F	F	A
4.8	RETEST	A	A	A	A
4.8.1	During Qualification or Acceptance	F	F	F	A
4.8.2	During Prelaunch Validation			A	A
4.8.3	During Operational Tests and Evaluations				A
4.9	DOCUMENTATION				
4.9.1	Test Documentation Files	F	F	F	A
4.9.2	Test Data	F	F	F	A
4.9.3	Test Log	F	F	F	A

(table continued next page)

TABLE IV. Sample of Table II, Requirements Applicability Matrix, Detailed Form. (continued) (second of 5 pages)

	Section	Units	Sub-systems	Vehicles	Integrated Systems
5.1	GENERAL DEVELOPMENT TESTS	A	A	A	A
5.2	PMP DEVELOPMENT TESTS AND EVALUATIONS	F			
5.3	SUBASSEMBLY DEVELOPMENT TESTS,	A			
	IN-PROCESS TESTS AND INSPECTIONS	A			
5.4	UNIT DEVELOPMENT TESTS	A			
5.4.1	Structural Composite Development Tests	F			
5.4.2	Thermal Development Tests	A			
5.4.3	Shock & Vibration Isolator Development	A			
5.5	VEHICLE AND SUBSYSTEM DEVELOPMENT TESTS		G	G	
5.5.1	Mechanical Fit Development Tests		G	A	
5.5.2	Mode Survey Development Tests		G	F	
5.5.3	Structural Development Tests		A	A	
5.5.4	Acoustic and Shock Development Tests		A	F	
5.5.5	Thermal Balance Cevelopment Tests		G	F	
5.5.6	Transport & Handling Development Tests		G	A	
5.5.7	Wind-tunnel Development Tests		G	A	

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TABLE IV. Sample of Table II, Requirements Applicability Matrix, Detailed Form. (continued) (third of 5 pages)

Section		Units	Sub-systems	Vehicles	Integrated Systems
6.1	GENERAL QUALIFICATIONS TESTS	A	A	A	
6.1.1	Qualification Hardware	A	A	A	
6.1.2	Test Levels and Durations	A	A	A	
6.1.3	Thermal Vacuum and Cycle Tests	A	A	A	
6.1.4	Acoustic & Vibration Qualification Tests	A	A	A	
6.2	VEHICLE QUALIFICATION TESTS - Baseline			A	
6.2.1	Functional Test, Vehicle Qualification			A	
6.2.2	EMC, Vehicle Qualification			A	
6.2.3	Shock Test, Vehicle Qualification			A	
6.2.4	Acoustic Test, Vehicle Qualification			F	
6.2.5	Vibtration Test, Vehicle Qualification			G	
6.2.6	Pressure and Leakage, Vehicle Qualification			A	
6.2.7	Thermal Cycle Test, Vehicle Qualification			G	
6.2.8	Thermal Balance Test, Vehicle Qualification			A	
6.2.9	Thermal Vacuum Test, Vehicle Qualification			A	
6.2.10	Mode Survey Test, Vehicle Qualification			F	
6.3	SUBSYSTEM QUALIFICATION TESTS - Baseline		A		
6.3.1	Structural Static Load Test		A		
6.3.2	Vibration Test		A		
6.3.3	Acoustic Test		A		
6.3.4	Thermal Vacuum Test		A		
6.3.5	Separation Test		A		
6.4	UNIT QUALIFICATION TESTS - Baseline	A			
6.4.1	Functional Test	A			
6.4.2	Thermal Cycle Test	A			
6.4.3	Thermal Vacuum Test	A			
6.4.4	Vibration Test	A			
6.4.5	Acoustic Test	A			
6.4.6	Shock Test	A			
6.4.7	Leakage Test	A			
6.4.8	Pressure Test	A			
6.4.9	Acceleration Test	A			
6.4.10	Life Test	A			
6.4.11	EMC Test	A			
6.4.12	Climatic Test	A			

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TABLE IV. Sample of Table II, Requirements Applicability Matrix, Detailed Form. (continued) (fourth of 5 pages)

	Section	Units	Sub-systems	Vehicles	Integrated Systems
7.1	GENERAL ACCEPTANCE TESTS	A	G	A	
7.1.1	Temperature Range & No. of Thermal Cycles	A	G	A	
7.1.2	Acoustic Environment	A	G	A	
7.1.3	Vibdration Environment	A	G	A	
7.1.4	Storage Tests	A		A	
7.2	VEHICLE ACCEPTANCE TESTS - Baseline			A	
7.2.1	Functional Test			A	
7.2.2	EMC Test			A	
7.2.3	Shock Test			A	
7.2.4	Acoustic Test			A	
7.2.5	Vibration Test			A	
7.2.6	Pressure and Leakage Test			A	
7.2.7	Thermal Cycle Test			A	
7.2.8	Thermal Vacuum Test			A	
7.3	SYBSYSTEM ACCEPTANCE TESTS - Baseline		A	A	
7.3.1	Proof Load Test		A		
7.3.2	Proof Pressure		A		
7.4	UNIT QUALIFICATION TESTS - Baseline	A			
7.4.1	Functional Test	A			
7.4.2	Thermal Cycle Test	A			
7.4.3	Thermal Vacuum Test	A			
7.4.4	Vibration Test	A			
7.4.5	Acoustic Test	A			
7.4.6	Shock Test	A			
7.4.7	Leakage Test	A			
7.4.8	Proof Pressure Test	A			
7.4.9	Proof Load Test	A			
7.4.10	Wear-in Test	A			
7.4.11	EMC Test	A			
		A			
		A			

(table continued next page)

TABLE IV. Sample of Table II, Requirements Applicability Matrix, Detailed Form. (continued) (last of 5 pages)

Section		Units	Sub-systems	Vehicles	Integrated Systems
8.1	SPARES STRATEGY	G			
8.2	FLIGHTPROOF STRATEGY	N	N	N	
8.2.1	Vehicle Tests			N	
8.2.2	Subsystem Tests		N		
8.2.3	Unit Tests	N			
8.3	PROTOQUALIFICATION STRATEGY	N	N	N	
8.3.1	Vehicle Tests			N	
8.3.2	Subsystem Tests		N		
8.3.3	Unit Tests	N			
8.4	COMBINATION TEST STRATEGIES	G	N	N	
9	PRELAUNCH TEST STRATEGIES			G	A
9.1	GENERAL REQUIREMENTS			G	A
9.2	TEST FLOW			G	A
9.3	TEST CONFIGURATION			G	A
9.4	TEST DESCRIPTIONS			G	A
9.4.1	Functional Test			G	A
9.4.2	Propulsion Leakage & Function Tests			G	A
9.4.3	Critical Ground Support Tests			G	A
9.4.4	Compatibility Test, On-orbit System			G	A
9.5	FOLLOW-ON OPERATIONAL TESTS			G	A
9.5.1	Operational Tests and Evaluations				A
9.5.2	On-orbit Testing				A
9.5.3	Tests of Reusable Flight Hardware	A	A	A	

SECTION 5.

TEST PHILOSOPHY AND CRITERIA

5.1 INSPECTIONS

5.1.1 Standard Criteria for Inspections. Paragraphs 4.5 and 6.4 of VOL I of the Handbook provide information for establishing inspection requirements before and after testing of hardware.

5.1.2 Rationale for Inspections. Inspections are performed before and after testing primarily to ascertain whether the test conditions altered the hardware in an unacceptable manner.

5.1.3 Guidance for Inspections. The results of inspections must be recorded in sufficient detail to determine significant changes in the condition of the hardware undergoing test. Disassembly of flight hardware or the removal of covers for inspection after test is generally not allowed unless it is planned to repeat the normal set of acceptance tests. However, disassembly of qualification hardware not planned for flight including removal of covers for inspection is encouraged.

5.2 FUNCTIONAL TESTS

5.2.1 Standard Criteria for Functional Tests. Requirements related to vehicle and units for qualification and acceptance testing are described in paragraphs 6.2.1, 6.4.1, 7.2.1 and 7.4.1 in VOL I of the Handbook. In addition, for thermal, shock, acoustic, and random vibration testing, supplemental requirements are specified which include aspects of functional testing.

5.2.2 Rationale for Functional Testing. The assembly of hardware composed of mechanical and electrical parts has the possibility of containing flaws due to workmanship error, design errors, interface incompatibilities, and process problems. In addition, proof of performance eventually needs to be shown to satisfy customer requirements that the item being delivered meets the conditions of service. Functional testing is part of risk mitigation by being: 1) perceptive enough to find faults or flaws and 2) certifiable to prove form, fit and function aspects of the hardware. A third area exists, that of qualifying the hardware beyond certifiable aspects to prove robustness. VOL I of the Handbook addresses functional testing in terms of meeting these objectives to show that hardware performs within the worst case operating conditions expected during its life cycle and that adequate margin exists to prove that the design is sufficiently robust to accommodate manufacturing variability, test inadequacies, and design uncertainties. It is important that functional testing be designed to be perceptive. Perceptiveness is the ability to obtain measurable information that can indicate a deficiency or flaw. Functional testing, along with physical inspection and environmental data observed during the test, form the basis of perceptiveness. Certain parameters are used for pass/fail criteria and others might be used for discrepancy resolution. Pretest functional testing forms a baseline performance record so any change in trend from the exposure to test conditions can be noted. Perceptiveness is an important aspect of certification testing and supports the objective of proving that a product operates as required.

5.2.3 Guidance for Functional Tests. Specific guidance for every functional test of digital, RF, optical, or mechanical equipment could not be specified in this document. In general, specific performance requirements are certified by using verification matrices in standards and control documents. Functional tests are called out against specific performance standards in the design specification paragraph by paragraph and form the matrix defining inspection, demonstration, test, or analysis techniques.

In general, test perceptiveness is highest when it involves measuring continuity, trends, threshold sensitivity, or the variation of input parameters with steady state and/or transient conditions and observing the output characteristics. The latter can involve showing that output parameters meet specified performance within tolerance or that transfer relationships between the input and output maintain stability or signal quality requirements. Examples might be the axial ratio for an antenna under thermal elastic distortion, control loop gain and phase margins, nonlinear amplitude intermodulation products, N^2 energy measurements over a 5x5 pixel matrix while the optics train is under jitter excitation, or the torque response of a multi-hinge solar array with misaligned joints. Examples of continuity might be bridge wire resistance testing, polarity testing of an inertial navigation unit relative to an actuator when rotated, and command response behavior. Trend testing can involve simply the measurement of a performance parameter over time. Examples might be battery charge capacity, channel power, valve response time, wheel spin down time, antenna surface reflectivity, or drift rates. Threshold sensitivity might involve varying a frequency offset beyond specification and determining when sync occurs, raising a source signature until the system acquires and moving the source until track is determined, or increasing temperatures until a heater switches off.

Another aspect of functional testing covered in VOL I of the Handbook is the exercise of the modes of an item, redundant paths, autonomous functions, satisfaction of preconditions, and monitor testing. Modes of operation during the mission profile should be demonstrated at least once for qualification of an item or system under test. This needs to be done for all applicable events as they occur in the launch and operational sequence. Redundancy should be verified for every item involved in a signal path which includes wire, switches, and devices. Problems can be subtle, like a battery cell out condition where a blocking diode prevents the battery from shorting out. Other examples of problems are certain bypass modes within a unit, like in a command authenticator, which is not part of a normal operating mode, or the firing of a redundant bolt cutter with the primary cutter bound in the bolt. Although not always identified as a redundant path, they permit function of a chain of events and thus should be certified or at least qualified when testing to validate the flight configuration. Autonomous functions involve proof that the mode operates under the conditions for which it was designed. This may require simulation, such as a vehicle undergoing a safe haven mode, or piecemeal application such as monitoring a deployment sequence where only first release conditions are tested. Proof that a circuit only fires under set preconditions or the vehicle can not be commanded unless certain sequences are established are examples of showing satisfaction of preconditions. Finally, the monitoring of current to a unit under test, or RF power while temperatures change on a receiver are examples of perceptive monitoring during environmental exposures to identify problems and allow problem isolation. Care should be taken when digital sampling is used during monitoring to assure that the rates are high enough to detect problems. For example, using telemetry current monitors that sample at a 25 Hz rate may not be sufficient to detect a card arcing to the chassis during a vibration test due to excessive card displacement.

Special consideration should be given when ground test effects versus the space environments are important. For example gravity effects can aid a deployment. Ambient temperature and pressure could also aid deployment where worst-case conditions are not simulated. In cases where the space effect can not be reasonably simulated, the performance parameters specified under space conditions may need to be modified to reflect the effect of ground conditions.

5.3 VIBRATION TESTS

5.3.1 Standard Criteria for Vibration Tests. Definitions related to the vibration environment are presented in paragraphs 3.3.2, 3.3.3, and 3.3.5 of VOL I of the Handbook. Vibration requirements that apply to systems, subsystems, and units for development, qualification and acceptance testing are described in VOL I of the Handbook paragraphs 6.1.4, 6.2.5, 6.3.2, 6.4.4, 7.1.3, 7.2.5, and 7.4.4. Table's III, IV, VII, and XI and Figures 5 and 6 of VOL I of the Handbook provide tolerance and summary criteria for vibration testing.

5.3.2 **Rationale for Vibration Tests**. The random vibration environment imposed on space equipment is due to the liftoff acoustic field, aerodynamic excitations, and transmitted structure-borne vibration. Such vibration environments occur during ignition, liftoff, ascent and engine operation. The response may result from an acoustic forcing function, a start transient, engine self generated vibration or from vibration transmitted through the structure from other sources.

The maximum predicted vibration environments are needed early in the development cycle of a vehicle to establish design and testing requirements for units and subsystems. Often, the vibration environments must be established well before the vehicle structural design has matured. Information available to establish predictions is usually very limited. The variability of the environment is great, due to the large number of parameters which influence levels for any given unit location. Substantial cost and schedule impacts are incurred if levels are raised after release of procurement contracts. For these reasons, considerable care and foresight are needed in establishing maximum predicted vibration environments.

Vibration environments in space equipment at frequencies above approximately 50 Hz are primarily the result of acoustic forcing functions. The vibration environment in a given vehicle will be proportional to the level of acoustic excitation. Vibration levels throughout a vehicle are highly variable and dependent upon factors such as orientation, local resonances, damping, structural mass loading, and degree of coupling with adjacent structures. In establishing a maximum predicted environment, one must decide whether this is to be the maximum environment for a specific axis, for a specific location, for a given zone, or possibly the maximum for the entire vehicle or family of vehicles. Selection of the correct maximum vibration environment for a particular program situation will be dependent on considerations such as the number of vehicles in the program, the design maturity of the vehicle, and available test data. It is recommended that maximum predicted vibration levels be established for vehicle zones. In general, the practice of establishing vibration levels for individual units for specific locations should be avoided. Experience on past programs has shown that it can lead to numerous specification changes late in the program and costly retests.

5.3.2.1 **Technical Basis-Environmental Stress Screening**. Exposure of units, subsystems and systems to a test environment represents the principle demonstration of the item's ability to function during and after exposure to the flight environment. When the flight environment is benign, demonstration of functionality is achieved by performing a test using a minimum environment sufficient to detect workmanship defects. The chosen level and/or duration represents a test that serves as an "environmental stress screen" (ESS). An environmental stress screen for a dynamic environment should consist of an amplitude and duration sufficient to result in structural responses that apply stresses to mechanical and electrical interfaces, subject units to vibration responses high enough to detect latent defects and verify the integrity of the item dynamic characteristics prior to use. In many instances, the vibration test environment is that associated with a simulation of the flight environment. There are however, cases where the flight environment is relatively benign and therefore, for environmental stress screening purposes amplitudes and duration's are in excess of those experienced during flight. The selection of a minimum (or ESS) vibration test level is based upon review of the industry minimum test requirements. Figures 5 and 6 of VOL I of the Handbook are minimum vibration levels for units and vehicles respectively that will provide an adequate screen.

The spectrum shape for unit testing, Figure 5, was developed with a recognition that most units are designed with the first resonance above 100 Hertz. Experience has shown that some organizations increase the levels below 100 Hertz to account for low frequency phenomena peculiar to their application. The constant minimum test level of 0.04 g^2/Hz from 150 to 600 Hertz of VOL I of the Handbook recognizes that most units have their principle resonances in this frequency range. The roll-off above 600 Hertz recognizes the reduced damage potential for higher frequency energy. An amplitude of spectral energy was selected based upon industry environmental stress screening values currently in use.

5.3.3 **Guidance for Vibration**.

5.3.3.1 Statistical Basis. Vibration levels for units are often based on combining multiple axes and locations within a vehicle. The following discussion provides guidance for establishing levels at a given location and axis within a vehicle. Depending on program considerations these individual axis/location estimates may or may not be combined to establish unit test criteria.

Statistical estimates for vibration (also, for acoustic and shock) are described in paragraph 3.3.2 of VOL I . Acceptance testing is at the maximum expected level established as the P95/50 level (consistent with past 1540 practice), as long as workmanship requirements do not impose a higher level (see Paragraph 5.3.3.3). The P95/50 is defined to be the level not exceeded on at least 95 percent of flights, estimated probabilistically with 50-percent confidence. For qualification, the basis is the extreme expected level defined to be the P99/90 level. This statistical basis replaces the past use of a fixed qualification margin in order to better account for the typically very limited number of independent samples of environmental data. The P99/90 level is consistent with a practice for statistical evaluations of dynamic loads. Lognormal flight-to-flight variability is assumed. The estimate for nonexceedence at the P-percentile with C-percent confidence is given by

$$E_x(P/C) = x_m + (Z_P + Z_C / N^{1/2})\ \sigma_x$$

E_x is the level being estimated for the Pth percentile and Cth confidence percentile, respectively (for random vibration, the spectral density in g^2/Hz).

x_m is the log mean of the levels from N independent samples (different flights or, if pertinent, ground tests). Namely, x_m is the value whose log is the mean of the logs of the N samples of data. Equivalently, x_m is the Nth root of the product of the N values of x.

σ_x is the standard deviation of x expressed in dB. It is taken to be 3 dB in the absence of a database sufficient to obtain a reliable estimate. The estimation of σ_x from a few samples of data is considered to be so unreliable as to be potentially misleading. The assumption of lognormality with 3dB standard deviation is based on a repeated measurement on over 40 flights of a launch vehicle.

Z_P and Z_C are the standardized normal variables for nonexceedence of the Pth percentile and Cth confidence percentile, respectively. For most cases required by the standard, the needed values of Z_p and Z_p would be for the 50, 90, 95 and 99 percentiles. From a standard normal table, Z_p and Z_c for the previously stated percentiles would respectively be 0, 1.282, 1.645, and 2.322.

An example calculation for the typical Pth percentiles and Cth confidence percentiles used in the standard, E_x (95/50) and E_x (99/90), is as follows:

$$E_x(95/50) = X_m + (1.645 + 0/N^{1/2})\ 3dB$$
$$= X_m + 4.9dB \text{ which is independent of N}$$

$$E_x(99/90) = X_m + (2.322 + 1.282/N^{1/2})\ 3dB$$
$$= X_m + 7.0 + 3.9/\ N^{1/2}\ dB$$

For this example, if X_m is characterized by a single sample (N=1), $E_x(95/50) = X_m + 4.9$ dB and $E_x(99/90) = X_m + 10.9$ dB, the difference between the two being 6.0dB.

5.3.3.2 **Qualification Testing**. As stated in paragraph 6.1.4 of VOL I of the Handbook, qualification is designed to encompass both repeated acceptance testing and the extreme expected flight environment (P99/90), accounting for both fatigue and maximum stress types of failure potential. Either of two approaches, discussed in the sections below, may be selected. It is assumed that no other significant contributors to the extreme expected vibration environments exist, such as handling and transportation. If this is not true, such contributions must be taken into account.

5.3.3.2.1 **Accelerated Testing**. One qualification approach involves acceleration (use of higher level for a shorter time) of the acceptance testing using the P99/90 level, based on fatigue equivalence. This is a simpler practice from a test standpoint, but involves greater conservatism in its severity (assumption about fatigue damage and possibility of nonlinear increase in severity at the P99/90 level relative to P95/50). It is assumed that the difference in the P99/90 and P95/50 spectra in dB, actually accomplished due to test tolerances, has a uniform probability density between M-T and M+T, where M is the nominal margin between the two levels in dB and T is the test tolerance in dB (assumed to be the same for acceptance and qualification). The time reduction factor, α, for the fatigue exponent p = 4 is given by

$$\alpha = 10^{M/5} [1 + (4/3)\sinh^2(T/M)]^{-1}$$

Values for various M,T combinations appear in Table VII of VOL I of the Handbook. In order to deal with the possibility that the tolerances may be different for the two types of tests or unequal in the positive and negative senses in either test, T is taken to be the greater of the positive tolerance for acceptance and the absolute value of the negative tolerance for qualification.

