SECTION 2.4

STRUCTURAL AND MECHANICAL

2.4 STRUCTURAL AND MECHANICAL VERIFICATION REQUIREMENTS

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

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

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

The program outlined in Table 2.4-1 assumes that the payload is sufficiently modularized to permit realistic environmental exposures at the subsystem level. When that is not possible, or at the project's discretion, compliance with the subsystem requirements must be accomplished at a higher or lower level of assembly. For example, structural load tests of some components may be necessary if they cannot be properly applied during testing at higher levels of assembly.

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

2.4.1 Structural Loads Qualification

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

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

TABLE 2.4-1
Structural and Mechanical Verification Test Requirements

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

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

A = Analysis required.

T = Test required.

A/T = Analysis and/or test.

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

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

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

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

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

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

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

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

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

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

Notes:

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

2.4.1.4.1 Primary and Secondary Structure:

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

c. Weldments,

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

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

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

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

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

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

a. For Both STS and ELV Payloads

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

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

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

b. For STS Payloads Only

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

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

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

2.4.2 Vibroacoustic Qualification

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

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

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

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

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

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

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

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

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

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

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

a. Facilities and Test Control - The acoustic test shall be conducted in a reverberant chamber large enough to maintain a uniform sound field at all points surrounding the test item. The sound pressure level is controlled at one-third octave band resolution. The preferred method of control is to average four or more microphones with a real-time device that effectively averages the sound pressure level in each filter band. When real-time averaging is not practicable, a survey of the chamber shall be performed to determine the single point that is most suitable for control of the acoustic test.

Regardless of the control method employed, a minimum of four microphones shall be positioned around the test chamber at sufficient distance from all surfaces to avoid absorption or re-radiation effects. A distance from any surface of at least l/4 the wavelength of the lowest frequency of interest is recommended. It is recognized that this cannot be achieved in some facilities, particularly when noise levels are specified to frequencies as low as 25 Hz. In such cases, the microphones shall be located in positions so as to be affected as little as possible by surface effects.

The preferred method of preparing for an acoustic test is to preshape the spectrum of the acoustic field with a dummy test item in the chamber. If no such item is readily available, it is possible to preshape the spectrum in an empty chamber. In that case, however, a low-level test should be performed after the test item has been placed in the chamber to permit final adjustments to the shape of the acoustic spectrum.

b. Test Setup - The boundary conditions under which the hardware is supported during test shall duplicate those expected during flight. When that is not feasible, the test item shall be mounted in the test chamber in such a manner as to be isolated from all energy inputs on a soft suspension system (natural frequency less than 20 Hz) and a sufficient distance from chamber surfaces to minimize surface effects. During test, the test item should be in an operational configuration, both electrically and mechanically, representative of its configuration at lift-off.

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

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

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

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

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

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

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

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

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

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

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

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

The test item shall be attached to the test equipment by a rigid fixture. The mounting shall simulate, insofar as practicable, the actual mounting of the item in the payload with particular attention given to duplicating the mounting contact area. In mating the test item to the fixture, a flight-type mounting (including vibration isolators or kinematic mounts, if part of the design) and fasteners should be used. Normally sealed items shall be pressurized during test to their prelaunch pressure.

In cases where significant changes in strength, stiffness, or applied load result from variations in internal and external pressure during the launch phase, a special test shall be considered to cover those effects.

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

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

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

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

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

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

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

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

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

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

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

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

Where W = component weight.

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

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

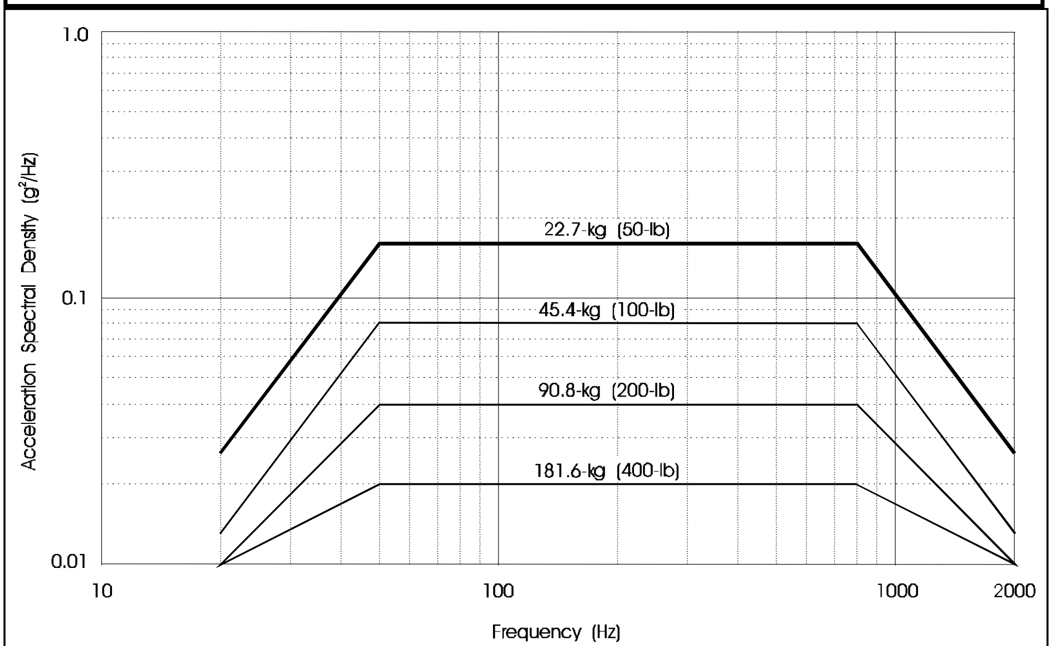


Table 2.4-5
Component Minimum Workmanship
Random Vibration Test Levels
45.4-kg (100-lb) or less