For this approach, involving test only at the P99/90 level, the time of exposure per axis, T_Q, is given by

$$T_Q = 4[(1/\propto)\ T_{AMAX} + N_F\ T_{eq}]$$

T_{AMAX} is the maximum time allowed for acceptance testing per axis for which qualification is to be achieved. A recommended value is 360 seconds, i.e., six one minute acceptance tests.

N_F is the number of flights for which qualification is to be achieved.

An example is given in 6.1.4.1of VOL I. Another example in which only the test tolerance T is reduced: M = 6 dB and T = 1.5 dB yielding α = 15 (Table VII), T_{AMAX} = 360 seconds, T_{eq} = 15 seconds, N_F = 1 yields

$$T_Q = 4[(1/15)(360) + (1)(15) = 156 \text{ seconds}$$

5.3.3.2.2 **Two-condition Testing**. Another qualification approach is called two-condition testing (Paragraph 6.1.4.2 of VOL I). The first condition qualifies for the maximum allowable amount of acceptance testing for the particular unit. The acceptance spectrum is applied for four times the predetermined maximum allowable duration of acceptance testing, performed in all three axes before proceeding to the second condition which addresses the flight environment. As opposed to the simpler test practice of concluding all testing in one axis before proceeding to another axis, this sequence meets the intent of qualifying for the flight environment after acceptance testing has been experienced. The test duration factor of four is consistent with structural test practice for a fatigue-life factor. For the second condition, the P99/90 level is applied for one minute in each axis or, if longer, the fatigue equivalent duration in flight. If an item is recovered and reflown, the foregoing time at the P99/90 level is multiplied by the number of flights for which qualification is to be achieved.

The fatigue equivalent duration of a flight, T_{eq}, is the time at the maximax spectrum that is the fatigue equivalent of the nonstationary environment in flight. It is assumed that the root-mean-square (rms) velocity is the indicator of stressing for random vibration. Equating the test and flight fatigue potential yields

$$T_{eq} V_m^p = \Sigma_i T_i V_i^p$$

T_i is the length of the ith time segment used for spectral analysis, typically one second.

V_i and V_m are the root-mean-square velocities for the ith time segment and for the maximax spectrum, respectively, the latter being the envelope of the spectra for each time segment. In equation form

$$V^2 = g^2 \Sigma_j G_j Df_j / 4\pi^2 f_{cj}^2$$

where G_j, Df_j, f_{cj} are the acceleration spectral density (g^2/Hz), bandwidth (Hz), center frequency (Hz), respectively, of the jth frequency band and g is the acceleration of gravity in units consistent with the derived velocity units.

pis the fatigue exponent, the exponent of stress (S) for the assumed linear relation of log stress amplitude versus log number of cycles (N) to failure. In equation form

$$S^p N = \text{constant}$$

The exponent p is taken to be four for conservatism unless a different value can be justified for the particular unit and its critical failure mode.

5.3.3.3 Acceptance Test Limits. The random vibration spectrum for acceptance tests is based on the maximum expected vibration (P95/50) subject to a minimum spectrum, as discussed in paragraph 5.3.2.1 here and in paragraph 7.1.3 of VOL I. The minimum spectra in VOL I Figure 5 for units and Figure 6 for vehicles are lower bounds to assure an adequate workmanship screen. The minimum duration of one minute per axis generally allows for adequate functional evaluation during the applied vibration, as well as being a minimum from a workmanship screen standpoint. The test duration should be lengthened if demanded by either of these aspects. An upper bound on the accumulated duration of acceptance testing per axis, due to retesting, must be established in advance of qualification so that the fatigue potential is properly taken into account by the qualification testing (Paragraph 5.3.3.2).

5.3.3.4 Force or Response Limiting. Vibration testing inevitably involves unnatural constraints on the state of vibration experienced in flight. Most testing, for example, strives to achieve equal input motions at all attachment points for one direction of input. Another is the use of envelope spectra of flight data, bridging over spectral dips. Such constraints can lead to excessively conservative test levels (although one unconservative aspect is the one-axis-at-a-time testing). Force or response limiting refers to the practice of notching (reduction of level in frequency bands) of the input acceleration spectrum to a test item to reduce either the applied force spectrum or to reduce the magnitude of the spectrum of test item response at critical locations. In both cases the reduction is in frequency bands which contain major resonant behavior of the test item. A justifiable basis for such limiting is necessary in order to avoid excessive reduction of inputs that will result in inadequate acceptance or qualification.

Force limiting requires that three-axis force transducers be positioned at each attachment point during the test and that an upper-bound net force spectrum be developed and imposed for each direction of translation, and perhaps also net moments about rotational axes. The concept is that the input motion is reduced because a relatively high mechanical impedance of the test item inhibits the motion of the supporting structure that would occur in the absence of the test item being in place, as well as the fact that test specifications are based on enveloping nulls in environmental spectra which may be due to such impedance interaction. The establishment of the limiting force spectra requires a mathematical model of the interaction and predicted or test derived impedance data. This is an evolving art and must be cautiously practiced.

Response limiting requires that positions of large response on the test item be instrumented during test and that a response limit at those positions be imposed during the test. In this way analytical or test information for the response positions can be used to limit input to the test item.

5.3.3.5 **Isolated Units**. Dynamic isolators often will exhibit significant variability in resonant frequency (f_n) and resonant amplification factor (Q). The required unit design margins may not be met during qualification testing, due to this variability. Therefore, to preserve the design margins, the variability must be known and controlled. In addition, since the purpose of a dynamic isolator is to attenuate the input to the unit above the isolator resonant frequency, it is likely that the unit response will fall below that level necessary for proper environmental stress screening for manufacturing defects. Therefore, the development of test specifications for isolator mounted units requires careful planning.

5.3.3.5.1 **Isolator Performance Requirements**. A source control drawing (SCD) should specify the desired characteristics of the isolators. A copy of the isolator drawing from the manufacturer's catalog would be an adequate pictorial representation of the desired isolator. The allowable variation of resonant amplification factor (Q) and isolator resonant frequencies with specified supported mass about the nominal values should be stated, in the three principle axes, for example on a source control drawing (SCD). This performance variation should address influences of combined environments, including but not limited to, vibration, temperature, acceleration and chemical exposure. Paragraph 7.4.4. VOL I of the Handbook provides supplementary lot acceptance test requirements for isolators.

It is recommended that isolator mounting system design verification and acceptance testing be performed using either a unit or simulator (mass and center of gravity) to verify system performance. Parameters of interest are system resonance in three orthogonal axes and fatigue life. If responses at all corners of the unit are obtained, they may be used to determine the unit hard-mounted acceptance test levels.

It is recommended that all isolators be acceptance tested in at least one principle axis to verify that Q and resonant frequency are within the specification limits. Either random or sine vibration may be used, however, the input should be equivalent to the flight vibration level.

5.3.3.5.2 **Vibration Testing of Isolated Units**. Paragraph 7.4.4.2 of VOL I of the Handbook states that, "Units mounted on shock or vibration isolators shall normally be tested hardmounted to assure the minimum spectrum shown in Figure 5 is input to the test item." The acceptance test level for hard-mounted testing should envelope responses due to isolator resonances and the spectrum of Figure 5. The acceptance test level should be increased by a factor based on the allowable variation of Q for the lot of isolators, as shown below.

Allowable Variation of Q from Source Contral Drawing (SCD)	dB Increase Nominal Q<5	Nominal Q>5
Less than 5%	0.5	1.0
5% to 10%	1.5	2.0
10% to 15%	2.0	2.5
15% to 20%	2.5	3.0

Paragraph 6.4.4.2 of VOL I of the Handbook specifies that, *"Units mounted on shock or vibration isolators shall typically require vibration testing at qualification levels in two configurations. A first configuration is with the unit hard-mounted to qualify for the acceptance-level testing if, as is typical, the acceptance testing is performed without isolators present. The second configuration is with the unit mounted on the isolators to qualify for the flight environment. The unit shall be mounted on isolators of the same lot as those used in service, if practicable. Units mounted on isolators shall be controlled at the locations where the isolators are attached to the structure. Hard-mounted units shall be controlled at the unit mounting attachments."* The qualification test levels should show an appropriate margin over the acceptance test levels, including the environmental stress screening requirement.

5.3.3.6 Sine vs. Random. For qualification and acceptance testing the determination of whether a particular vibration test is sinusoidal, or random or a combination of the two should always be made on the basis of providing the best simulation of the service environment. Except for equipment mounted in close proximity to rotating machinery such as turbo pumps, most flight vibration environments tend to be random. Consequently, a large majority of acceptance and qualification vibration testing will also be random. Industry experience has shown that it is inadvisable to substitute one of these forms of vibration for the other. This is because qualification and acceptance testing have the common objective of preventing failures in flight. Conducting a sinusoidal vibration test to simulate a random flight environment can cause unrealistic ground test failures such as might be caused by excitation of high amplification resonances that would not occur in a flight random vibration environment. Conversely a sinusoidal test may not identify other failure mechanisms that may only be associated with a random environment such as those associated with the effects of simultaneous excitation of multiple resonances. In some cases the flight environment may be a combination of sinusoidal and random excitations in which case combined sinusoidal and random testing should be performed.

For diagnostic testing the form of excitation is left to the discretion of the design and test engineers. A design engineer may prefer sinusoidal excitation in order to more easily identify individual resonances and the test engineer's preferences may be based on available laboratory equipment, be it sine or random.

5.3.3.6.1 Isolator Performance Requirements. Isolator performance may exhibit variations due to manufacturing processes and/or variations resulting from conditions of usage, including but not limited to, vibration, temperature, acceleration and chemical exposure. Tests should be conducted to verify that isolator performance is predictable. These tests should include measurements of both individual and system isolator performance. Isolator performance characteristics are typically found in the manufacturers source control drawing (SCD). A copy of the isolator drawing from the manufacturer's catalog would be an adequate pictorial representation of the desired isolator. The allowable variation of resonant amplification factor (Q) and isolator resonant frequencies about the nominal values for the conditions of usage should be stated, in the three principal axes on the SCD. Paragraph 7.4.4.5 of VOL I of the Handbook provides supplementary lot acceptance test requirements for isolators

5.3.3.7 Fixture Evaluation, Test Control and Tolerances. Paragraphs 6.4.4.5 and 7.4.4.4 of VOL I of the Handbook address the need for fixture evaluation prior to unit vibration testing. The issue is not directly addressed for system or subsystem vibration testing as this case does not often occur, but should be strongly considered. Experience has shown that, on occasion, inadvertent damage to a unit under test has occurred from fixture resonances that were undetected. The damage would have been avoided had a fixture evaluation been performed prior to start of testing. Experience has also shown, that even in the simplest fixture, repeated use can cause structural cracks or loosening of bolts which can introduce higher than planned input to the unit under test. The goal is to minimize unit exposure to this possibility of damage by performing fixture evaluation as a minimum before qualification testing and periodically for unit acceptance testing.

Maximum allowable test tolerances for vibration testing are contained in Table III VOL I of the Handbook. Changes from the previous edition include provisions that allow the use of both analog and digital controller capability. The maximum control bandwidth is meant to be a maximum allowable with the commensurate tolerances shown. Narrower bandwidths may be used, however, provision for different tolerance levels will be needed and are not precluded. Above 1000 Hertz, the tolerance bands have been retained as +/- 3 dB in recognition of problems in a frequency region where shaker armature and fixture responses make control difficult.

5.3.3.8 Unit vs. Subsystem vs. Vehicle. With few exceptions vibration testing is required and most effectively performed at the unit level. When a vehicle is small, without significant surface areas that can respond to an acoustic field, vibration at the system level of assembly may also be performed. Care should be taken that appropriate boundary conditions are maintained so that overtest does not result.

5.3.3.9 Operating vs. Non-Operating. At the unit level, all vibration testing should be performed with the unit powered and functioning. Experience has shown that this is the most perceptive method of detecting functional problems during exposure to a dynamic environment. It is recognized that time may not permit all unit functions to be performed during the acceptance or qualification duration period. Provision has, therefore, been made to reduce the test level and continue functional testing.

Motion sensitive guidance and control units such as inertial measurement units that perform their intended function during launch and ascent must be tested in a powered and operational state. Operational performance limits are required to be met during testing at acceptance levels, but may be exceeded at qualification test levels.

5.4 ACOUSTIC TESTS

5.4.1 Standard Criteria for Acoustic Tests. Definitions related to the acoustic environment are presented in paragraphs 3.3.2, 3.3.3, and 3.3.4 of VOL I of the Handbook. Acoustic test requirements apply to systems, subsystems, and units for development, qualification and acceptance testing as described in paragraphs 5.5.4, 6.1.4, 6.2.4, 6.3.3, 6.4.5, 7.1.2, 7.2.4, and 7.4.5. Table's III, IV, VII, and XI and Figure 4 of VOL I of the Handbook provide tolerance and summary criteria for acoustic testing.

5.4.2 Rationale for Acoustics Tests. The acoustic environment during liftoff and ascent represents a significant forcing function for structural responses at frequencies above 50 Hertz. The relative contributions of the forcing function producing these vibration responses are dependent upon the launch vehicle, the upper stage and/or space vehicle configuration and the particular location of interest. The dominant acoustic forcing function and the resultant structural vibration response for launch vehicles may occur either during the relatively short liftoff event or the longer period of ascent encompassed by transition from subsonic to supersonic flight through the period of maximum dynamic pressure. The dominant acoustic forcing function and the resultant structural vibration response for upper stages and space vehicles, may occur over the same period, but is modified by the noise transmission performance afforded by the payload fairing. Vibration requirements for units on upper stages and space vehicles, therefore, are nearly always linked directly to the acoustic environment to which the vehicle is exposed. The acoustic time history is non-stationary and its amplitude dependent upon the ground-reflected acoustic energy emanating from the exhaust flow of the propulsion system during liftoff and the fluctuating pressure field during flight. Acoustic environments during these time periods can have large spatial variations. Consequently, acoustic design criteria for space vehicles are sometimes defined by zones. More commonly, however, a single criterion is defined which represents the maximum environment in one-third octave bands to which any vehicle surface is expected to be exposed. The goal is to define the extreme and maximum level in statistical terms as discussed in paragraphs 3.3.4 of VOL I of the Handbook and 5.4.3.1 of VOL I. Seldom, however, does sufficient data exist to allow performance of rigorous statistical analysis. Nevertheless, the extreme and maximum expected acoustic environment is usually developed considering variations such as different launch pads, different trajectories, spatial variations within the launch vehicle payload compartment, and in some cases different launch vehicles.

5.4.2.1 Technical Basis - Environmental Stress Screening. Exposure of units, subsystems and systems to an environment represents the principle demonstration of the item's ability to function during and after exposure to a simulation of the flight environment. When the flight environment is benign, demonstration of functionality is achieved by performing a test using a minimum environment sufficient to detect workmanship defects. The chosen level and/or duration represents a test that serves as an “environmental stress screen” (ESS). An environmental stress screen for a dynamic environment should consist of an amplitude and duration sufficient to result in structural responses that apply stresses to mechanical and electrical interfaces, subject units to vibration responses high enough to detect latent defects and verify the integrity of the item dynamic characteristics prior to use. In many instances, the acoustic test environment is that associated with a simulation of the flight environment. There are however, cases where the acoustic environment is relatively benign and therefore, environmental stress screening amplitudes and duration’s are in excess of those experienced during flight. The selection of the minimum (or ESS) acoustic test level given in VOL I of the Handbook was based upon review of the range of actual acoustic test levels and the associated vibration responses of flight hardware. The spectrum in

Figure 4 of VOL I of the Handbook is a minimum acoustic level that will provide a vibration response sufficient to detect workmanship defects in typical equipment.

5.4.3 Guidance for Acoustics.

5.4.3.1 Statistical Basis. Statistical estimates for an acoustic environment are made identically to those made for a vibration environment. Paragraph 5.3.3.1 is applicable, only requiring that the level x be the sound pressure level (SPL) typically expressed in dB.

5.4.3.2 Qualification Testing. Same as 5.3.3.2.

5.4.3.2.1 Accelerated Testing. Same as 5.3.3.2.1.

5.4.3.2.2 Two-condition Testing. Same as paragraph 5.3.3.2.2, with the exception that rms pressure is used as the indicator of stressing for determining the fatigue equivalent duration of a flight. Therefore

$$T_{eq}\, P_m^{\,p} = \Sigma_i\, T_i\, P_i^{\,p}$$

where P_i and P_m are the root-mean-square pressures for the ith time segment and for the maximax spectrum. That is, the sound pressure levels in dB (SPL) are converted to rms pressures as follows:

$$P = P_{ref}\, 10^{SPL/20}$$

$$P_{ref} \text{ is } 2 \times 10^{-5}\ \text{N/m}^2 \text{ or } 2.9 \times 10^{-9}\ \text{lb/in}^2$$

where P_{ref} is 20 micropascals, the reference pressure basis for expressing the SPL in dB. The exponent p is the fatigue exponent described in 5.3.3.2.2.

5.4.3.3 Acceptance Test Limits. The acoustic spectrum for acceptance tests is based on the maximum expected acoustic level (P95/50) subject to a minimum spectrum, as discussed in paragraph 5.3.2.1 here and in 7.1.2 of VOL I. The minimum free-field spectrum in Figure 4 is a lower bound to assure an adequate workmanship screen. The minimum duration of one minute generally allows for adequate functional evaluation during the applied acoustic field, as well as being a minimum from a workmanship screen standpoint. The test duration should be lengthened if demanded by either of these aspects. An upper bound on the accumulated duration of acceptance testing, due to retesting, must be established in advance of qualification so that the fatigue potential is properly taken into account by the qualification testing (Paragraph 5.3.3.2). A baseline bound of three test durations has been used as a typical value in VOL I and is open to change on a case-by-case basis.

5.4.3.4 Fixtures, Test Control and Tolerances. The vehicle, subsystem, or unit shall be mounted on a flight-type support structure or reasonable simulation thereof for an acoustic test. Flight-type structure is important so that acoustic energy is converted to a vibration response and transmitted to the article under test in a manner similar to that experienced in flight. Poor simulation of boundary conditions may adversely affect the response of those units on the test article that are mounted nearest the interface. The degree of simulation should be sufficient to replicate the first several modes of the support structure. Mounting of the test article in an acoustic chamber must consider the choice of attachment to the floor versus supporting the test article by a suspension system. Such issues as higher acoustic energy near the floor due to reflections and the effect of free-free boundary conditions upon test results must be addressed. It is generally recommended that support adapters be bolted to the chamber floor provided the floor response is known to be rigid. Otherwise, an isolator pad is recommended. When the support adapter is of open construction, consideration should be given to the avoidance of high local reflective energy building up beneath the test article.

Control of the acoustic test is provided by a minimum of 4 control microphones, placed as described in paragraph 6.2.4.2 of VOL I of the Handbook. A minimum of 4 microphones was chosen to account for the large size of current test articles and to achieve a reasonable control of energy surrounding the test article. A requirement to place control microphones no closer than 0.5 meters to the nearest chamber wall and test article is in recognition that pressure doubling effects can occur at locations near reflective surfaces. Test control shall be in terms of 1/3 octave band frequencies having center frequencies ranging from 31.5 to 10,000 Hertz. Control up to the 1/3 octave band centered at 10,000 Hertz is recommended. The energy above 2000 Hertz is generally considered to be of lower damage potential, however, it provides added environmental stress screening at frequencies above those imposed during a vibration test and is achieved at small added cost.

Test control tolerances of +/- 5 dB are retained at the 1/3 octave bands centered at 31.5 and 40 Hertz due to limited acoustic modes in typical test chambers. The test tolerances from the 1/3 octave bands centered at 2500 to 10,000 Hertz have been increased from +/-3.0 dB to +/- 5.0 dB to reflect the actual experience of typical chambers.

5.4.3.5 Unit vs. Subsystem vs. Vehicle. Acoustic testing is usually performed at the vehicle level. However, when the vehicle is too large to fit into a chamber, its major subsystems can be tested separately. This is particularly true for launch vehicles where size is a factor and where only selected portions of the vehicle such as the guidance equipment compartment would be suitable for testing acoustically. Subsystems with large surfaces such as solar arrays and antennas are often separately tested. Care should be taken that appropriate boundary conditions are maintained so that significant structural-borne energy paths are not overlooked, which can result in under-test.