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

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

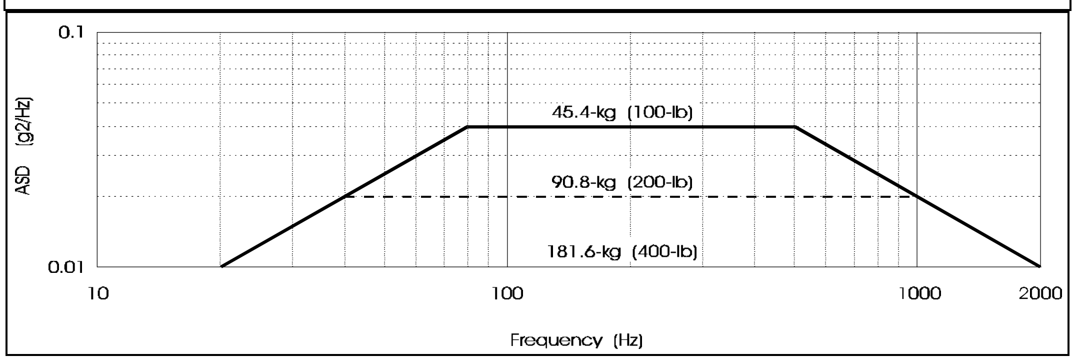
		Weight in kg	Weight in lb
dB reduction	=	10 log(W/45.4)	10 log(W/100)
$ASD_{(plateau)}$ level	=	0.04•(45.4/W)	0.04•(100/W)

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

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

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

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



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

2.4.3 Sinusoidal Sweep Vibration Qualification

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

In cases where significant changes in strength, stiffness, or applied load result from variations in internal and external pressure during the launch phase, a special test shall be considered to cover those effects.

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

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

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

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

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

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

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

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

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

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

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

2.4.4 Mechanical Shock Qualification

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

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

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

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

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

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

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

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

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

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

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

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

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

2.4.5 Mechanical Function Verification

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

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

2.4.5.1 Life Testing

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

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

The torque ratio is then given by:

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

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

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

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

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

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

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

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

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

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

2.4.6 Pressure Profile Qualification

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

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

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

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

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

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

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

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

2.4.7 Mass Properties Verification

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

2.4.7.1 Demonstration - The mass properties of the payload are verified by analysis and/or measurement.

When mass properties are to be derived by analysis, it may be necessary to make some direct measurements of subsystems and components in order to attain the accuracy required for the mission and to ensure that analytical determination of payload mass properties is feasible. Determination of the various subsystem properties should be sufficiently accurate that, when combined analytically to derive the mass properties of the payload, the uncertainties will be small enough to ensure compliance with payload mass property requirements. If analytic determination of payload mass properties is not feasible, then direct measurement is required. The following mass properties must be determined:

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

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

(1) Procedure for Direct Measurement - The usual procedure for direct measurement is to perform an initial balance before beginning the environmental verification program and a final balance after completing the program. One purpose of the initial balance is to ensure the feasibility of attaining the stipulated final balance. A residual unbalance of not more than four times the final balance requirement is the recommended objective of initial balance. Another reason for doing the initial balance prior to environmental exposures is to evaluate the method of attaching the balance weights and the effect of the weights on the operation of the hardware during the environmental exposures. Final balance is done after completion of all environmental testing in order to properly adjust for all changes to weight distribution made during the verification program such as hardware replacement or redesign.

(2) Maintaining Balance - It is recommended that changes to the hardware that may affect weight distribution be minimized after completion of final balance. The effects of such changes (including any disassembly, hardware substitution, etc.) on the residual unbalance of the hardware should be assessed. That involves sufficient dimensional measurement and mass properties determination to permit a judgment as to whether the configuration changes have caused the residual unbalance to exceed requirements. If so, additional balance operations may be necessary.

(3) Correcting Unbalance - To correct unbalance, weights may be attached, removed, or relocated. The amount of residual unbalance for all appropriate configurations is determined and recorded for comparison with the balance requirements of the verification specification. Balance operations include interface, fit, and alignment checks as necessary to ensure that alignment of geometric axes is comparable with requirements.

Balancing operations include measurement and tabulation of weights and mass center locations (referenced to hardware coordinates) of appendages, motors, and other elements that may not be assembled for balancing.

The data is analyzed to determine unbalance contributed by such elements to each appropriate configuration.

The facilities and procedures for balancing shall be fully defined at the time of initial balance, and sufficient exploratory balancing operations shall be performed to provide confidence that the final balance can be accomplished satisfactorily and expeditiously.

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