Acoustic testing of units is seldom necessary, with the possible exception of items having large surfaces. Such a large surface would be expected to respond to the incident acoustic energy and generate vibration responses within the item that equal or exceed the responses that might be imposed at the mounting points.

5.4.3.6 Operating vs. Non-Operating. Paragraph 6.2.4.4 of VOL I of the Handbook requires that during the vehicle qualification acoustic test, all electrical and electronic units, even if not operating during launch, shall be electrically energized and sequenced through operational modes to the maximum extent possible. For acceptance testing, however, the requirement is limited to only equipment operating during launch, ascent, or re-entry. Experience has shown that few programs perform credible functional testing during the acoustic exposure, especially for units not normally energized during ascent. The expense of transporting, and setting up test support hardware, the short acoustic exposure time that limits time to exercise multiple functional paths, limitations of telemetry transmission rates, and thermal constraints placed on units normally powered off during ascent are factors that may reduce the amount of power-on testing.

Acoustic acceptance test results for eight major Space and Missile Systems Center (SMC) programs composed of 75 total space vehicles and contractor provided data for eight programs (81 space vehicles) were reviewed in 1988. In all but one case, all equipment operating during ascent plus additional components and sensors was “powered-on." These data indicated that two out of a total of 118 failures could have only been revealed with “power-on” during the acoustic acceptance test. Neither failure was apparent during the vehicle post-test functional testing. One of the failures would have been catastrophic to the mission. The equipment involved in both failures would normally have been “powered-on” during launch and ascent. This data prompted a reduction in the “power-on” requirement for acceptance testing. However, for qualification testing the requirement for powering all equipment has not been relaxed.

Acoustic testing at the subsystem or unit level may overcome many of the issues that make “power-on” testing at the vehicle level so difficult. In that case, it is recommended that full “power-on” testing be performed when units or subsystems are acoustic tested.

5.5 SHOCK TESTS

5.5.1 **Standard Criteria for Shock Tests**. Definitions related to the shock environment are presented in paragraphs 3.3.2 and 3.3.7 of VOL I of the Handbook. Shock requirements apply to systems, subsystems, and units for development, qualification and acceptance testing as described in paragraphs 5.4.3, 5.5.4, 5.5.6, 6.2.3, 6.3.5, 6.4.6, 7.2.3, and 7.4.6. Tables III, IV and XI of VOL I of the Handbook provide tolerance and summary criteria for shock testing.

5.5.2 **Rationale for Shock Tests.** The sudden application or release of energy attendant to separations or deployments associated with aerospace mission events can generate brief impulsive loads. Examples of such events include staging, fairing separation, solar panel deployment or boom release. When these events are initiated by the activation of explosive ordnance devices they are commonly called pyroshocks. The shock environment is typified by high frequency content (generally 1000 Hz and above), high shock response amplitudes (in the thousands of g's) and brief duration (20 milliseconds or less). Not included in this category are phenomena referred to as transient loads. Such events as those generated by ignition and shutdown of rocket engines generate loads which are of considerable importance to primary and secondary structure, but due to their low frequency characteristics are of less concern to units. Shock has been frequently underestimated in terms of its damage potential. The absence of significant energy at frequencies traditionally of concern to aerospace hardware, and the contention that random vibration testing can demonstrate resistance to shock have led to inadequate shock testing in the past with disastrous consequences. The high frequency energy characteristic of pyroshock makes it difficult to analyze and design for VOL I of the Handbook, therefore, prescribes development tests intended to aid in environment definition, qualification requirements at the unit through system level of assembly to assure robust designs, and acceptance test requirements where appreciable levels of shock will be encountered.

5.5.3 **Guidance for Shock.** Accurate prediction of high frequency shock levels, such as those associated with explosive ordnance-initiated events, remains an unachieved goal. Therefore, it is important that the shock environment be assessed during the development phase of the program through test simulations. These tests should evaluate all significant shock events with sufficient fidelity to provide valid unit shock criteria. Timing of redundant ordnance initiation signals should comply with that planned for flight. Shock tests should employ an accurate replica of the flight structure with all significant units simulated. Elements intended to be deployed should be permitted to physically separate at least a minimum amount to provide realistic shock transmission paths. Increased effort expended in simulating the flight configuration will yield more accurate results. If practical, a given shock-producing event should be repeated an adequate number of times to permit meaningful statistical evaluation of resulting data. In order to assure quality results, careful consideration must be paid to the shock data acquisition system. The challenges of measuring the high amplitude short duration events are significant. The reader is advised to consult technical reference 5 to enhance the prospects for a successful shock measurement program. The development phase is also the appropriate time to conduct evaluation tests and characterizations of dynamic isolators that may be planned to protect selected units from the shock and vibration environments. The maximum and extreme expected shock levels are applicable to acceptance and qualification of units. These levels, generally determined from data acquired from the development tests represent the 95 percent probability level with 50 percent confidence and the 99 percent probability level with 90 percent confidence, respectively. The test fixtures and procedures employed in the system development testing are frequently applied for system-level qualification and acceptance testing.

5.5.3.1 Statistical Basis. Statistical estimates for a shock environment are made identically to those made for a vibration environment. Paragraph 5.3.3.1 is applicable, only requiring that the level x be the shock spectrum level.

5.5.3.2 Qualification Margin. At the system level of assembly it is generally impractical to impose an amplitude qualification margin. Instead, the qualification requirement involves repeated application of significant shock events. Significant shock-producing events are to be imposed three times by activation of separation and deployment devices. It is intended that only those shocks that represent the controlling shock environment for equipment be applied. For example, if a given shock-producing deployment is exceeded by at least 6 dB at all frequencies at all equipment locations by another shock event, only the higher shock-producing event need be applied. System testing does not lend itself to achieving amplitude margins. On the other hand system test provides a realistic application of the environment.

VOL I of the Handbook prescribes that the unit level qualification test be performed at the extreme expected, or 99 percent probability level with 90 percent confidence. This level is roughly 6 dB above the maximum expected level (defined as the 95 percent probability with 50 percent confidence) applicable to acceptance testing. In addition the shock environment is to be imposed three times the number of exposures expected in service. If the selected shock test procedure achieves the required shock spectrum in both directions of each of the orthogonal axes in one application, that shock should be applied three times. If, on the other hand, the applied shock only meets the specified level in one direction of one axis, the number of repetitions becomes 18 to achieve the required level three times in each of the six directions. Between these two extreme cases lay a number of other possible scenarios. The amplitude and exposure margins are intended to demonstrate that the design is sufficiently robust to allow for a degree of variability in each ensuing production unit and to demonstrate a basis for flight unit acceptance testing, where applicable.

5.5.3.3 Test Methods. System-level shock tests usually involve activation of separation or deployment systems leading to a direct simulation of the mission event. To that end, test procedures should be designed to assure that the test realistically preserves the important features of the shock environment. Elements that separate or deploy should be allowed to physically separate by a finite amount. Test fixtures are frequently needed to provide support to hardware that has been released so that subsequent damage or recontact is avoided. Flight preloads and sequences for initiation of separation signals should be used. System-level shock tests provide an excellent opportunity to measure the shock environment incident on units throughout the vehicle. The nature of the shock environment requires that the measurement system span a broad frequency range from 100 to 10,000 Hz, acceleration amplitudes in the thousands of g's, and duration in the range of 20 milliseconds. These features of the environment make the acquisition of flight shock data extremely difficult to facilitate and justify. Ground test simulations, therefore, offer the best opportunity to acquire this important data.

The requirement for unit shock testing prescribes that a specified shock spectrum be met with a transient event whose duration generally conforms with that of the service environment. Unit-level shock test methods are frequently selected based upon the amplitudes required and the degree of realism sought. Electromagnetic vibration exciters are sometimes used to impart shocks to modest-sized units with peak shock response spectrum amplitudes less than 3500 g's. The technique involves generation of a pulse which is passed through an array of shaping filters to result in a shock input. The resulting shock spectrum may be shaped by adjusting the gain of the appropriate filters. A deficiency is the inability to provide realistic spectral input at frequencies above the armature resonance of the vibration exciter. Aside from the amplitude and frequency limitations, the shock environment generated in this fashion is frequently more damaging than the service shock, because the pulse is imparted via a rigid fixture with frequency phasing and uniformity which do not simulate typical pyroshock characteristics. Another method involves mounting the test unit on a structural element, which is struck by an impacting device or shocked by activation of an attached explosive ordnance device. Such test devices are capable of more closely simulating the service pyroshock environment. They cover a broad range of individual laboratory applications. Examples include; a metal plate which is struck by a driven mass whose velocity, weight and surface characteristics can be varied in combination with orientation and distance for the unit undergoing test; a similar plate where the shock is imparted by a length of mild detonating fuse attached to the plate; or a segment of the actual vehicle structure where a shock is imparted by an impacting mass. There are many variations of such test setups. They frequently require an extensive pretest calibration phase, using a simulator for the test specimen, wherein details of the test variables are determined by trial and error.

5.5.3.4 Fixtures, Test Control and Tolerances. System-level shock test fixtures generally consist of devices used to support separated or deployed elements to safeguard them from damage and prevent recontact. Examples include devices to catch and support a skirt which represents a mating stage in a staging test, or a holding device to support a solar array boom after its initial movement in a deployment test. Test control in system test is accomplished by direct simulation of the parameters associated with the generation and transmission of the shock environment. Mostly, this entails the use of flight separation hardware, flight structure and activation of separation sequences in accordance with mission plans. Since system shock tests are configuration controlled, there are usually no tolerances as such. However, the measured shock amplitudes should show minimal variability for repeated activations of a given separation event. It is expected that the variation from the average shock spectrum to the maximum levels observed would be 5dB or less. A larger variation could indicate that some feature of the shock test is not adequately controlled.

Shock fixtures for unit testing are dependent on the selected shock test method. If an electrodynamic exciter is used for shock testing, the fixture is generally similar to one used for vibration testing i.e., a very stiff holding device designed to transmit the shock impulse from the exciter with minimum variability at mounting points and cross axis motion. For shock test facilities which use a resonant plate, the test specimen is mounted directly to the plate through a simple adapter fashioned to accommodate individual mounting provisions of the test unit. The shock test fixture may also replicate the vehicle structure from the shock source to the test specimen, complete with intervening units, joints and interfaces. Test control methods are also dependent on the shock test technique selected. For shock tests using an electrodynamic exciter the shock incident on the test specimen is analyzed during calibration runs and fed back to the filters shaping the shock spectrum. The settings are adjusted manually or automatically to impose the specified shock environment. Control authority is limited by the force rating of the exciter and frequency range of the filters. When a resonant plate shock facility is used, the impulse imparted to the plate can be varied by control of a number of parameters. These include control of the impactor velocity and mass, cushioning at the impact site, and location and orientation of the test specimen. If the shock is generated by ignition of an explosive ordnance device, control may be exercised by the amount of explosive material, placement of the explosive material, location and orientation of the specimen or by characteristics of the plate itself. Some explosive ordnance based facilities use materials such as linear shaped charges to cut a thickness of metal to generate the shock impulse. For such devices, the characteristics of the material being severed in addition to the explosive material will influence the shock generated. Regardless of the test technique, unit shock test levels are expressed as a shock spectrum (analyzed at 1/6th octave intervals or less and resonant amplification factor ,Q, of ten). The levels should lie within ± 6dB below 3000 Hz and +9/-6dB above 3000 Hz of the specified requirement. In addition, at least 50 percent of the spectrum values must exceed the specified shock spectrum. These tolerances are quite broad in recognition of the difficulties inherent in many of the current shock test facilities; especially in the higher frequency range, where an additional 3dB of tolerance in the upper band was added relative to MIL-STD-1540B. On the other hand, VOL I of the Handbook has tightened shock tolerances by requiring that 50 rather than 30 percent of the spectrum values exceed the required spectrum. This feature of the tolerance specification compensates for the fact that shock spectrum does not have a concept of overall value, such as acoustic or vibration spectra. For acoustics or vibration the desired goal of centering the test exposure in the tolerance band is accomplished by imposing a tight tolerance on the overall value. The 50 percent requirement for shock avoids the situation where most of the applied spectrum falls below the requirement.

5.5.3.5 Requirement Satisfaction by Random Vibration. There are occasions when the demonstration of shock resistance can be satisfied by random vibration testing. This demonstration is accomplished by comparing the peak response of a single degree of freedom resonator to the random vibration environment with the shock response requirement. For random vibration the peak response is a function of acceleration spectral density at the resonator natural frequency, the resonator natural frequency, and the system amplification factor (transmissibility). For electronic equipment the transmissibility is assumed to lie between 10 and 20, unless otherwise known. In any event, consistent transmissibility values should be used for both vibration and shock. The equation to perform the calculation is:

$$G_{pk} = 3[(\pi/2)(GQf_n)]^{1/2}$$

G_{pk} = 3 sigma peak response as a multiple of 32.2 ft/sec^2

G = acceleration spectral density, (g^2/Hz)

Q = transmissibility = 1/2ζ

ζ = damping (% of critical)

f_n = natural frequency, (Hertz)

An example of peak response is given in the following Table for the minimum random vibration unit acceptance test requirement from Figure 5 of VOL I of the Handbook.

Frequency	Random Vibration(g^2/Hz)	G_{pk} Equivalent Shock, Q=10	G_{pk} Equivalent Shock, Q=20
20	.0053	3.9	5.5
150	.04	29.1	41.1
600	.04	58.2	82.4
2000	.0036	31.9	45.1

Note that the resulting equivalent shock level is rather low when compared to typical unit shock levels (>1000 g's peak) normally specified. However, there can be instances when a high random vibration and a low shock specification occur where the method is appropriate.

5.5.3.6 Damage Potential. Potentially damaging shocks may result from the sudden release of a load or stored energy, the impact of separation hardware, an explosive detonation or a combination of the foregoing. The shock event, characterized by high acceleration levels and brief duration may cause damage to equipment with sensitivity at high frequencies. Most failures due to shock occur in electrical units and small, brittle devices. Structural elements are seldom affected. Experience from flight and ground test has provided some insight into potential damage sites in aerospace equipment. At relatively low shock acceleration levels (shock spectrum values in the hundreds of g's) failures have been observed due to the liberation of minute contaminants in piece parts. If these contaminants are electrical conductors and bridge electrical elements to cause a short of even brief duration, serious consequences may result. This class of failure has been rendered far less likely by protective coatings over electrical elements in piece parts, and improved screening for internal contaminants. The next category involves units with movable devices. Devices, such as relays and valves, have been observed to chatter, and at times, in the case of relays, to transfer when subjected to shock. Such malfunctions may occur at shock acceleration amplitudes below those where permanent damage is anticipated. At higher levels shock may cause permanent damage to equipment, especially electrical units. Pyroshocks with their high acceleration level at high frequencies can crack glass and ceramic elements, or cause parts bonded with brittle epoxies to come loose. This latter failure mode provides part of the rationale for recommending that shock testing precede vibration. Elements which have been loosened by shock may be observed more thoroughly during an operating vibration test, or the vibration test may cause electrical leads of a part loosened by shock to fail. Past experience has shown that the high frequency shock environment does not pose a hazard to typical aerospace equipment at shock spectrum amplitudes (with Q=10) below 0.8 times the frequency in Hz. Accordingly, VOL I of the Handbook does not require shock qualification testing of units for which the maximum expected shock spectrum at frequencies above 2000 Hz does not exceed a value of 0.8 times the frequency in Hz; as long as the qualification random vibration test spectrum, when converted to an equivalent 3 sigma response, exceeds the qualification shock spectrum at frequencies below 2000 Hz. On the other hand, VOL I of the Handbook imposes a requirement that shock tests be performed as part of the acceptance testing of electrical and optical units where the maximum expected shock spectrum exceeds a line drawn at 1.6 times frequency in Hz. In such instances the shock is judged to be severe thereby warranting its inclusion in the acceptance program.

When shock acceptance testing is applied to units, consideration must be given to potential fatigue damage to flight equipment. On an axis-by-axis basis it must be demonstrated that acceptance exposure consumes no more than one fourth of the demonstrated dynamic design capability. Using methodology which parallels that described in paragraph 5.3.3.2.1 for vibration, the allowable number of acceptance shock applications is equivalent to one fourth of the demonstrated design capability. The computation should consider the three to one ratio of repetitions for qualification and one acceptance exposure and the effect of the qualification test margin. Where the qualification margin is 6 dB the following number of acceptance tests would be permitted:

$$n = N\frac{(Q/A)^p}{4} = 3\frac{(2)^4}{4} = 12$$

Where: n is the allowable number of acceptance exposures
N is the number of qualification exposures
p is the fatigue exponent
Q/A is the ratio of the qualification to acceptance amplitude.

5.5.3.7 **Unit vs. Subsystem vs. Vehicle**. In contrast to vehicle level testing, shock testing of units affords an opportunity to incorporate qualification margin in the environmental test levels, and more thoroughly monitor functional performance. The levels imposed may be controlled at the maximum or the extreme expected amplitudes, depending on the test objectives. The test specification for units is generally a smooth spectrum envelope based upon data summaries or analyses. The generation of such specifications may incorporate some degree of conservatism at certain frequencies. In addition, depending upon the test facility selected, the test environment imposed can involve features of coherence or phasing, which may tend to be inherently more damaging than the actual shock environment. These inherent qualities of the test method should be considered in selection of the test approach. In contrast, vehicle level shock testing usually provides an environmental exposure with excellent simulation qualities, but very little opportunity to incorporate amplitude margins above a nominal level of severity, or to monitor functional performance of individual units. The inability to incorporate amplitude margins is compensated in part by requiring that significant shock events at the vehicle level of assembly be performed three times for qualification. This repetition is intended to afford some degree of amplitude variability in an environment where experience has shown the maximum expected level (P95/50) exceeds the average value by approximately 5dB. Shock testing of subsystems may follow the methods used for unit or vehicle testing, depending on the extent of the subsystem and particular test objectives. The subsystem could involve a number of related units which are grouped such that they are exposed to roughly the same mission shock environment. Further, it may enhance test effectiveness to assess the ability of the subsystem to operate during and after the shock event. In this case, assuming weight and size permit, the subsystem could be subjected to a shock test in the same fashion as one would test an individual unit. The more general subsystem-level test typically involves a segment of the vehicle, such as a major space vehicle payload, which is tested separate from the vehicle for logistical reasons. In this case the test method would follow the vehicle test approach. The relationship of the test rigor with level of assembly is important to bear in mind. The conservatism inherent in the unit test provides assurance that the units will not fail during vehicle shock testing. The vehicle shock tests are intended to test the assembly and integration hardware, as well as devices which do not lend themselves to be tested as units. Discovery of unit shock problems during vehicle testing creates serious implications to the program in terms of timeliness and recovery difficulty.

5.5.3.8 Operating vs. Non-Operating. Units and subsystems undergoing shock test should be in their electrically operational state and monitored for evidence of failure or malfunction. In addition to functional performance checks conducted before and after shock exposure, monitored operation is extremely important in enhancing the effectiveness of shock testing. Evidence of failure, such as liberated contaminants or parts broken loose by the shock may not reveal themselves under bench operation. For the same reason shock testing should precede vibration or acoustic testing to permit functional performance checks to be performed during longer duration dynamic excitation. The requirement for monitored functional performance is imposed regardless of the operational condition which will exist when the mission shock environment is encountered. Its purpose is to significantly improve test effectiveness by increasing the likelihood of detecting anomalous behavior.

5.6 THERMAL CYCLING TESTS

5.6.1 Standard Criteria for Thermal Cycling Tests. Definition of terms applicable to the thermal environment in general are contained in paragraphs 3.3.1, 3.5.7, and 3.5.10 of Vol I of the Handbook. Paragraphs 6.1.3 and 7.1.1 set forth the philosophy of thermal cycling for qualification and acceptance respectively. They are complemented by Tables V and VI and Figures 1 and 3. In Vol I vehicle and unit qualification and acceptance test requirements are presented in paragraphs 6.2.7, 6.4.2, 7.2.7, and 7.4.2. Test tolerances are covered in Table III, and overall test requirements are summarized in Tables IV and XI of Vol I.

5.6.2 Rationale for Thermal Cycling Tests. Thermal cycling is required for electrical and electronic units primarily as an environmental stress screen. It is intended to enhance quality assurance by revealing latent workmanship or material defects. The types of defects found in thermal cycling include loose connectors, defective solder joints, inadequate stress relief, performance drift, and material deficiencies. See technical reference 6, for additional discussion of thermal test requirements. In the absence of experience which would prompt a change in the screening requirements contained in MIL-STD-1540B, it was decided to retain the level of severity for unit thermal cycle acceptance testing. However, it was determined that the qualification thermal cycling tests specified in MIL-STD-1540B did not provide an adequate demonstration of design fatigue resistance relative to the acceptance test requirements. Taking the position that thermal cycling poses a fatigue risk to flight equipment it was determined that the qualification testing should demonstrate fatigue insensitivity to the exposure experienced by flight units with a safety margin of four. Furthermore, for planning purposes the qualification test makes allowance for flight units to accumulate up to twice the required number of prescribed acceptance cycles due to failures or rework. (Fatigue issues are discussed in further detail in Paragraph 5.6.3.2 below.)

The parameters deemed important in regard to thermal cycling of units are the temperature range, number of cycles, dwell or soak duration, rate of change during temperature transitions, and operational conditions. For acceptance testing the first two parameters have been changed from MIL-STD-1540B in accordance with the relationship that number of cycles times the temperature range raised to the 1.4 power equals a constant. This change reduced the number of cycles, increased the temperature range, and generally retained the other features of the MIL-STD-1540B thermal cycling acceptance test. Thus the thermal cycling baseline in VOL I of the Handbook is 12.5 cycles over a 105 degrees C range, as contrasted with the 18 cycles over an 85 degrees C range for MIL-STD-1540B. The temperature default limits have been adjusted by extending the lower temperature value from -24 to -44 degrees C. These temperature limits may be tailored to conform with mission temperatures, while preserving the temperature range. It is recognized that the extension of the lower temperature bound, while less troublesome than increasing the hot limit, could cause unrealistic failure modes to be encountered in units, which are of adequate design for mission temperatures. In this event the user may tailor the requirement in accordance with the foregoing relationship. The intent of the half cycle set forth in the baseline requirement is to begin and end the cycling test with a hot cycle to minimize moisture condensation problems.

The number of cycles called for in the optional vehicle thermal cycling acceptance test has been reduced from 40 in MIL-STD-1540B to 4. For qualification the reduction in number of cycles is from 50 to 10. The temperature range is unchanged. Electrical units undergo considerably less environmental stress during the vehicle testing than they experience at the unit level of assembly. This vehicle level thermal cycle test is intended to test those devices and elements not tested as units as well as interconnecting hardware, such as propellant lines and wire harnesses. Experience indicates that the system-level stress-screening objective of this test is effectively satisfied by the diminished test exposure.

5.6.3 Guidance for Thermal Cycling Tests. Figure 1 of VOL I of the Handbook describes the important features of unit thermal cycling. Throughout thermal cycling, except for brief periods when the unit is turned off, the unit undergoing test is to be operating with its performance monitored. Monitored operation enables the identification of latent defects, and is, therefore, considered a vital ingredient in assuring effective testing. Prior to the formal start of testing, steps should be taken to preclude the unwarranted accumulation of moisture within unsealed test units. This may be accomplished by imposing a number of pretest cycles using dry air or nitrogen, where the cold temperatures are not permitted to fall below the dew point of the air entrapped within the unit. These pretest cycles provide a pumping action to expel moisture-laden air from internal spaces. To further reduce risk of condensation the test begins and ends with a hot half cycle. Temperature transition rates as measured at a representative location on the unit should generally exceed 1 degree C per minute, with a goal of 3 to 5 degrees C per minute. For a special category of units, such as digital computers, the final cycle should have a slow temperature transition to permit repetitious functional checkout over a narrow range of temperatures. Representative measurement locations should be chosen at a mounting point on a base plate for conduction-cooled designs, or on the unit enclosure for radiation-cooled designs. After stabilizing at the required hot temperature or cold extreme for each cycle, the unit is turned off and restarted. Turn-on at temperature extremes significantly enhances the effectiveness of the test. As an aid to more rapidly reaching the cold extreme, the unit may be powered off during cool-down when the temperature has dropped below the minimum expected service low temperature for acceptance, or minimum expected service temperature minus the qualification margin for qualification testing. The hot and cold dwells on the first and last cycles are 6 hours long to permit thorough functional performance testing. This testing should demonstrate that the unit meets its performance requirements within acceptable tolerances. In many cases, the maximum expected hot or cold service temperatures may lie within the prescribed range of 105 degrees C. In these cases, the maximum expected hot or cold temperatures may be employed for the performance demonstration rather than the prescribed +61 and -44 degrees C temperatures. For qualification testing, performance may be demonstrated during the first and last cycles to the maximum expected hot and cold limits extended, in this case, by the qualification margin.

Vehicle thermal cycling at acceptance and qualification levels is an optional test, which may be selected to augment the required thermal vacuum test. It represents thermal cycling environmental stress screening carried to the system level of assembly. Vehicle thermal cycling is generally less stressful on the units, which have undergone cycling over a much broader temperature range during unit testing. However, it exposes those units not subject to thermal cycling as well as interconnecting and integration hardware to cyclic thermal stresses. Although an additional test setup is needed if thermal cycling is selected, the facility requirements are considerably more modest than those needed for thermal vacuum testing. As with unit testing, measures should be taken to avoid condensation in internal spaces, and the vehicle should be operating and monitored for malfunction or failure.

5.6.3.1 Combination with Thermal Vacuum. As described in Table VI of VOL I of the Handbook, unit thermal vacuum cycles are counted toward accumulating the required number of thermal cycles for qualification or acceptance. In fact, all of the unit thermal cycling may be performed under vacuum conditions, as long as a consistent choice is made for both qualification and acceptance testing. The cyclic soak and transition times for thermal cycling and thermal vacuum testing are compatible, although temperature transitions may be slower for thermal vacuum cycling. If the thermal cycling option is selected for vehicle testing, the thermal vacuum testing can be reduced in number of cycles required (see Table VI). In thermal cycling the required cycles are accumulated more rapidly than in thermal vacuum testing. However, this efficiency is gained at the expense of the logistics associated with a separate test setup.

5.6.3.2 Qualification Testing. Qualification testing for thermal cycling (also thermal vacuum) is discussed in 6.1.3 of VOL I. Such testing is only required for electrical and electronic units, or units containing such elements, for the purpose of uncovering workmanship deficiencies that may result in flight failures.

Qualification for thermal cycling is based on a fatigue failure mechanism of the same form as for structural stressing. The temperature range ΔT (temperature difference between the cold and hot temperatures) takes on the role of stress and the number of thermal cycles N corresponds to stress cycles. The assumed relationship is

$$\Delta T^{p} N = \text{constant}$$

The fatigue exponent is p and its range is 1.4 to 2. For conservatism, 1.4 is used in VOL I. Also, as was done for vibration and acoustics, the fatigue factor of four is used as a qualification margin. As a result,

$$N_Q \, \Delta T_Q^{\,p} = 4 \, (N_{AMAX} \, \Delta T_A^{\,p} + N_f \, \Delta T_f^{\,p})$$

N_Q, N_{AMAX}, N_f are the number of cycles for qualification, acceptance and maximum number of thermal cycles in flight.

ΔT_Q, ΔT_A, ΔT_f are the corresponding temperature ranges.

The baseline in VOL I (Note 4, Table VI) is that N_{AMAX} equals twice the specified number of acceptance thermal cycles for a unit. Also, for a vehicle, N_{AMAX} equals the number specified for acceptance, assuming that acceptance cycles will not be repeated (Note 6, Table VI of Vol I). These assumptions are open to tailoring on a unit-by-unit basis. It was also assumed that temperature cycling in flight contributes negligibly to fatigue in comparison to acceptance testing (Note 5, Table VI). Any sources of significant thermal cycling, for which qualification is to be achieved (perhaps in production), should be accounted for on the right side of the above equation.

5.6.3.3 **Acceptance Test Limits**. Unit temperature limits for acceptance test are designed to provide a temperature range for effective thermal cycling, and generally represent terrestrial conditions which may be encountered world wide. While the latter goal may appear to be relatively unimportant for aerospace equipment, experience in designing equipment for aircraft applications provided much of the knowledge base used for early space equipment design, and still represents a familiar frame of reference. In order to avoid specifying a large number of cycles for qualification testing, it became desirable to reduce the number of thermal cycles required for acceptance testing from those specified in MIL-STD 1540B. Therefore, the environmental stress screening associated with each thermal cycle was increased. The temperature range was broadened from 85 to 105 degrees C by lowering the cold temperature bound from -24 to -44 degrees C. The decision to extend the lower temperature bound was deemed less likely to cause unreasonable design problems than increasing the hot temperature limit. Consultation with materials scientists indicated that some materials used for conformal coating of printed circuit boards may undergo property changes due to exposure to these low temperatures. However, no other major difficulties are anticipated. Nevertheless, if the extended lower temperature limits of VOL I of the Handbook impose unreasonable design penalties, and mission temperatures do not require these low temperatures, the lower limit can be raised with a commensurate increase in the number of cycles. See VOL I of the Handbook, Table VI, note 3.

5.6.3.4 **Fixtures, Test Control, and Tolerances**. Unit thermal cycling is typically performed in a thermal chamber, where temperature-controlled dry air or nitrogen is used to heat or cool the unit undergoing test. The flow rate of the heated or cooled medium and the rapidity with which its temperature can be changed influences the rate of temperature change achievable at the test article. It is a goal that the average temperature transition rate, as measured at a representative location on the unit undergoing test, be 3 to 5 degrees C per minute for effective stress screening. However, since the unit undergoing test may represent considerable thermal inertia, such a rapid transition may not be readily achieved, so a minimum requirement of 1 degree C per minute is permissible. As discussed earlier, unwarranted moisture condensation is to be avoided by pretest cycling or another method to purge the unit of moisture-laden air. Within the temperature range of -54 to +100 degrees C the test tolerance is ±3 degrees C. At temperatures below or above that range, tolerances appropriate to the method of test should be utilized. Furthermore, during steady state soak periods unit base plate temperatures are to be stabilized at a temperature within the allowable test tolerance and the rate of change of the temperature of less than 3 degrees C over a 30 minute period.

5.6.3.5 Unit vs. Subsystem vs. Vehicle. The use of thermal cycling as an environmental stress screening test is mostly oriented toward the unit level. In unit testing the temperature range, rate of temperature transition, and monitored performance can be optimized to achieve the goal of screening for quality defects. The advantage of thermal cycling at the vehicle level of assembly is that all elements of the system are subjected to stress screening, including those which are not thermal cycled as units, and interconnecting hardware. This advantage is acquired at the expense of a more modest temperature range, a slower transition rate, and diminished monitoring of the performance of individual units. Subsystem testing frequently retains some of the benefits and disadvantages associated with both unit and vehicle testing. Perhaps one of the most useful applications of subsystem thermal cycling is represented by its application to a deliverable subsystem preparatory to its shipment for integration into the vehicle. The subsystem usually represents a modest assembly of units where a reasonably rigorous test can be imposed, and it provides an excellent assurance of satisfactory quality.

5.6.3.6 Operating vs. Non-Operating. To assure maximum perceptiveness, the unit, subsystem or vehicle should be operating and its performance monitored as much as possible throughout thermal cycling testing. In this manner, anomalous performance or drift can be detected. Selected periods when a unit can be inoperative are after reaching required soak temperatures preparatory to hot or cold starting the unit under test. Hot and cold starting are also considered to offer a significant benefit in terms of environmental stress screening. Another time period when the unit may be powered off is during transition to the cold temperature extreme. The conditions for turning the unit off are that the temperature has decreased to the minimum expected temperature, if conducting acceptance testing, or minimum expected temperature lowered by the qualification margin in the case of qualification testing.

5.7 THERMAL VACUUM/THERMAL BALANCE TESTS

5.7.1 Standard Criteria for Thermal Vacuum/Thermal Balance Tests. Definitions applicable to the thermal environment are contained in paragraphs 3.3.1, 3.5.7, and 3.5.10 of VOL I of the Handbook. Thermal development tests for units and vehicles are covered in paragraphs 5.4.2 and 5.5.5, respectively. Paragraphs 6.1.3 and 7.1.1 set forth the overall test philosophy for thermal qualification and acceptance. They are complemented by Tables V and VI, and Figures 1 and 3 of VOL I of the Handbook. Vehicle qualification thermal balance and thermal vacuum test requirements are described in paragraphs 6.2.8 and 6.2.9. For subsystem thermal vacuum qualification see paragraph 6.3.4. Unit thermal vacuum qualification testing is described in paragraph 6.4.3. Acceptance thermal vacuum test requirements for vehicles and units are described in paragraphs 7.2.8 and 7.4.3, of Vol I respectively.

5.7.2 Rationale for Thermal Vacuum/Balance Tests. Thermal vacuum testing is vital in ensuring successful mission operation for units, subsystems and vehicles which operate at high altitudes. For upper stage and space vehicles it represents the essential conditions of the operating environment. Thermal vacuum testing provides assurance that the unit, subsystem or vehicle will operate successfully, and within expected thermal extremes in its mission environment. Thermal balance testing is normally conducted during thermal vacuum qualification tests of vehicles or subsystems. This test should be conducted for one-of-a-kind spacecraft, the lead vehicle of a series of spacecraft, a block change in a series of vehicles, upper stages, and sortie pallets designed to fly with the shuttle. The principal role of thermal balance testing is validation of the thermal analysis model. In fulfilling that role, careful attention must be given to features of the facility and test specimen which may influence test fidelity. It is also important that the test cases and instrumentation be fashioned to provide the insight needed to intelligently modify the analytic model, or to perhaps alter the test setup. In addition to the model validation role, this test provides a verification of the capability of the thermal control system to maintain temperatures within prescribed limits for a bounding variety of mission phases that are simulated in the test. Futher discussion of thermal vacuum/balance tests can be found in technical reference 6.

5.7.3 Guidance for Thermal Vacuum/Balance. Thermal vacuum testing is conducted at the unit, subsystem and system level for upper stage and space vehicles for both qualification and acceptance. It represents the basic required thermal test for all units. When both thermal vacuum and thermal cycling testing are required, as in the case of electrical units, the number of thermal vacuum cycles is increased to augment its environmental stress screening role, along with thermal cycling (see Table VI of VOL I of the Handbook). For other units, where only thermal vacuum is required, the number of cycles is just one for acceptance, and six for qualification, reflecting a role of verifying performance in the mission environment without stress screening features. For launch vehicle units the vacuum conditions may be tailored to reflect the appropriate mission altitude, thus avoiding the expense associated with achieving very low pressures. In addition, the vacuum requirement for acceptance testing of units which are sealed may be waived, as long as the integrity of the seal is verified during acceptance testing. Thermal balance testing is normally conducted as an adjunct of qualification thermal vacuum testing of systems or subsystems. Dedicated test phases are established specifically for collecting thermal balance test data. Functional testing should be curtailed during these phases because they will interfere with establishing thermal stabilization. VOL I of the Handbook states that at least two test conditions shall be imposed: a hot case and cold case. These test environments need not be the hottest or coldest environments expected in flight. Test or subsystem restrictions may prevent running the hottest and coldest environments and the purpose of the thermal balance test is to provide data that verifies the thermal model, not to test the spacecraft at its extremes. Additional phases in the thermal balance test should include: a transient case for thermal model correlation, a model validation case and cases that specifically verify the thermal control subsystem. Test temperatures from the model validation case are compared to temperature predictions from the correlated thermal model to assess the model's ability to predict temperatures in a known environment and to determine the model correlation errors. Thermal control subsystem verification cases should include environments that can specifically validate the performance of such items as heaters, thermostats, radiators, heat pipes and louvers. By exposing the test article to hot, cold, and transient thermal conditions with high fidelity simulation techniques, the analytic thermal model can be validated, and the performance of the thermal control system verified. Prior to the test, thermal predictions for temperature and heater activity should be made for each thermal balance phase in the test. After the test is completed, the temperature predications are compared to the corresponding test data. Differences greater than the correlation goal of ±3 degrees C require either a model adjustment or a good explanation. The correlated thermal model is then used to make the final temperature predictions for the various mission phases, including prelaunch, ascent and on-orbit. The thermal margins are the differences between these temperature predictions and the associated temperature limits. For passive thermal control, these margins should be 11 degrees C or greater and for active thermal control, these margins should be at least 25 percent control authority, in accordance with paragraphs 3.3.1.1 and 3.3.1.2 of VOL I of the Handbook. If these margins are less then those stated, a design change may be required. Although thermal balance testing is included in the section of VOL I of the Handbook which deals with qualification testing, thermal balance retest may become necessary as part of acceptance testing due to block changes or significant configuration modifications. Where experience is lacking with respect to thermal conditions, thermal balance testing during the development phase is recommended to aid in the analytic thermal modeling process.

5.7.3.1 Basis for Thermal Vacuum Criteria. Thermal vacuum test simulates thermal conditions encountered during high altitude flight conditions by elimination of convective heat transfer. Thermal conditions, then, are dependent upon conductive and radiative heat transport mechanisms. For unit testing the temperature range and cyclic parameters are the same as thermal cycling, with exception that transition rates may not be as rapid. System or subsystem thermal vacuum testing verifies unit interactions and interfaces, as well as overall system performance. Beyond functional performance verification the thermal vacuum test also provides a check of the thermal control system, including thermostat and heater operation, heater duty cycling, louver operation, heat pipe performance, radiator sizing and insulation effectiveness.

5.7.3.2 Qualification Testing. Same as paragraph 5.6.3.2, except that thermal vacuum qualification is applicable to a wide range of equipment

5.7.3.3 Acceptance Test Limits. Temperature limits for acceptance test of units are the same as those used for thermal cycling testing. As discussed under that heading, the temperature ranges recommended in VOL I of the Handbook are extended by 20 degrees C beyond those contained in MIL-STD-1540B by lowering the cold limit by 20 degrees C. The overriding goal in defining the temperature limits in VOL I of the Handbook was based upon preservation of the environmental stress screening qualities of MIL-STD-1540B acceptance testing with fewer cycles, without invalidating current successful design approaches. While it is understood that some materials problems may be encountered at the lower temperature extremes, the new cold limits of -44 and - 54 degrees C for acceptance and qualification are not beyond the range of experience for military equipment exposed to terrestial extremes. It is considered to pose far less of a problem than would result from broadening the thermal range by raising the hot limit. In any event, provision is made to tailor for a more modest temperature range by a commensurate increase in number of cycles, if required by design constraints, as long as the reduced thermal limits are beyond predicted service temperatures including appropriate uncertainty and qualification margins. The relationship for such tailoring is given in the notes to Table VI of VOL I of the Handbook. It is also permissible to adjust the temperature limits to correspond with service conditions, such as cryogenic temperatures, rather than the prescribed limits, while preserving the 105 and 125 degree C cycling range. If the predicted temperature range exceeds the baseline range in VOL I of the Handbook, the number of cycles for acceptance can also be commensurately reduced, using the same relationship. For system level thermal vacuum testing the system is typically subdivided into logical and manageable zones wherein thermal conditions can be controlled. In each of these zones as many units as practical (but at least one) should be driven to maximum and minimum expected temperature extremes with qualification margins if appropriate. Care must be exercised and sufficient instrumentation installed to assure that no units will be exposed to temperature conditions beyond their unit test extremes.

5.7.3.4 Vacuum Conditions. For thermal vacuum testing of space and upper stage vehicles and units VOL I of the Handbook calls for the pressure to be 13.3 millipascals (10^{-4} Torr) or less. Low pressures are necessary to exclude the unrealistic effects of convective heat transfer in simulating thermal conditions encountered in space applications. Although VOL I of the Handbook calls for pressures to be 13.3 millipascals, it is highly desirable that lower pressures be achieved where practical. Serious consideration was given to requiring a lower pressure value, but the increase in test cost for general thermal vacuum testing could not be justified. For launch vehicle components the vacuum pressures should be conservatively defined based upon mission exposures. An important feature of thermal vacuum testing is to provide for monitoring of units which may exhibit anomalous behavior in certain ranges of reduced pressure. Electrical and radio-frequency equipment, which may operate during ascent, or which may be operated before trapped gasses are able to fully escape should be checked for corona arcing and multipacting. When multipacting is a possibility a nuclear radiation environment should be simulated to initiate possible multipacting.

5.7.3.5 Thermal Uncertainty Margin. Reasons for utilizing a thermal uncertainty margin are discussed in paragraph 3.3.1 of VOL I of the Handbook. Comparison of temperature predictions with actual flight data for various spacecraft show that about 95 percent (2 sigma) of flight temperatures have been within $\pm$11 C of the value predicted by the analytical thermal model. Thus, the $\pm$11 C uncertainty margin has been shown by experience to be necessary in order to assure high confidence that flight temperatures will not exceed the maximum and minimum expected unit temperatures. Paragraph 3.3.1.1 of VOL I of the Handbook states that the thermal uncertainty margin for passive thermal control subsystems should be $\pm$17 C prior to validation of the thermal analytical model with thermal balance test data. The intent of employing an uncertainty margin greater than $\pm$11 C in the design development of a space program is to reduce the likelihood of necessary design changes following the thermal balance test to maintain the $\pm$11 C uncertainty margin. Comparison of temperature predictions with flight data for programs that did not undergo a qualification thermal balance test show that 95 percent of flight temperatures have been within ±17 C of the predicted temperature value. Flight experience shows that a ±17 C thermal uncertainty margin is appropriate for programs without a thermal balance test. For passive cryogenic subsystems, the thermal uncertainty margin is a function of the operational temperature range. Table II of VOL I of the Handbook specifies the appropriate margin, before and after thermal balance test validation, for temperature prediction ranges. The uncertainty margin is smaller at cryogenic temperature predictions, because, typically, the operating temperature range and the thermal design requirements are narrower. Furthermore, the decreasing temperature margin attempts to retain a constant equivalent heat load margin.

For active thermal control subsystems, paragraph 3.3.1.2 of VOL I of the Handbook states that a heat load margin of 25 percent may be used in lieu of the temperature thermal margins. This margin is also established on the basis of experience and is demonstrated in tests by monitoring the heater duty cycle. A maximum duty cycle of 80 percent demonstrates that the heater system has the required margin. Analysis may be necessary to show the equivalency of the 80-percent duty cycle when the heater temperature set point is greater than the minimum design requirement or when the input voltage is greater than the minimum design value. For example, a unit heater might be selected with a set point 6 C higher than the minimum specified temperature of 4 C. Because it requires more heat to maintain the unit at 10 C than would be required to maintain it at the minimum design temperature of 4 C, the demonstrated duty cycle can be greater than 80 percent. In this case, a 92-percent duty cycle measured with the 10 C set point might be shown by analytical means to have equal or greater capability than the 80-percent duty cycle design requirement for a set point of 4 C.

Likewise, analyses may be necessary to show margin compliance when external constraints, from testing or from adjacent hardware, prevent a heater zone location from reaching its heater set point temperature. For example, a unit heater might be selected with a set point of 10 C, but test constraints or adjacent heaters limit testing of that zone to 20 C. Because it requires less heat to maintain the unit at 20 C than would be required to maintain it at 10 C, the heater selected would have a lower duty cycle. In this case, analysis results might show that a 72-percent duty cycle measured at the minimum test temperature of 20 C has an equal or greater capability than the 80-percent duty cycle design requirement at the 10 C set point.

The requirement for heater margin in excess of 25 percent (i.e., duty cycles of less than 80 percent) may apply where small capacity heaters are used or where an 11 C decrease in the minimum local environment may cause a heater with 25-percent margin to lose control authority.

5.7.3.6 Passive and Active Thermal Control Subsystems. Table I of VOL I of the Handbook categorizes passive and active thermal control subsystems for the purpose of imposing the appropriate thermal uncertainty margin. For passive subsystems, thermal margins are demonstrated using a temperature margin described in paragraph 3.3.1.1. For active subsystems, thermal margins are demonstrated using a heat-load margin described in paragraph 3.3.1.2. It should be noted that the temperature margins for passive cryogenic systems given in Table II of VOL I of the Handbook are comparable to the heat load margins given for active systems in paragraph 3.3.1.2. The post-validation column of Table II corresponds to a 25 percent heat load margin while the pre-validation column corresponds to the 50 percent heat load design margin for active systems at the conceptual design stage. Thus, the user may apply the heat load margins described for active systems (for various milestones in the system development) in paragraph 3.3.1.2 of VOL I of the Handbook to passive systems (in lieu of temperature margins). This provides a more gradual reduction in uncertainty margins as the system matures. Also, it is generally easier to apply heat load margins during the development stage than specific temperature margins which in turn are generally easier to apply during hardware testing. Additional test guidance for specific devices are provided in the following paragraphs.

5.7.3.6.1 Constant-conductance or Diode Heat Pipes. Constant-conductance and diode heat pipes are categorized as passive devices. Thermal performance testing, which is conducted at the highest assembly level practical (subsystem or space vehicle level), should demonstrate the $\pm$11 C margin and should also provide, if possible, the data to demonstrate that each pipe is functional at the system level acceptance test. The design is verified by demonstrating at the unit level the heat transport capability with at least 125 percent of that required for the nominal predicted heat under the temperature conditions providing the smallest capacity margin. The nominal heat load is defined as that predicted by the analytical model for the worst combination of operational modes, environments, and surface properties.

5.7.3.6.2 Variable-conductance Heat Pipes. Variable-conductance heat pipes, using noncondensible gas reservoirs for temperature control, are categorized as active devices. Thermal performance testing, which is also conducted at the highest assembly level practical, should demonstrate an acceptable heat rejection margin, variable conductance range and heat pipe turn-off. These system operating parameters are frequently functions of the available radiator area and environment, and must be demonstrated at a high enough level to make the test meaningful. They are not solely dependent upon the design of the heat pipe. The heat rejection margin is shown when 125 percent of the nominal predicted heat load is applied to the evaporator mounting plate, under the worst hot case simulated conditions, and the plate temperature is equal to or less than the maximum expected temperature. The variable conductance range is shown when 110 percent of the nominal predicted heat load is applied to the evaporator mounting plate, under the worst hot case simulated environmental conditions, and the heat pipe still possesses variable conductance, as proven by the location of the gas or working fluid vapor interface within the condenser portion of the pipe. Heat pipe turn-off requirements depend upon the type of reservoir in the system. For a heat pipe reservoir with active temperature control, the heat pipe is turned off, i.e., decoupled from the condenser by virtue of the gas (vapor) location, when the evaporator mounting plate temperature is at least 6 C or higher than the minimum expected temperature. For a heat pipe with a passively controlled reservoir, the turn-off points should be at least 11 C higher than the minimum expected temperature.

At the unit level, the heat transport capability should be the same as defined for constant-conductance heat pipes, at least 125 percent of that required for the nominal predicted heat load at the maximum expected temperature of the evaporator. The reservoir and evaporator temperatures may be adjusted as required to facilitate the simplest test procedure with the ambient environment available.

5.7.3.6.3 **Heaters**. Hardwired heaters or heaters using fixed or variable resistance elements which demonstrate a large variation in resistance with temperature (such as “auto trace” or positive coefficient thermistors) are to be treated as passive devices. Resistance heaters with mechanical controllers (such as bimetallic thermostats), or commandable or electronic controllers are active devices.

5.7.3.7 **Fixtures, Test Control and Tolerances**. Thermal vacuum testing of units is performed with the unit mounted on a heat sink for designs where heat is rejected by conduction, by cold surrounding enclosures where heat rejection is by radiation or a combination where both mechanisms represent service conditions. Temperatures are controlled by heat generated within the unit under test, by temperature control of the conducting heat sink and/or the surrounding enclosure, and by use of insulation blankets as a means to control radiation. Conduction and radiation heat transfer should be controlled such that the same proportions are simulated in test as are predicted to occur in service, so that temperatures and gradients will reflect the mission environment. Vehicle thermal vacuum testing is performed in a vacuum chamber where cooling is provided by liquid nitrogen-cooled walls. Vol I of the Handbook describes four possible methods of supplying the thermal control for this test. External heating is usually provided by solar simulation or infrared radiance simulators. Solar simulation more realistically simulates the space environment, providing direct and reflected solar-like radiant heating, while allowing for shadowing and cavity effects; however, cost, test complexity, facility availability, and thermal considerations associated with the mission may lead to a choice of heat lamps rather than solar simulation. Temperature tolerances are specified at ±3 degrees C for temperatures between -54 and +100 degrees C. Beyond that temperature range tolerances should be determined by controllability of the selected test mode. Vol I of the Handbook specifies 13.3 millipascals (10^{-4} Torr) as the pressure for thermal vacuum testing of space vehicles, upper stages and their units. Vacuum pressures as low as .133 millipascals (10^{-6} Torr) are available in chambers using mechanical and diffusion pumps. Lower pressures are achievable through the use of cryopumps, sputter-ion pumps or turbo-molecular pumps. At pressures below 133 millipascals (10^{-3} Torr) the allowable test tolerance is given as ±80 percent.

5.7.3.8 Unit vs. Subsystem vs. Vehicle. Similar to other environmental tests it is intended that thermal vacuum test requirements be most rigorous at the lowest levels of assembly to aid in early problem identification and minimize the impact of failures which may otherwise occur at higher levels of assembly. As the level of assembly grows, the accuracy of environmental simulation improves, and the integrated system performance becomes more directly assessable. Unit-level thermal vacuum testing is intended primarily to verify performance under mission thermal environments. When conducted on electrical units it also contributes to the goal of environmental stress screening. Subsystem thermal vacuum testing may resemble unit testing, wherein a group of units are exposed to unit-type tests as an efficiency measure, or to allow more realistic performance verification than provided by testing individual units. More commonly, subsystem testing resembles vehicle level test characteristics. Subsystem testing offers the advantages of more readily controlled environmental conditions and the use of more modest test chambers than needed for vehicle testing. Care must be exercised, however to account for environmental effects of interfaces not present in the subsystem test. The vehicle thermal vacuum testing is intended to most closely simulate mission conditions. The attention given to simulation procedures and configuration realism is most pronounced in vehicle testing. Vehicle thermal vacuum testing should verify system performance under the conditions encountered during the mission. The degree of realism provides an opportunity to verify the analytic models, which are used to predict thermal conditions, and to check the ability of the thermal control system to maintain temperatures within expected bounds. Since unit temperatures are maintained within the extremes demonstrated during unit testing, verification of integration hardware, interconnecting elements, and thermal control subsystems represent the primary test objectives of vehicle testing.

5.7.3.9 Operating vs. Non-Operating. Most of the thermal vacuum testing should be conducted with equipment operating and performance monitored for indications of failure, malfunction or performance degradation. There are occasions when equipment may be placed in a non-operating state. In unit test for example, equipment which is not operating during ascent need not be on during the pressure reduction period of the thermal vacuum test. During unit thermal cycling and thermal vacuum testing, the unit should be powered off prior to hot and cold starts, and may be turned off during the final phase of cool down to aid in temperature reduction. The temperature at which the unit may be turned off to speed cool down is the minimum expected temperature for acceptance testing or the minimum expected temperature lowered by the qualification margin for qualification testing. During vehicle testing various units may be powered off to simulate individual duty cycles especially those associated with cold cases. Minimum powered operation may be used to check performance of thermostats and heaters, or to simulate conditions for cold turn-on following events such as transfer orbit or safe mode recovery.

5.8 LEAKAGE TESTS

5.8.1 Standard Criteria for Leakage Tests. Requirements related to vehicle and units for qualification and acceptance leak testing are described in paragraphs 6.2.6, 6.4.7, 7.2.6, and 7.4.9 in the standard. In addition, leak testing requirements are listed in paragraph 9.4.2 as part of pre-launch validation of the propulsion subsystem. Leak before burst criteria are under pressure testing and not part of leakage testing.

5.8.2 Rationale for Leakage Tests. The leakage tests are intended to demonstrate the capability of pressurized and hermetically sealed units and pressurized subsystems to meet their design leakage rate constraints. For both long life spacecraft and shorter duration experimental vehicles, leakage of propellant has been suspected in numerous catastrophic failures as well as fuel depletion that results in shorter mission life than expected.

For hermetically sealed units, the proper convection within the unit or maintenance of pressure at high enough levels to prevent arcing or corona effects can be important. Electric motors with brushes that are used on upper stages, inertial navigation units that rely on convection to distribute heat are examples of units that require leakage testing to prove flight worthiness. Batteries could short if sealed cells leak leaving a conductive liquid trail between an anode and ground. Safety related to propellant loading may need to be proven before the hazardous operation is performed. Confidence of regulators, isolation valves, valve seats, and seals in general may need certification before one vendor accepts hardware from another. Leakage testing needs to be conducted on all pressurized qualification and acceptance units. Acceptance leakage testing for thrusters or thermal units is optional. A good example of a leak test on a thermal unit would be a sealed heat pipe that undergoes vibration and thermal cycle testing.

Finally, leakage can become a perceptive parameter in determining if hardware items degrade as a result of environmental exposure. One of the main failure modes noted from acoustics testing is propellant leaks. All these considerations justify application of leak testing for all categories of vehicles for both qualification and acceptance.

5.8.3 **Guidance for Leakage Tests**. The leakage test method should be selected to suit the design and performance requirements of the hardware item. It should prove that the item can function in its operational environment within specification and without degrading leakage. Leakage should be determined after any testing that results in a mechanical stress on the seal. If dynamic seals are used, then the hardware should be exercised in its operational mode and environment to the extent possible. VOL I of the Handbook does not call for a specific method such as immersion test, helium sniffers, or fluid indicators. The test method is to be selected as appropriate for the leak rate and substance being sealed. When using a gross leak check by an immersion method, any observed leakage during immersion as evidenced by a continuous stream of bubbles emanating from the component indicates a failure of seals. Pass/fail criteria by other methods can be based on measured levels when tested in the environmental operational conditions or by the observation of the presence of the leaked medium if the test is not in the operational condition and leak rates are not measured.

5.8.3.1 **Unit vs. Subsystem vs. Vehicle**. Some propulsion systems allow integration where the subsystem and core structure can be assembled and proof/leak tested before integration into a vehicle. Then only limited leak testing at the system level is performed after the lines are mated. Some environments like shipping a vehicle require coarse leak checking to insure safety and can only be proven at a system level.

5.9 EMC TESTS

5.9.1 Standard Criteria for EMC Tests. Requirements related to vehicle level and units for qualification EMC testing are described in paragraphs 6.2.2 and 6.4.11 in Vol I of the Handbook. If required, acceptance requirements are outlined in Vol I paragraphs 7.2.2 and 7.4.11 for vehicle and unit levels respectively. Vol I of the Handbook refers to MIL-STD-1541, "Electromagnetic Compatibility Requirements for Space Systems," for specific requirements.

5.9.2 Rationale for EMC Tests. Electromagnetic Compatibility is achieved when the electronic system (1) functions properly in its intended electromagnetic environment, and (2) neither the system nor its units are a source of Electromagnetic Interference (EMI) or Radio Frequency Interference (RFI) to its intrasystem or intersystem environments. The electromagnetic environment is composed of both EMI and RFI radiated and conducted energy. EMC achievement therefore addresses control of two aspects of each, emissions and susceptibility.

Emissions must be controlled to limit the electromagnetic energy emitted, and thereby control the electromagnetic environment in which the equipment must operate. Controlling the emissions may eliminate an interference problem for much equipment.

Equipment susceptibilities to an electromagnetic environment must be considered during the design and development process. As equipment development progresses into production fewer EMI/RFI mitigation techniques are available. Early solutions to interference are therefore usually best and least expensive. The designer must anticipate EMC problems at the beginning of the process; find and mitigate at the breadboard and early prototype stages, and test the final prototypes for EMC as thoroughly as possible. This way EMC becomes an integral part of both the electrical and mechanical design of the equipment. As a result EMC is designed into and not added onto the equipment, and the results are more cost effective.

5.9.3 Guidance for EMC Tests. MIL-STD-1541 and MIL-STDs-461 and -462 provide significant guidance on the analysis and test applications for electromagnetic effects testing. Tests are performed at fixed levels to envelope conditions expected during a typical life cycle which includes integration, allows for degradation of hardware, and covers adequate safety issues from self-generated or external sources that are intentional or unintentional. The low risk approach to minimizing failures in the electromagnetic effects area are built on a tier testing system starting with unit level testing, proceeding into subsystem tests, and culminating with system level testing. Testing at lower levels of assembly other than unit level has many advantages in terms of engineering development testing in order to minimize costs and design changes on flight hardware. It can allow circuit to circuit manufacturing variability to be characterized and aid in troubleshooting. The following paragraph discusses testing at higher levels of assembly.

5.9.3.1 Unit vs. Subsystem vs. Vehicle. The cost of discovery of an electromagnetic effects design deficiency generally increases by a factor of 8 with each higher level of assembly. Finding a conducted susceptibility problem at the subsystem level puts many component circuits at risk and requires extensive analysis to show that latent defects have not been induced. Programs that tailor testing to minimize unit level EMC/EMI testing often have self induced compatibility problems at the vehicle level. The recommended testing approaches referenced in the MIL-STDs are designed to minimize risk. Where higher risk approaches are acceptable due to programmatics, certain testing should still be required. At a minimum, tests should be performed at the vehicle level. Some tailoring may be justified in applying lower voltage transient levels where analysis shows the voltage used in the test would not be exceeded in service conditions.

At the unit level, test criteria may be tailored when experience and analysis shows circuit robustness. When higher frequency communication links are involved, relaxation of requirements should be avoided. Magnetic field effects are not always an issue with certain types of payloads and need to be reviewed on a case by case basis. Care should be taken in reducing levels especially around link frequencies. If switching circuitry is involved, tests should be performed across the recommended frequency range. Requirements may be tailored if a clamped bus-battery approach is being used.

5.10 CLIMATIC TESTS

5.10.1 Standard Criteria for Climatic Tests. Climatic testing involves a family of qualification tests outlined in VOL I of the Handbook for units subject to exposure to Humidity, paragraph 6.4.12.2; Sand and Dust, paragraph 6.4.12.3; Rain, paragraph 6.4.12.4; or Salt Fog, paragraph 6.4.12.5. If the self induced condition of explosive atmospheric conditions exist, then tests per paragraph 6.4.12.6 of VOL I of the Handbook are outlined.

5.10.2 Rationale for Climatic Tests. Certain units or environmental protection enclosures can be exposed to significant natural or induced atmospheric environments for the shipment/storage/logistic events during their life cycle usage. Rain, high humidity, or salt fog can degrade the strength of some materials, promote corrosion, deteriorate surface coatings and can render electrical or electronic devices inoperable or even dangerous. Stresses from swelling, freezing or corrosion can occur. Failures of protective enclosures and hardware due to proximity to salt water are a well-known experience at most launch sites. Sand and dust can foul moving mechanical assemblies, plug vents, and degrade exposed electrical connections. Where applicable, devices operating in explosive atmospheric conditions need to be proven incapable of igniting a fuel-air mixture of concern. In general, fungus, ozone, and sunshine exposures are addressed by design and not test. VOL I of the Handbook provides basic requirements and refers to certain methods in MIL-STD-810, "Environmental Test Methods and Engineering Guidelines” which have wide acceptance within the industry.

5.10.3 Guidance for Climatic Tests. Most of the climatic test approaches are discussed by reference to MIL-STD-810 except for humidity and rain testing. Humidity is varied over temperatures and can be measured by a wet/dry wick or electronic sensing system. The chamber can be programmed per the steps outlined in Figure 2 of VOL I of the Handbook. Rain testing is modified from Method 506.3 in MIL-STD-810E, Procedure III for testing of blowing rain for shelters.

5.11 STORAGE TESTS

5.11.1 Standard Criteria for Storage Tests. The criterion for storage tests is contained in paragraph 7.1.4 Storage Tests: Vehicle, Subsystem, or Unit Acceptance of VOL I of the Handbook. General requirements are given for acceptance tests performed both during and after a period of storage. Such testing can be performed at any level of assembly and must relate to storage life tests performed as part of the item development or qualification program.

5.11.2 Rationale for Storage Tests. The service life (paragraph 3.5.6 of VOL I of the Handbook) of an item includes storage. Therefore, consideration is given to establishing test criteria assuring that operational capability has not been impaired by periods of storage. Storage is defined as the state of an item when placed in a depository for safekeeping after manufacturing and acceptance testing has been completed. The item may be packaged and environmentally conditioned. Storage may occur at the manufacturer, at a government facility or other bonded warehouse. Under certain conditions, storage may also occur on the launch pad. Experience on a number of space and launch vehicle systems has shown that periods of storage can easily exceed 10 years.

5.11.3 Guidance for Storage Tests. Storage tests are of two types; in-storage testing, performed periodically during the period of storage and post-storage testing, performed after storage to assure item operational readiness. In-storage testing is associated with actions necessary for preserving functionality in the face of natural environments such as time, temperature, relative humidity, chemical de-composition and gravity. Examples of items having potential for degradation during storage include batteries, composite structures, optical solar radiator assemblies and heat pipes, thermal control paints, deployable devices, optics, moving mechanical assemblies, electronic units (piece parts, potting compounds, conformal coatings, adhesives, solder and printed wire boards), solid rocket motor segments, liquid engine seals, and ordnance systems. Experience with DOD spacecraft programs indicate that multiple re-work of systems can often occur during storage. Such activity is outside the normal definition of storage and represents a break in configuration. Therefore, a broader range of tests, including but not limited to the normal acceptance testing recommended VOL I of the Handbook, Tables XII and XIII should be performed when re-work occurs during storage.

5.12 MODE SURVEY TEST

5.12.1 Standard Criteria for Mode Survey Test. Requirements for development and qualification mode survey testing are contained in paragraphs 5.5.2 and 6.2.10, respectively, of VOL I of the Handbook.

5.12.2 Rationale for Mode Survey Test. The mode survey test is an important element in defining the flight loads environment, which in turn is essential to verification of structural integrity. Structural integrity is a major consideration in assuring mission success, and in satisfying safety requirements for manned launch vehicles or for large space systems transported by aircraft. Consequently, this test is classed a qualification test. Unlike other qualification tests, unfortunately, there are no clear pass/fail criteria—it is more in the nature of an experimental investigation and its success depends on the care taken in setting up the test and on the skill of the test operator(s). Nevertheless, its satisfactory completion is critical to mission success.

The qualification mode survey test experimentally determines the system's modes of vibration and corresponding frequencies and damping. The measured information is used to produce a test-verified dynamic model, either by directly defining the model or by providing the basis for modifying the analytical model. The frequency range of interest is at least 0 to 50 Hz, although this can be launch vehicle dependent.

In general the mode survey test discussion and criteria contained herein apply to new or significantly modified space systems which require the use of high fidelity dynamic models to analytically predict the (low frequency) structural loads environment. The dynamic models are used in a series of dynamic response analyses, referred to as load cycles, simulating transient dynamic events occurring in the course of the mission. Typical events include liftoff, stage ignitions and shutdowns, gust, buffet, and—when applicable— landing. Each load cycle utilizes an updated model; the last, referred to as the verification load cycle, utilizes a test-verified model based on the mode survey test. The need for the test is due to a high degree of sensitivity of the response results (i.e. accelerations, internal member loads, etc.) to the structural dynamic characteristics of the major elements of the system. The output of the verification load cycle, which is the last set of loads predictions prior to flight, is used to determine adequacy of the structural design. As part of the structural evaluation, the loads are used to confirm that the static test loads were sufficient to verify structural capabilities.

5.12.3 Guidance for Mode Survey Test.

5.12.3.1 Application. Development mode survey testing may be conducted at the system level but is more generally done at the subsystem level, and then only to reduce the risk of uncovering a major local modeling problem during the qualification mode survey test. Such testing could be conducted, for example, on unusually complex subsystems or on long lead-time items where structural modifications to correct a design deficiency due to a modeling error late in the program could result in a costly schedule delay. Scale model development testing is a consideration for new booster systems which differ radically from existing systems.

Qualification mode survey testing is usually conducted at the system level for a spacecraft, upper stage, or launch vehicle. A common practice, however, is to break it down into a series of subsystem tests whose combined results are equivalent to the full system level test. A mode survey test on a single vehicle applies to a fleet, provided structural changes which significantly affect stiffness do not subsequently take place. Limited additional testing or even a new mode survey test may be necessary if local attachment structure for major components and/or subsystems are changed for later vehicles or new items are added and tied in at locations not adequately verified in previous testing.

The success of the mode survey test is as dependent upon the test planning and preparations as on the actual testing itself. A failure in the preparation process can lead to a costly recovery program or even irreparably compromise the loads analyses and hence the structural integrity assessments. Retest is generally impractical, because of cost and other demands for the hardware.

5.12.3.2 Test Configuration. The test configuration is key to achieving a successful test. Consideration should be given to separate testing of dynamically complex subsystems, such as solar arrays, replacing them with mass simulators in the test of the primary structural system. This approach has an analytical basis in the Hurty/Craig-Bampton and Benfield/Hruda modal coupling techniques (see References 7-9) and can substantially reduce test complexity. Boundaries with adjacent major systems must be well-defined and must be exercised by the test. In general, this means essentially fixed or mass loaded interfaces. (Special care must be taken where a local flexibility exists at the interface, such as at a V-band. In such cases a special fixture may be necessary, including sufficient adjoining structure so as to totally include the local effect or making the interface so rigid as to totally exclude it. This aspect must be coordinated with the testing of adjacent systems to avoid either including the effect more than once or not at all.) A third well-defined alternative is an unconstrained interface. When an interface is left unconstrained, additional testing directly loading the interface is required in order to exercise it and obtain data to verify its character.

Usually, the choice of a fixed, mass loaded, or unconstrained boundary condition is based upon which constraint is closest to the flight configuration: fixed constraints at the launch vehicle interface are chosen for spacecraft; fixed constraints are chosen at the launch vehicle interface with mass loading at the spacecraft interface for upper stage vehicles; and unconstrained boundaries are chosen, with mass loading at the spacecraft/upper stage interface, for launch vehicles. In the latter case care must be taken to avoid local stiffening of the structure by the mass simulator. A possible alternative for large systems (such as Shuttle cargo elements), which has seen limited use, is to establish unconstrained conditions at all boundaries, and to use static or dynamic testing involving direct loading of the interface degrees of freedom which are to be constrained when coupling the system to other structures. In such an application, care must be taken to acquire accurate and complete data at the interface and to integrate such data into the model. At the present time, no accepted criteria exist which quantify accuracy requirements for this alternative process, so as to assure acceptability of the approach.

5.12.3.3 Test Stand. Minimization/elimination of test stand/test article interaction should be a primary objective in designing the test article support structure. Stand models should only be required to evaluate small effects. In particular, the test stand should be designed so as to preclude stand induced characteristics within the frequency range of interest and an effort made to verify this by test prior to installation of the test article. Applicable tests could include determining the low frequency modes of seismic masses supported on soft springs (such as airbags); tap tests of (rigid) mass loaded stand support structure; or impedance/admittance testing of stand supports. The alternative of attempting to analytically account for substantive stand characteristics should be avoided; it can seriously compromise the model, since the stand data acquired during testing is often inaccurate and may not be sufficient to differentiate between test article and stand model corrections. Consequently, failure to provide the well defined boundary conditions described in the previous section could jeopardize the entire test program.

5.12.3.4 Test Success Criteria.

5.12.3.4.1 <u>Completeness</u>. Identification of all modes in the range of interest is accomplished by a physical review of transfer function data to be sure that all modes are adequately excited. The review of transfer functions is made more manageable by the use of Mode Indicator Functions (MIFs) and Kinetic Energy Distribution Functions (KEDFs). The MIFs are used to identify resonances and assess adequacy of their excitation; KEDFs identify the accelerometers which are the major participants in the modes and can be used to quickly identify the transfer functions which are of most significance in each identified mode. The success of this effort is dependent on shaker location and can be facilitated by pretest analysis. In general a number of locations are required to obtain adequate response in all modes. Reference 10 is one useful and systematic approach to determining optimum shaker locations. Unfortunately, tested system behavior often differs from model predictions, and adequate excitation of all modes may require adjustment of the points of excitation during the test. Consequently, using small, easily handled shakers has distinct advantages (see also Paragraph 5.12.3.7 on test levels). In any event, pretest planning should assure adequate support structure exists to facilitate alternative and unexpected shaker relocations to assure that adequate response levels are achieved for all modes.

5.12.3.4.2 <u>Test Mode Quality</u>. To minimize shaker relocations and time consuming review of transfer function data, data quality needs to be assessed. Verifying acceptable quality of the test modes requires a quantitative measure. Visual comparison of test and analysis modes, for example, is not an adequate check on the quality of the test modes, in part because the physics of the problem requires that the mode shapes be mass weighted. A quantitative measure of quality is achieved by applying a self-orthogonality check, using the analytic mass matrix. This has its basis in the physical problem, is part of the mathematical eigenproblem, and assumes that the mass matrix is accurate and modeling errors are in the stiffness matrix. It is noted that although this latter assumption is usually the case, it is not necessarily so. In addition to errors or misjudgments which may happen in developing a model and a mass matrix, a particular problem may occur with the use of a Guyan reduction (Reference 11) to obtain a reduced order mass matrix consistent with the test instrumentation. The reduction transformation is stiffness dependent. Consequently, corrections to stiffness values through the correlation process may affect the reduced mass matrix and hence also the self-orthogonality check of the test modes.

The ten (10) percent orthogonality criterion provided in VOL I of the Handbook is written as

$$\{f\}_i^T [M] \{f\}_j \leq 0.10 \qquad \text{for } i \neq j$$

where $\{f\}_i = i_{th}$ mode, $[M]$ = analytic mass matrix, and the modes are normalized to a unit generalized mass. The ten per-cent criterion is primarily based on experience; however, analysis results have demonstrated that significant deviations from this criterion can result in significant effects on the loads computed from dynamic response analyses.

Nevertheless, it should be noted that there are technically acceptable exceptions to this criteria. The most common is for two modes whose frequencies are nearly coincident. For close modes significant coupling terms can result from very small stiffness coupling terms. A typical exception applied on DoD programs is to ignore contamination between two modes whose frequencies are within 3 percent if an effort to decouple the modes fails to achieve significant improvement. The 3 percent term is related to resonance bandwidth and corresponds to 1.5 percent of critical damping, a common value for aerospace structures.

In addition to verifying that an orthogonal set of modes have been determined from the test, the orthogonality check identifies multiple measurements of the same mode— for example, from data for different shaker locations. In this case, the orthogonality check provides a basis, along with other criteria, such as amplitude of response, for selecting the best of the measurements.

An alternative criterion often applied is known as the Modal Assurance Criteria (MAC). This is sometimes referred to as an “orthogonality check with a unit mass matrix” because of its form, but it is more appropriately classed as a check for linear independence. The MAC is widely applied because of its simplicity—it doesn’t require the development of a model mass matrix—but its usefulness is limited. For example, it is quite effective for assessing repeated modes. However, since even the use of true modes in a MAC calculation can produce significant off-diagonal terms, its use to quantitatively compare different modes as a check on the level of contamination between modes, is inappropriate and can produce very misleading results.

For either the self-orthogonality check or the MAC, coupling terms near the value 1.0 may actually reflect two separate modes—not a repeated mode—and indicate a problem of insufficient instrumentation. Distinct frequencies imply separate modes. Although this issue is clear for modes well separated in frequency, for close modes a careful examination of transfer function data may be necessary to determine if the modes obtained are distinct but lack some key measurement or are repeated.

Finally, it should be noted that orthogonality checks assume a valid mass matrix, which may not be true. Repeated and consistent mode shape measurements for a problem mode (i.e. one which shows significant coupling with other modes), taken from response for different shaker locations and response levels, would indicate a mass matrix problem if it is clear that there is adequate instrumentation.

5.12.3.5 **<u>Test Method</u>**. Mode survey tests are essentially based on frequency response methods. There are basically two approaches to such testing: (1) the multi-shaker sine dwell technique, which essentially isolates and measures individual mode shapes directly, and (2) the curve fit mode extraction approach, in which all modal parameters are extracted from transfer function data through a curve fitting process, either by directly curve fitting the data or by curve fitting time histories of impulse response functions obtained by transforming the frequency response function data.

The sine dwell test has the advantage of requiring only small shakers easily moved to new locations, and of allowing both definition of the modal parameters and evaluation of orthogonality checks before teardown of the test setup; it is, however, time consuming. The transfer function approach requires much less test time but generally requires more care in acquiring the data since a detailed evaluation of the data generally doesn’t take place until after teardown of the test configuration. In this case any data deficiencies may be disastrous, requiring extensive analytical efforts to recover.

5.12.3.5.1 **<u>Multi-shaker Sine Dwell</u>**. In this technique multiple shakers (typically up to four) are used to “tune-in” a mode, exciting it at its resonant frequency. The phasing of the various shakers is adjusted so as to maximize the response of the target mode and minimize the response of nearby modes. The quadrature component of the resulting response shape, ninety (90) degrees out of phase with the input forces, is essentially the mode shape of the system. Each mode is acquired separately. Although the process is time consuming, significant response amplitudes are easily achieved, the mode shape data is acquired directly, and orthogonality checks can be made at once to verify the quality of the data. One advantage of this approach is that poorly excited modes are quickly identified as such and shakers can be relocated or the phasing changed to obtain better excitation. The decision to teardown the test setup can therefore be made fully knowing the quality of the data.

5.12.3.5.2 Curve Fit Mode Extraction Process. This process consists of extracting modal parameters via a curve fitting process using experimentally derived transfer function data. The curve fit can either be in the frequency domain using the transfer function data itself, or in the time domain using impulse response functions obtained by appropriately transforming the transfer function data. A variety of excitation methods can be used to acquire the transfer functions, such as direct frequency response measurements, or single/multi-point random response using either stationary or burst random inputs.

Each excitation method has its own advantages and disadvantages. For example, direct measurement through a stepped sine dwell procedure over the entire frequency range easily achieves significant response levels and is particularly useful for evaluating nonlinear behavior. It is, however, a more time consuming approach than other techniques and may inhibit nonlinearity studies. Determination by use of band-limited random inputs has the advantages that it produces data quickly across the entire frequency spectrum and linearizes nonlinearities in the system, but has the drawback of low response levels at individual frequencies because the input is dispersed over a relatively wide frequency range; hence, it may require the use of larger shakers which are more difficult to move to new locations. In all cases efforts should be made in the process to achieve adequate response levels and to minimize any errors in developing the transfer functions. Particular attention should also be paid to the frequency increment used in developing the data (4-5 spectral lines per bandwidth of modal response is a suggested increment).

The advantage of the curve-fit approach is that it is not necessary to obtain nearly pure response in individual modes, hence the data acquisition process is simpler and shorter. However, although preliminary estimates of the modal parameters can be readily acquired with today's software, the best extraction of the data and identification of any data deficiencies often requires additional data analysis following teardown of the test, particularly for complex systems. As a consequence, the test teardown decision may be made with data of limited accuracy, and the later analysis may show the data is inadequate, leading to costly and time consuming efforts to recover. Although rerun of the test to correct the problem is often not practical, because of cost and hardware scheduling problems, the necessary analytical efforts to recover may be just as expensive. The primary dangers in this approach are excitation at too low a level, using shaker locations which do not adequately excite some modes of the system, and inadequate instrumentation.

5.12.3.5.3 **Hybrid**. A hybrid method known as multi-shaker stepped sine testing combines the best features of each of the above approaches into a powerful and cost-effective mode test methodology. Transfer function data is used to obtain preliminary estimates of the modes, using multi-shaker random or sine sweep testing for a limited set of shaker locations (typically up to four). Shaker inputs are adjusted to acquire data at reasonable response levels to obtain realistic damping in the modes and provide preliminary assessments of nonlinearities. For modes which are not well excited or for which additional nonlinearity studies at higher response levels are desired, combinations of the shakers are then applied and narrow-band stepped sine sweep data is acquired in the vicinity of each target mode. The curve fit algorithms are then used to obtain more accurate estimates of frequency, damping, and mode shape. By varying force input levels, the frequency, damping and mode shape changes due to nonlinearities can be determined and recorded. For modes not adequately excited by the initial shaker locations, additional shaker locations are identified and the process repeated as required. Orthogonality checks and repeatability of mode estimates are used together to assess data quality. Typically, 4-6 additional shaker locations are sufficient to acquire all the necessary data. With shakers up to 50 pounds force capability and adequate pretest planning to provide shaker support structure, the shaker relocations and data acquisitions can be completed in time periods much shorter than required for a sine dwell test. This combining of test methodologies can produce excellent data quality with reasonable test time/cost.

5.12.3.6 **Test Instrumentation**. Substantive efforts should be undertaken to assure adequate instrumentation, and plans developed to facilitate adding measurements during the test if necessary, including possible use of a rover measurement. In particular, pretest analyses should be performed to verify adequacy of the instrumentation plan. Specifically, an analytical model reduced to the accelerometer locations (or to derived degrees of freedom) should be developed and verified that it produces the same modes and frequencies as the detailed model, and that the reduced order mass matrix is adequate for performing orthogonality checks. The instrumentation set is checked by using the "true" (analytical) modes obtained by extracting the accelerometer location mode shape data from the modes derived from the detailed analytical model and demonstrating orthogonality relative to the reduced mass matrix, and then demonstrating via cross-orthogonality checks with the "true" modes that the full and reduced model mode shapes are equivalent. This latter cross-orthogonality check is most effectively made by expanding the reduced order modes to the size of the detailed model and using the large order mass matrix. For this check it is recommended that the cross-orthogonality criteria be a maximum of 3 percent for off-diagonal terms and a minimum of 98 percent for the diagonal, since they use an analytical model uncorrupted by experimental error. In addition, because real structures typically exhibit modes of vibration with lower frequencies than predicted, the frequency range of interest should be extended by at least 20 percent in evaluating instrumentation adequacy. Finally, once this process is completed, a final review should be conducted to assure that major subsystems are adequately instrumented, considering inputs to the subsystems as well as responses, in order to facilitate identification of modeling errors when discrepancies between the test data and model predictions are identified.

5.12.3.7 **Test Levels and Linearity Checks**. The test is conducted at levels which are low compared to flight levels, to avoid introducing design requirements for conditions which are not encountered in flight. However, the levels should be sufficiently high as to exercise joints (particularly those designed as pin-ended joints), resulting in modes of vibration more representative of the flight hardware under flight loading conditions and developing realistic damping values. Otherwise the modes may be ambiguous, the basic test objective compromised, and the damping unrealistically low. Although each system must be evaluated, typical test levels should be at least 15 percent -30 percent of limit levels, reaching at least 0.2g-0.4g response levels on prime structure and as much as 1g on subsystems where high response levels are expected in flight.

At least limited testing should be conducted at levels sufficiently high as to evaluate linearity. In addition to looking at primary modes to evaluate linearity of the overall structure, the higher level tests should investigate modes dominated by any major subsystems suspected of being nonlinear. The effect of higher test level on modal frequency and damping should be recorded. Only damping values demonstrated in the test should be used in the loads analyses. Where a high degree of nonlinearity exists, every effort should be made to eliminate it, such as through shimming, and, to the extent practicable, the effect evaluated through separate testing or by analysis.

5.12.3.8 **Simulator Characteristics**. Other than engineering models which are flight-equivalent structures, simulators should be “rigid” (i.e., with first mode significantly above the frequency range of interest); they should reflect the item’s mass properties, including moments of inertia at the attach points; and unless impractical, they should be attached using flight hardware. Simple, quick, and inexpensive tests, such as tap tests, can and should be used to demonstrate that the simulators meet the frequency criteria.

5.12.3.9 **Liquid Fill Levels**. Liquid-filled tanks should utilize inert fluids replicating density and kinematic viscosity to the extent practicable. Fluid levels should be determined during the late phases of test planning to best reflect flight plans. Changes during testing could be a subject for the Test Evaluation Team. The objective is to achieve flight mass loading, representative of fluid slosh and fluid moment of inertia properties.

5.12.3.10 **Model-to-Test Correlation**. When a test-verified analytical model is to be developed rather than a test-derived model, adequacy of the model-to-test correlation must be quantified. Again, visual comparison of mode shapes is inadequate, primarily because each value of the shape must be mass weighted. This is accomplished mathematically by the cross-orthogonality check. (The failure to account for mass weighting is the primary shortcoming of the form of the Modal Assurance Criteria (MAC) used to compare test data to analysis predictions. In effect, the MAC weighs areas of highest response most heavily, even if the item whose response is measured is light in weight and not a significant factor in characterizing the mode.) Other methods, such as comparison of transfer functions, can be used; however, criteria of acceptability are not defined for such approaches at the present time. VOL I of the Handbook provides for consideration of such alternatives via the Test Evaluation Team. Any approach proposed, however, should quantitatively evaluate the correlation.

The cross-orthogonality criteria of VOL I of the Handbook is written as

$$\{f_{test}\}_i^T [M] \{f_a\}_j \leq 0.10 \quad \text{for } i \neq j$$

$$0.95 \leq \{f_{test}\}_i^T [M] \{f_a\}_i \leq 1.00$$

where $\{f_{test}\}_i$ and $\{f_a\}_i$ are the test and analytical mode shapes, respectively. As with the self-orthogonality check, there are technically acceptable exceptions to this criteria. In particular, close frequencies is again one such case, and the three (3) per cent criteria can be used here as well. Again, technical justification should be provided for ignoring the cross-orthogonality criteria for any other exceedances since failure to meet it could reflect an uncertainty in the model test-verification which could significantly impact the loads results. Although a model uncertainty factor is often used to deal with this problem, this is an imperfect approach which may unnecessarily penalize other areas of the structure.

5.12.3.11 Test Evaluation Team. The test evaluation team is an effective approach to cost/schedule control of the test. Real-time decisions are needed in nearly every test for deviations from the formulated test plan and test success criteria. The deviations must have a technical basis, but the decisions must be made quickly to avoid overruns and wasted test time. Because of the criticality of the test, however, the customer needs to have a representative on the Test Evaluation Team and participate in these decisions at the technical level.

5.13 STATIC LOADS TESTS

5.13.1 Standard Criteria for Static Loads Tests. Contents of paragraphs 6.3.1.3, 7.3.1.3, and 7.4.7.3 of VOL I of the Handbook provide the detailed static load test requirements for structural subsystems and units.

5.13.2 Rationale for Static Load Tests. These test requirements are intended to determine the adequacy of the structural strength and stiffness of the launch, upper-stage, and space vehicles.

5.13.3 Guidance for Static Load Tests.

5.13.3.1 Establishing Loads. For a new launch or upper-stage vehicle program, or for a space vehicle program which intends to build a fleet of vehicles, a dedicated qualification test article is needed. Static loads representing the design yield load and the design ultimate load should be applied to the qualification test article. Table V shows the required design ultimate loads and design yield loads for both metallic and composite structures used in a manned or unmanned flight.

If the structure is made of composite materials and/or with bonded construction, an acceptance proof load test is needed for every flight article in order to verify workmanship. The required proof loads for manned and unmanned flight are also shown in Table V. It should be noted that for Space Shuttle payloads, the required proof load is 1.2 times the limit load unless a damage tolerance test is planned.

Table V. Static Load Test Requirements

Item	Test Factors [1]	
	Acceptance Test	Qualification Test [2]
Metallic Structures		
Manned Flight	N/A	1.40
Unmanned Flight	N/A	1.25
Composite/Bonded Structures		
Manned Flight	1.2	1.40
Unmanned Flight	1.1	1.25

Notes: (1) Static load level is test factor times the limit load

(2) Design ultimate load. Design yield load is 1.0

For situations in which a dedicated test article is not built, options for the use of test hardware as flight articles are allowed as described in VOL I of the Handbook, Subsections 8.2, 8.3, and 8.4. The selection of a specific option should be made on the basis of program-unique needs. Increased design levels may be used to reduce program risk for either the flight-test or ground-test phase of the program, with attendant weight penalties. The test factors given in Table VI are minimum factors of safety to be used in conjunction with sound design practices and thorough analytical and test verifications for the design. These verifications include fully coupled dynamic load analysis and modal surveys; detailed stress analyses to show positive margins of safety; use of proven materials with well characterized allowables; and adequate development programs.

Option A is for a space vehicle program which has only a few flight articles and would like to select the flightproof strategy. To adopt this option, a proof load test should be conducted on all structures in the structural subsystem. The proof load should be at least 1.1 times the limit load.

If the protoqualification strategy is selected, the structural load test requirements as specified in option B of Table VI may be implemented. However, compliance with the no-detrimental deformation condition requires additional test instrumentation, at carefully chosen locations on critical structural elements, and careful post-test inspection. In addition to the required static load test at 1.25 times the limit load level, an ultimate design factor of safety equal to 1.4 or higher should be used in the structural analysis in order to provide assurance of structural integrity equivalent to the approach specified in Section 6 and 7 of VOL I of the Handbook.

Utilization of qualification test articles for flight generally leads to overdesign. Therefore, consideration should be giving to using a dedicated structural qualification subsystem only, with smaller payload items being qualified by options A or B, Table VI.

Table VI. Structural Test Requirements for Alternative Strategies

Option	Test Factor (1)	Design Factor of Safety			Strategy
		Yield	Ultimate		
			Manned Flight	Unmanned Flight	
A.	1.10	1.10	1.25	1.25	Flightproof (2)
B.	1.25	1.25	1.40	1.40	Protoqualification (3)

Notes: (1) Static load test level is the test factor times the design limit load

(2) Proof test on each flight article

(3) Reduced qualification test level on single flight article

5.13.3.2 **Unit vs. Subsystem vs. Vehicle Tests**. Static load tests are usually conducted at the subsystem level in the qualification stage because the inclusion of structural connections and joints is very critical and boundary conditions are often more readily simulated at the higher assembly level. However, in the acceptance of composite and/or bonded flight hardware, proof load tests may be performed at the unit level, depending on the nature of joints. Proof load tests of adhesively joined parts should be conducted at the subsystem level in order to detect material, process and workmanship defects of the joints.

5.13.3.3 Development Tests for Composite Structures. For composite and/or bonded structures, development tests are usually needed to provide an adequate design database and to establish the accept and reject criteria for nondestructive evaluation (NDE). The following paragraphs describe pertinent details:

a. The material system design values or allowables should be established at the laminate level by test of the laminate or by test of the lamina in conjunction with a test validated analytical method.

b. For a specific structural configuration of an individual unit, design values may be established by tests which include the effects of appropriate design features (holes, joints, etc.)

c. The effect of the service environments (including humidity and temperature) on static strength, fatigue and stiffness properties should be determined for the composite or bonded materials systems through tests such as accelerated environmental tests.

d. The effects of defects such as surface cuts or scratches, delamination or debonding and impact damage should be evaluated for manned flight hardware. The accept and reject criteria should be established if NDE will be performed.

5.14 PRESSURE TESTS

5.14.1 Standard Criteria for Pressure Tests. Contents of paragraphs 6.2.6.3, 6.4.8.3, 7.3.2.3 and 7.4.8.3 of VOL I of the Handbook provide the detailed pressure test requirements for pressurized subsystems.

5.14.2 Rationale for Pressure Tests. As specified in VOL I of the Handbook, the pressure tests should normally be conducted at two assembly levels: unit level and vehicle level. At the unit level, the pressure tests demonstrate adequate structural margin so that premature structural failure or excessive deformation will not occur at the maximum expected operating pressure (MEOP). At the vehicle level, the required pressure tests demonstrate the capability of pressurized subsystems to meet the flow, pressure, and leakage rate requirements specified.

The pressure test requirements for pressure vessels, pressurized structures and pressure components are specified in MIL-STD-1522A. Table VII summarizes the required test factors for qualification and acceptance. For pressure vessels, MIL-STD-1522A allows multiple approaches for the design, analysis and verification of metallic pressure vessels as illustrated in Figure 4. The selection of a specific approach depends on the desired efficiency of design coupled with the level of analysis and verification testing. For example, if the weight of the pressure vessel is critical, approach A is the obvious choice. By selecting this approach, the pressure vessel can be designed with a much smaller safety factor than that required by the American Society of Mechanical Engineers (ASME) Boiler and Pressure Vessel Codes. As shown in Figure 4, the minimum design burst factor can be as small as 1.5 while the minimum design burst factor is 4.0 according to ASME code.

Table VII. Pressure Test Requirements

Component	**Test Factors(1)**	
	Proof Test	**Burst Test(2)**
Pressure Vessels		
LBB Failure Mode	1.5	2.0
Brittle or Hazardous LBB	1.25(3)	1.5
Pressurized Structures		
Manned Flight	1.1	1.4
Unmanned Flight	1.1	1.25
Pressure Components		
Lines, Fittings, Hoses		

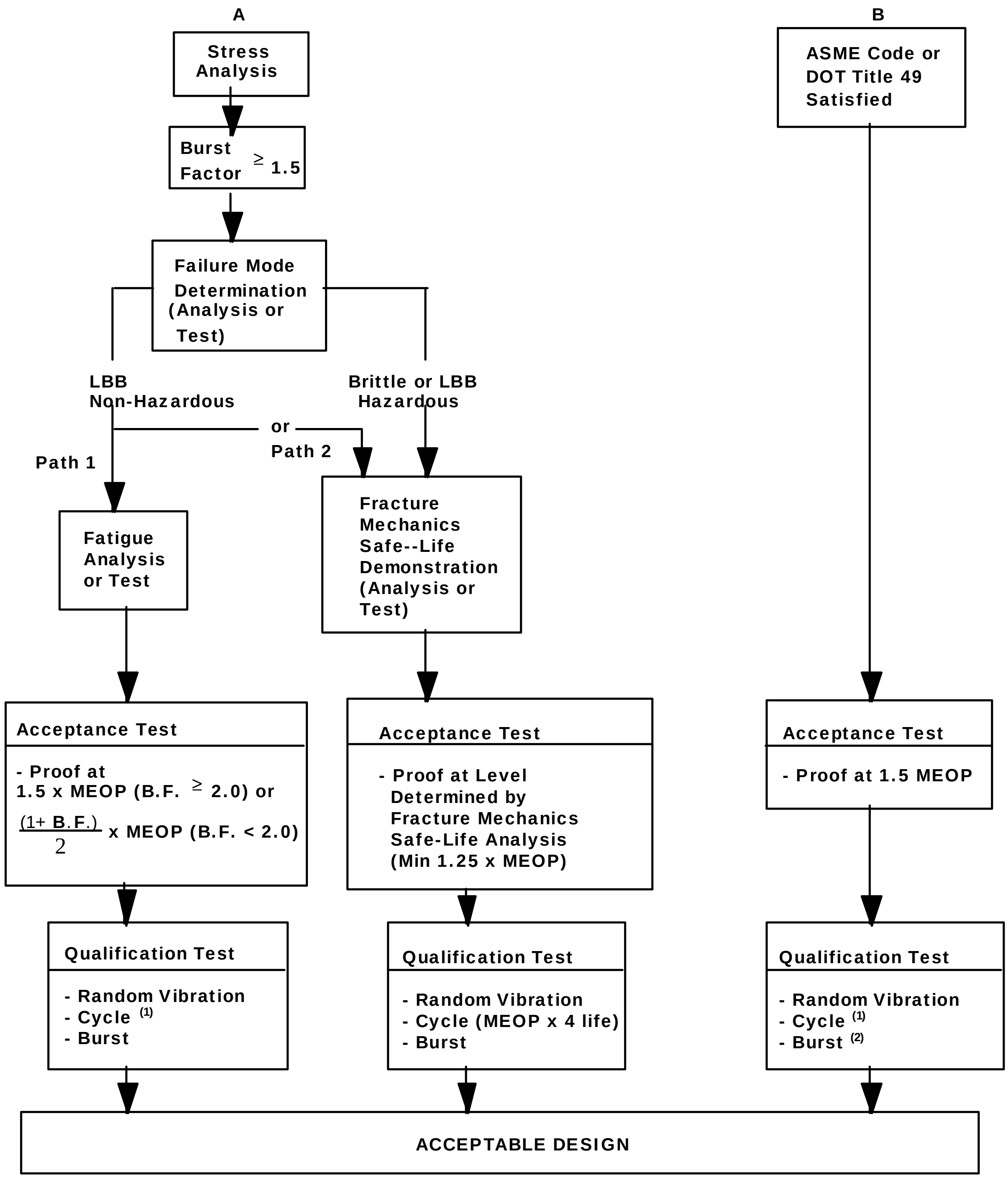
Component	Test Factors[1]	
	Proof Test	Burst Test[2]
diameter < 1.5 in.	1.5	4.0
diameter 1.5 in.	1.5	2.5
Fluid Return Section	1.5	3.0
Fluid Return Hose	1.5	5.0
Others	1.5	2.5

Notes: (1) Pressure level is test factor times MEOP

(2) Design minimum burst

(3) Proof test factor should be determined by fracture mechanics, 1.25 minimum

However, to select Approach A, extra verification steps are required in order to assure a high level of confidence in achieving safe operation and mission success. The first step is to determine the failure mode of the pressure vessel at MEOP. Based on the failure mode determination, one of the two distinct paths must be satisfied: 1) Leak-Before-Burst (LBB) with leakage of the contents not creating a hazard and 2) Brittle Fracture failure mode or LBB in which, if allowed to leak, the leak will lead to a mishap such as toxic gas venting or pressurization of a compartment not capable of withstanding the pressure increase. For pressure vessels with brittle fracture or hazardous LBB failure modes, two essential activities are involved: the safe-life demonstration and the acceptance tests. Safe-life demonstration can be accomplished by either fracture mechanics safe-life analysis or by safe-life testing. The acceptance tests consist of nondestructive inspection



NOTES: (1) Cycle test at either MEOP x4 life or 1.5 MPOE x 2 life
(2) Burst or disposition vessel with approval of the procuring agency

Figure 4. Pressure Vessel Design Verification Approaches

(NDI) and proof at test. The level of the proof pressure test should be determined by fracture mechanics analysis with the intention to establish the maximum

possible initial flaw sizes. However, the minimum proof pressure is 1.25 times MEOP. For pressure vessels with thin walls or made of materials with very high fracture toughness, NDI is required for detecting potential flaws or defects.

The vehicle level pressure tests are conducted after the assembly of a pressurized subsystem. At this stage, each unit has already been subjected to acceptance tests. Consequently, the main emphasis of the pressure tests is the pressure and leakage integrity of the interconnects. However, some units might have degraded during storage or as a result of transport, handling and assembly procedures. Subsystem proof pressure tests are required in addition to inspection for leakage.

The main propellant tanks of the launch and upper-stage vehicles carry both internal pressure and external vehicle loads. This type of hardware is defined as pressurized structures in accordance with VOL I of the Handbook. For qualification tests, pressurized subsystems of launch and upper-stage vehicles require a one-cycle proof test at a minimum proof pressure of 1.1 times MEOP. For space vehicle qualification, the pressurized subsystems, in general, require a three-cycle proof test at a minimum proof pressure of 1.25 times MEOP. The rationale for the differences in proof pressure cycle and level is because of the required long service-life for the space vehicle pressurized subsystems. For acceptance tests of pressurized subsystems of all vehicles, the required pressure cycle is one. This is to avoid the potential damage of the flight hardware due to excessive testing.

5.14.3 Guidance for Pressure Tests. The test description of paragraphs 6.2.6.2 of VOL I of the Handbook provides a synopsis for guidance for vehicle level pressure test. Guidance for unit level proof tests, pressure cycle tests, and burst tests is provided in MIL-STD-1522A. The following sections provide further guidance for design approach selection and fracture control on metallic pressure vessels, development testing and impact damage control for composite over wrapped pressure vessels (COPVs).

5.14.3.1 Pressure Vessel Verification Approach Selection. As shown in Figure 4, metallic pressure vessels can be designed to either meet ASME Code or to meet requirements specified in Approach A. The ASME Code requires a minimum design burst factor of 4. Design of a space flight pressure vessel with such a high safety factor would incur a large weight penalty and is not desirable. Furthermore, Approach B is not acceptable for most space programs since it does not require fracture control on pressure vessels having a brittle fracture failure mode or a hazardous LBB failure mode. For personnel and launch facility protection, pressure vessels with these types of failure modes should be placed under fracture control. Hence, Path 2 of Approach A is the best selection. From a mission success point of view, a leaking pressure vessel is not desirable. Therefore, it is still better to select Approach A, Path 2, which specifies fracture control.

5.14.3.2 Fracture Control. Fracture control is the application of design criteria, manufacturing technology, and operating procedures to prevent failure due to the initiation or propagation of flaws or crack-like defects during fabrication, testing and operational life. Metallic pressure vessels under fracture control are designated as fracture critical items which require safe-life analysis or testing in addition to the acceptance proof test. The initial flaw sizes assumed in the safe-life analysis should be based on the NDI detectability. Minimum detectable initial flaw sizes for five standard NDI methods approved by Air Force and NASA for use in the safe-life analysis or testing are shown in Table VIII. The crack dimension "a" is the crack depth of a part-through crack (PTC) or a corner crack, where the crack dimension "c" is the half crack length of a through-the-thickness center crack or the length of a through-the-thickness corner crack.

For pressure vessels, PTC at an open surface is the most common flaw. Pressure histories representing the service pressure conditions should be used for safe-life testing. Test environments such as temperature and humidity should simulate the service conditions unless the pressure levels have been pre-adjusted to compensate for the environment effects on material properties such as tensile strength and fracture toughness. Safe-life testing should demonstrate that after the application of four life-time pressure histories, the pre-fabricated crack will not become unstable or propagate through the thickness of the specimen. Coupons which have the same material properties and identical thickness can be used as test specimens in lieu of full scale pressure vessels.

The flight vessel should be inspected with the selected NDI methods before the performance of the acceptance proof test. Post proof test NDI should be conducted on welds to make sure that no damages have been introduced to the flight hardware. Fracture control procedures for metallic liners of COPVs are identical to the all-metal vessels.

TABLE VIII. Minimum Initial Flaw Sizes For Safe-Life Testing Based On NDI

NDI Method	Crack Location	Part Thickness t. in.	Crack Type	Crack Size a. in.	Crack Size c. in.
Dye Penetrant	Open Surface	t<0.050 0.050<t<0.075 t>0.075	Through Through PTC*	t t 0.025 0.075	0.100 0.150 0.125 0.075
	Edge or Hole	t<0.100 t>0.100	Through Corner	t 0.100	0.100 0.100
Magnetic Particle	Open Surface	t<0.075 t>0.075	Through PTC*	t 0.038 0.075	0.125 0.188 0.125
	Edge or Hole	t<0.075 t>0.075	Through Corner	t 0.075	0.250 0.250
Eddy Current	Open Surface	t<0.050 t>0.050	Through PTC*	t 0.020 0.050	0.050 0.100 0.050
	Edge or Hole	t<0.075 t>0.075	Through Corner	t 0.075	0.100 0.075
Radiographic	Open Surface	0.025<t<0.10	PTC*	0.7t	0.075

NDI Method	Crack Location	Part Thickness t. in.	Crack Type	Crack Size a. in.	Crack Size c. in.
		7 t>0.107		0.7t	0.7t
Ultrasound	Open Surface	t>0.100	PTC*	0.030 0.065	0.150 0.065

*Part-through Crack

5.14.3.3 **<u>Development Tests for COPVs</u>**. Composite Overwrap Pressure Vessels (COPVs) are a special class of pressure vessels. Their burst strengths are highly dependent on the design patterns of the composite overwrap which carry a large portion of pressure loads. In most designs the metallic liners only serve as a barrier to prevent leaking of the contained gases or fluids. Development tests should be conducted for a new design. These development tests include development tests of composite overwrap properties, failure mode and safe-life demonstrations, impact damage threshold determinations, and material compatibility verifications. The most direct method of generating overwrap properties is to test full-scale COPVs. However, subscale specimens can be used when the subscale-to-the-full-scale relationship has been validated previously. Full scale test articles should be used in every other development tests.

5.14.3.4 **<u>Impact Damage Control</u>**. Due to the weight benefit, Graphite/Epoxy (Gr/Ep) composite materials have been widely used to build pressure vessels such as COPVs. The current trend is that large solid rocket motor cases may also use Gr/Ep materials. However, these types of composite materials are known to be very susceptible to impact damage. Burst strengths of Gr/Ep composite pressure vessels (not COPVs) have shown a reduction factor as high as two, when subjected to non-visible impact damage. Hence, it is very important to have effective impact damage control implemented on all pressure vessels and pressurized structures made of Gr/Ep composite materials. Impact damage control should include one or more of the methods listed below. The selection should be based on the weight, cost, and configuration of each space system.

- Impact Damage Threshold Tests: To demonstrate that the maximum strain is less than the threshold value of the impact energy below which no burst strength degradation will result.
- Impact Indication System: To adopt an indication scheme which will indicate if impact damage has occurred.
- Impact Protection System: To utilize devices which can be used to protect the hardware from impact damage.

5.15 ACCELERATION TESTS

5.15.1 Standard Criteria for Acceleration Tests. Paragraph 6.4.9.3 of VOL I of the Handbook provides the detail acceleration test requirements for antenna, optics, and other units such as electric and electronic equipment.

5.15.2 Rationale for Acceleration Tests. The acceleration test is to demonstrate the capability to withstand, or, if appropriate to perform mission operations during exposure to the qualification level acceleration environments.

5.15.3 Guidance for Acceleration Tests. Paragraphs 3.4.8 and 6.4.9.2 of VOL I of the Handbook provides guidance for acceleration tests. To determine the test accelerations that should be used, it is usually necessary to construct time lines for the thrust levels applied and for the mass of the flight vehicle. From these time lines the average acceleration time line can be determined. The mass-spring dynamics that can occur due the step function applications and terminations of thrust can then be estimated and included as may be appropriate for the unit being tested.

5.16 LIFE AND WEAR-IN TESTS

5.16.1 Standard Criteria for Life and Wear-in Tests. Requirements related to unit qualification life testing and acceptance wear-in testing are described in paragraphs 6.4.10 and 7.4.10 respectively in VOL I of the Handbook. Development testing in Subsection 5.4 addresses life testing as recommended practice on critical items involving a wear-out failure mode.

5.16.2 Rationale for Life and Wear-in Tests. Life testing deals with design verification of units undergoing repetitive stresses due to varying loading conditions during operations. Lubrication suitability, wear-out, fatigue, charge capacity, or material degradation are factors that determine the necessity to show robustness for acceptance retest, and service life conditions. Wear-in testing consists of exercising a unit through its cycle, stroke, or rotation to detect material and workmanship defects that occur early in the unit life cycle. It also "runs-in" the unit so that it performs in a consistent and controlled manner. This can be seen in reduction of running friction, bearing noise, or peak motor current.

5.16.3 Guidance for Life and Wear-in Tests. For moving mechanical assemblies (MMAs), guidance for life and wear-in testing can be found in MIL-A-83577B, "General Specifications for Assemblies, Moving Mechanical, for Space and Launch Vehicles," in paragraphs 4.7.1 and 4.6.1 respectively.

It is important that life testing include all service life conditions including a reasonable amount of retest conditions. For deployable items that require movement to gain access to an equipment compartment, these conditions should be included in determining the test cycles.

When planning wear-in or life testing, unrealistic failure modes should be avoided. An example of this would be running a reaction wheel under ambient pressure when the operational condition is in a vacuum. Accelerated testing may be used to reduce the duration of life testing. For batteries and other hardware that are cycled, an appropriate margin over the life-performance requirement should be used.

5.17 Special Tests

The Handbook does not specifically address every test that a spacecraft, upper stage, or launch vehicle will require, such as alignment, calibration, antenna patterns, mass properties, or other items typically performed. However, that does not mean these tests are not required. Requirements for these types of tests are typically found within the configured item performance document or the interface control document for the item. Guidelines on how to implement these tests depend on limits allocated for tolerances, test setup effects, manufacturing variability, test margins, and other test specific criteria. For example, if gravity in the test setup was expected to effect the alignment, then an axis up measurement, and an axis down measurement might be averaged out to account for gravity effects. A jitter test might involve a sweep over the frequency range of the vibration of interest plus or minus 10 percent.

SECTION 6.

PRELAUNCH VALIDATION AND OPERATIONAL TESTS

Prelaunch validation testing is primarily accomplished at the launch base, with the objective of demonstrating launch system, on-orbit system, and re-entry system readiness. Prelaunch validation testing is usually divided into two phases:

Phase a. Integrated system tests (Step 3 tests, MIL-STD-1833).

Phase b. Initial operational tests and evaluations (Step 4 tests, MIL-STD-1833).

During Phase a, the factory preshipment acceptance tests establish the vehicle test databaseline; all factory test data should accompany delivered flight hardware. When the launch vehicle(s), upper-stage vehicle(s), and space vehicle(s) are first delivered to the launch site, tests should be conducted as required to assure vehicle readiness for integration with the other vehicles. These tests also verify that no changes have occurred in vehicle parameters as a result of handling and transportation to the launch base. The launch vehicle(s), upper-stage vehicle(s), and space vehicle(s) may each be delivered as a complete vehicle or they may be delivered as separate sections and first assembled at the launch site as a complete launch system. 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 that ship a complete vehicle to the launch site, these tests primarily confirm vehicle performance, check for transportation damage, and demonstrate interface compatibility with the launch facilities.

During Phase b, initial operational tests and evaluations (Step 4 tests) are conducted following the integrated system tests to demonstrate successful integration of the vehicles with the launch facility, and that compatibility exists between the vehicle hardware, ground equipment, computer software, and within the entire launch system and on-orbit system. The point at which the integrated system tests end, and the initial operational tests and evaluations begin, is somewhat arbitrary since the tests may be scheduled to overlap in time. To the greatest extent practicable, the initial operational tests and evaluations should exercise all vehicles and subsystems through every operational mode in order to ensure that all mission requirements are satisfied. These Step 4 tests shall be conducted in an operational environment, with the equipment in its operational configuration, by the operating personnel in order to test and evaluate the effectiveness and suitability of the hardware and software. These tests should emphasize reliability, contingency plans, maintainability, supportability, and logistics. These tests should assure compatibility with scheduled range operations including range instrumentation.

6.1 STANDARD CRITERIA FOR PRELAUNCH VALIDATION AND OPERATIONAL TESTS.

The contents of Section 9 of VOL I of the Handbook provide baseline requirements to be used in establishing prelaunch validation tests and operational tests for a specific program.

6.2 RATIONALE FOR PRELAUNCH VALIDATION AND OPERATIONAL TESTS.

The purpose of the prelaunch validation tests is to verify by end-to-end tests that each critical path in the launch system, in the on-orbit system, and in the re-entry system is satisfactory: i.e., there are no out-of-tolerance conditions or anomalous behavior. Duplication of the factory functional tests is a requirement that is intended to provide data for trend analysis that might provide evidence of a problem, even though all measurements were still within tolerances. Whether electrical, mechanical, or both, all critical paths or circuits should be verified from the application of the initiating signal through completion of each event. This testing is intended to verify that an event, command, or signal was properly generated and sent on time, that it arrived at its correct destination, that no unintended function was performed, and that the signal was not present other than when programmed. Once successfully accomplished, that particular critical path or circuit is considered validated. Not all end-to-end tests can be performed with only flight hardware, as in the case where an explosive event is involved. In cases where end-to-end testing cannot be performed with the flight hardware, appropriate simulation devices should be used to exercise the flight hardware to the maximum extent possible. Simulation devices should be carefully controlled and should be permitted only when there is no feasible alternative for conducting the test. All of the events that occur during the mission profile should be tested in the flight sequence to the extent that is practical. The space vehicle should be operated through the ascent sequence, separation and engine ignition phase, orbital injection, on-orbit, and if applicable, recovery phase. Redundant units and subsystems should also be validated in the same manner.

The purpose of on-orbit operational tests vary from program to program, but generally the first operational tests are conducted to verify the functional integrity of the space vehicle following launch and orbital maneuvering. Other on-orbit testing requirements are an important consideration in the design of any space vehicle. For example, there may be a need to calibrate on-line equipment or to verify the operational status of off-line equipment while in orbit. However, on-orbit testing is dependent on the built-in design features, and if testing provisions were not provided, the desired tests cannot be accomplished.

6.3 GUIDANCE FOR PRELAUNCH VALIDATION AND OPERATIONAL TESTS.

As with other test categories, the prelaunch validation tests that are actually conducted for a particular mission are a function of both the acceptable risk of incurring a mission failure, and the validation status of the hardware, software, and procedures involved in the mission.

6.3.1 Validation Status. In general, the prelaunch validation tests should follow a progressive growth pattern to ensure proper operation of each 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 vehicles or subsystems are verified, assembly proceeds to the next level of assembly. Following testing of the vehicles and their interfaces, the vehicles are electrically and mechanically mated and integrated into the launch system. Following integration of the launch vehicle(s), upper-stage vehicle(s), and space vehicle(s), functional tests of each of the vehicles shall be conducted to ensure its proper operation following the handling operations involved in mating. At each step, the focus should be on ensuring that previous test results are still valid and that previously untested interfaces are validated.

All ground equipment should be validated prior to being connected to any flight hardware, to preclude the possibility to faulty ground equipment causing damage to the flight hardware or inducing ambiguous or invalid data. Test provisions should be made to verify integrity on circuits into which flight jumpers, arm plugs, or enable plugs have been inserted.

In general, the Step 4 testing of the launch system is conducted first, then the Step 4 testing of the on-orbit space system is conducted. Note that the test configuration, ground equipment, RF interfaces, simulators, software, procedures and people involved with the on-orbit system are usually different than those involved with the launch system.

6.3.2 Tailoring of Requirements. The reduction of prelaunch test requirements can usually be justified if any of the systems involved are identical to one that was previously launched successfully. Another strategy is to configure and test the complete launch vehicle at the factory so that launch site operations can be reduced (ship and shoot philosophy)

SECTION 7.

SPECIAL TOPICS

7.1 RETEST

7.1.1 Standard Criteria for Retest. Paragraphs 4.8, 4.8.1, 4.8.2 and 4.8.3 of Vol I of the Handbook provide general information regarding retest requirements. Retest limits are addressed in paragraphs 5.3.3.3, 5.4.3.3, 5.6.3.3 and 5.7.3.3 of Vol II.

7.1.2 General Rationale for Retest. Retest is the repeat of previously conducted tests due to a redesign, a change in a manufacturing process, a test discrepancy, an increase in flight environments, or rework of items previously tested. Minor changes in design, manufacturing processes, flight environments, or rework can have a significant effect on the reliability of flight hardware. The results of analysis play an important part in decisions on the degree of retest. Regardless of analysis however, retesting is often necessary to restore complete confidence in the functional and environmental performance of flight items.

7.1.3 General Guidance for Retest

7.1.3.1 Requalification After Redesign. A redesign may be due to failure during previous testing or due to evolutionary design improvements. Redesign usually requires requalification to verify that design modifications have not introduced unpredictable failure mechanisms in the hardware. Maximum confidence in the integrity of a redesigned item exists if all previous tests are repeated. Since this is costly, compromises often must be made on the degree of requalification. The degree of requalification should be evaluated for each case considering the nature of the redesign, criticality of the hardware, degree of redundancy and cost of requalification. A key consideration is whether the design change can in any way affect the confidence gained from qualification of the originally designed item. The decision to requalify or on the degree of requalification therefore becomes a judgment on the tradeoffs between cost and the amount of acceptable risk.

7.1.3.2 Requalification After Process Change. A change in a manufacturing process may require requalification to assure that the new process has not had a deleterious effect on the capability or reliability of the hardware. Significant changes in manufacturing processes require requalification to assure that unpredictable changes have not been induced in manufactured hardware. The degree of process change that can be made without requiring requalification must be evaluated for each case considering the nature of the change, criticality of the hardware, degree of design redundancy and cost of requalification. Minor changes in the process of a simple manufacturing step would generally not necessitate requalification. On the other hand, relocation of a manufacturing facility, even with no overt change in manufacturing processes, would require requalification. The decision to requalify or on the degree of requalification becomes a judgment on the tradeoffs between cost and the amount of acceptable risk.

7.1.3.3 Retest After Test Discrepancy. The definitions of a test discrepancy and a test item failure are given in VOL I of the Handbook paragraph 3.5.8 and 3.5.9 respectively. Discrepancies may occur at any point in the qualification or acceptance test sequence of vehicle systems or units. When a discrepancy occurs, the test is interrupted and a determination is made as to whether the discrepancy was due to a failure of the item under test or a failure of the system performing the test (test setup, software, or equipment). Even if the item under test did not initially fail, it is possible that it could have been over stressed by a failure of the test equipment. After a determination is made that no over stress of the test item has occurred, the test may be continued after repairs of the test equipment used for performing the tests are completed. If the test item has failed, either originally or due to over stress, test activities resume normally only after a preliminary failure analysis which determines the cause and corrective action. If at all possible, it is desirable to freeze the hardware and software in the discrepant mode to allow a determination of failure cause. It is recognized that complete failure analysis can be lengthy, and that often tests must be continued before failure analysis can be completed. A preliminary failure analysis can be conducted to determine whether test continuation is practical or whether the test must be aborted. Factors entering into this decision are ease of isolation, ease of repair, and feasibility of continuing the test without repairing the discrepancy. Such a situation might exist where redundancy exists within a unit and the test could be continued on the redundant leg. An additional reason for test continuation without repair would be the need to troubleshoot and isolate the failed hardware or parts by test. After the failed hardware is isolated, the unit is redesigned or repaired. In either case, the degree of unit rework governs the amount of retest necessary. If a defective part or subassembly can be replaced by simply disconnecting and reconnecting electrical connectors using plugs or pins, retests may be minimized. However, rework generally results in considerable uncertainty regarding the validity of previous tests, and considerable retest is necessary to keep risks acceptable. Paragraph 7.1.3.5 below provides further discussion of retest after rework.

Final failure analysis is a continuing function to determine whether initial evaluations were correct or whether further action may be required, particularly if the failure represents a generic or lot-related problem. For long-term corrective action, one should determine if the failure could or should have been detected at a lower level of assembly or in an earlier test. If that is the case, all corrective actions that are appropriate at each level of assembly should be documented including all changes in test procedures.

7.1.3.4 <u>Retest After Change in Flight Environments</u>. Flight environments or their predictions often change as design modifications are made and data is acquired to verify early predictions. In other situations qualified units from a program may be relocated resulting in different service environments, or may be selected for use on another program. In these cases decisions must be made regarding requalification testing of existing units. In general, when the predicted environments have increased to the point that qualification margins have been reduced to less than half the original qualification margins then requalification should be performed. This requalification, or delta qual, may only involve the specific environments that have been revised. For example, if vibration predictions were to increase by 6 dB for units only a requalification or delta-qual of the affected units to higher vibration levels may be necessary.

The question of reacceptance testing of units also arises when flight environments are revised upward. For example, suppose that certain units have already completed acceptance testing and flight vibration predictions are revised upward by 6 dB. In this case the question is; should the units that have already completed acceptance testing at 6 dB lower than the latest flight predictions be reacceptance tested. The general guidance for unit testing in these situations is given by the following:

Environment Exceeds Acceptance Level By: Dynamic (dB), Thermal (C)	Unit Integrated Into Next Assembly Level (Yes or No)	Reacceptance (Yes or No)	Requalification (Yes or No)
< 3 dB or 5 C	No	No	No[(1)]
	Yes	No	No[(1)]
>3 dB or 5 C < 6 dB or 10 C	No	Yes	Yes
	Yes	No	Yes

Environment Exceeds Acceptance Level By: Dynamic (dB), Thermal (C)	Unit Integrated Into Next Assembly Level (Yes or No)	Reacceptance (Yes or No)	Requalification (Yes or No)
> 6 dB or 10 C	No	Yes	Yes
	Yes	Yes	Yes

Notes

(1) Assuming original qualification margins of 6 dB and 10 C

The question of retesting of hardware above the unit level of assembly may also arise due to changes in flight environments or changes in application of hardware. These situations occur less frequently than at the unit level of assembly and are dependent on the specifics of each situation, therefore, no additional guidance is provided in this Handbook.

7.1.3.5 Reacceptance Test After Rework. Rework as a corrective action frequently occurs during acceptance testing. The rework may be a repair which does not change the design. The rework may be significant or relatively minor. A significant rework may invalidate a number of previously conducted tests. A minor rework may have relatively small effect on the validity of previous tests. It is the purpose of this discussion to provide some considerations leading to judgments on the significance of reworks.

a. **Amount of Disassembly and Reassembly**. If hardware requires considerable disassembly to obtain access to perform the repair and subsequent reassembly, the majority of previous tests are probably invalidated, even if the actual repairs are relatively simple.

b. **Quantity and Complexity of Disconnects and Reconnects**. The number of disconnects to remove a failed part or failed hardware, the nature of the disconnects, and the complexity of performing the repair are important in evaluating the risk of degrading the hardware. If a part or unit can be simply unplugged, the risk of invalidating a previous test would appear less, since a functional test after the repair is completed could verify the adequacy of the repair, and possible damage to surrounding hardware is low. A repair requiring soldering or welding involves the risk of damage to surrounding hardware which could invalidate previous tests.

c. **Access to Inspect**. In-process inspection is an important part of manufacturing. As hardware is manufactured, visual inspection with optical aids, local measurements using hand-held test equipment such as voltmeters, force-gauge measures of compression, tension, or torque, local temperature measurements, and other inspection devices are used to inspect the adequacy of the assembly as hardware is being installed. If a repair can be inspected locally in the same manner as it was inspected during original manufacture, considerable confidence in its adequacy can be obtained. In general, it is noted that a repair which does not allow the same degree of in-process inspection as was done during original manufacture has invalidated previous tests.

d. **Repair Techniques**. During original manufacture, automated or manual production tooling may be used, depending on quantity. As an example, the soldering or welding of parts may be fully or partially automated and may be performed within the confines of a clean bench which protects the system from contamination. If a repair is performed under different conditions, using considerably different tooling and techniques than were used during original manufacture; it has invalidated the previous tests.

As a general observation, note that judgments relative to the risk of component degradation by rework are highly dependent on knowledge of the processes used during original manufacture. Consequently, a repair on a unit preferably is coordinated with the original manufacturer. Regardless of how the repair is performed, a risk of not discovering some defect exists if all previous tests are not repeated.

7.2 TEST SEQUENCE

7.2.1 Standard Criteria for Test Sequence. VOL I of the Handbook contains suggested sequences for testing in Table VIII, X, XII and XIII.

7.2.2 Rationale for Test Sequence. In order of their importance, the factors to be weighed in determining a particular test sequence are test effectiveness, cost, and simulation of flight or orbital conditions. In some cases none of these factors are overriding in which case the sequence is unimportant. In most cases however, one or more of these or other factors may be very important and careful thought should be given to establishing the test sequence for a given test article.

7.2.3 Guidance for Test Sequence. The test sequences in VOL I of the Handbook are suggested based only on the factors of test effectiveness and simulation of flight or orbital conditions. Cost considerations could not be significantly addressed since they are generally programmatic in nature and depend on factors such as availability of test facilities, location of test facilities, and hardware build and assembly sequence. The suggested sequences therefore should be used as a baseline and modified based primarily on tradeoffs between test effectiveness and cost and to a lesser degree on simulation of flight conditions. The suggested sequence given in VOL I of the Handbook is based on assuring that potential failures will be detected early in the test sequence and that the sequence of the service environments is preserved. Therefore, dynamic tests, which simulate the launch and ascent environment and are generally of short duration with limited performance testing, should precede thermal vacuum and thermal cycling tests which simulate longer duration conditions where greater opportunity is afforded for more extensive diagnostic testing. Normally, the sequencing used should recognize, especially for upper stages and spacecraft, that the thermal vacuum test is an orbital performance check that should be run towards the end of the sequence.

For reasons of test effectiveness the sequence of shock and vibration or acoustics should always have shock before vibration or acoustics. Experience has shown that shock can often induce a failure, an intermittent, or a latent defect that may not be detectable during the few milliseconds of shock exposure or after the shock test when at ambient conditions. A vibration or acoustic test of a minute or more will generally surface these problems which might otherwise be undetectable.

The mechanical and electrical functional tests are extremely important elements in the test baselines. The functional tests are conducted prior to and after each of the environmental tests. They should be designed to verify that performance of the units and of the vehicle meets the specification requirements, that the units and the vehicle are compatible with ground support equipment, and that all software used is validated, such as in computer-assisted commanding and data processing.

7.3 TEST DATA ANALYSIS

7.3.1 Standard Criteria for Test Data Analysis. Paragraphs 4.9.1, 4.9.2 and 4.9.3 of VOL I of the Handbook provide information regarding requirements for test data analysis.

7.3.2 Rationale for Test Data Analysis. Test data analysis is conducted to ensure that all specification requirements are met and to eliminate any incipient failures. Also, analysis ensures that a database exists from unit to system level, and among all like items of hardware, from which nominal performance variability can be determined and degrading trends identified. The database is also extremely important in evaluation of anomalies and to establish an industry database that can be used to refine and optimize test programs.

7.3.3 **Guidance for Use of Test Data Analysis**. Test methodology and monitored parameters should be the same from unit through system level to the maximum extent possible. Selected trends together with test data are recommended to be used as an integral element of hardware certification. Key parameter sheets should include all critical test parameters, and functional performance parameters. Any unusual or unexpected trends should be evaluated to determine the existence of trends towards an out-of-limit value or of an incipient failure within a unit or system interface. Comparison should be made to previous like-units to aid in determining whether an anomaly is peculiar to that unit or is generic in nature.

The requirement for trend data is applicable to those selected units, subsystems, and systems whose operating characteristics are judged complex and whose nominal repeatability is dependent on the stability of its constituent elements. Implementation requires a test methodology which looks at the same or related critical parameters at each level of test, such that degradation or failure detected at higher levels of assembly can be traced to the most probable cause at a lower level.

A matrix should be made showing evidence of test data review and data acceptance at each post-test review. Each matrix would then become part of the acceptance data package at the unit, vehicle, and system levels.

CONCLUDING MATERIAL

Custodians:

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Preparing Activity:

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Project 1810- 9902

STANDARDIZATION DOCUMENT IMPROVEMENT PROPOSAL

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I RECOMMEND A CHANGE:	**1. DOCUMENT NUMBER** MIL-HDBK-340A	**2. DOCUMENT DATE (YYYYMMDD)** 990401

3. DOCUMENT TITLE **TEST REQUIREMENTS FOR LAUNCH, UPPER-STAGE, AND SPACE VEHICLES, VOL II : APPLICATIONS GUIDELINES**

4. NATURE OF CHANGE *(Identify paragraph number and include proposed rewrite, if possible. Attach extra sheets as needed.)*

5. REASON FOR RECOMMENDATION

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