Appendix: Mission Specific Environmental Testing

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One of the challenges in space qualification is to define the operational environment of a part such that it is tested to the limits of a mission without requiring expensive overdesign. To aid this, this section defines, discusses and recommends environmental design and verification requirements for space microelectromechanical systems. Typical environmental program policies are presented, along with environmental design and test configuration requirements. Sample specifications are provided for a variety of environments, ranging from launch vehicle dynamics to ground handling conditions. Through judicious implementation of the analysis, test and verification techniques outlined herein, robust and reliable MEMS devices can be developed for long term survival in the unforgiving space environment.

A. Test Procedures

The fundamental purposes of an environmental test program are to simulate the launch environment, to qualify designs for launch and in-service conditions, and to screen flight hardware for manufacturing workmanship. Such a program should effectively demonstrate the quality and reliability of a design, as well as its suitability for the intended purpose or mission.

Environmental Compatibility Analyses are often conducted to verify hardware design compliance with mission environments that are impractical to verify by test. Design margins for these analyses must normally be higher than margins demonstrated by environmental test.

Such analyses are often conducted for the ground handling environment, including vibration and shock, temperature and humidity. Analyses are also normally conducted to demonstrate compatibility with explosive atmosphere requirements, and to prove structural integrity under launch pressure decay and thermal shock conditions.

Environmental Testing is conducted at two levels: the assembly/subsystem level and the system level. Assembly/subsystem level testing is completed prior to delivery for higher level integration into a flight system, and is generally the responsibility of a cognizant hardware engineer. The majority of space micromechanisms fall into this category. Post delivery environmental testing at a higher level of system integration is then usually conducted under the auspices of an Assembly, Test and Launch Operations, or ATLO, Manager.

Environmental tests are categorized for the purpose of hardware quality and reliability verification as *Protoflight*, *Qualification* and *Acceptance*, which are defined as follows.

Protoflight tests are conducted on flight hardware to demonstrate its ability to meet mission requirements. Protoflight test levels are generally equal to qualification levels, although test duration is often reduced.

Qualification tests are performed to a level and/or duration sufficient to demonstrate ability of a hardware design to meet mission requirements, with adequate margin. Such testing is generally conducted on a dedicated unit.

Acceptance tests are performed to detect workmanship or other defects which may have been introduced in the fabrication process, and to demonstrate hardware acceptability for flight. Acceptance testing is performed on flight hardware and spares when an adequate protoflight or qualification heritage exists.

In addition, development environmental testing is also often conducted to gain insight into design compatibility or functionality in expected mission environments. As an example, a dynamics test model of a flight system is sometimes assembled for purposes of structural verification.

i) Test Sequencing

To accurately simulate the environment sequence, flight hardware testing should be performed as follows:

1. Sinusoidal or transient vibration, random vibration, pyroshock and acoustics, as required. The order among these dynamics tests may be interchanged.

2. Thermal-vacuum testing.

During the normal flight sequence, the launch environment is followed by vacuum and potential temperature extremes. In this flight sequence, hardware is exposed to acoustics and vibration followed by vacuum and temperature variations. Consequently, by performing dynamics tests prior to thermal-vacuum tests, the actual flight sequence will be simulated. If the flight sequence produces synergistic effects, the synergism will also be simulated.

Experience has shown that until the thermal-vacuum tests are performed, many failures induced during dynamics testing are not detected because of the short duration of the dynamics tests. In addition, the thermal-vacuum test on flight hardware at both the assembly level and the system level provides a good screen for intermittent as well as incipient hardware failures.

Preserving the sequence of service environments in the environmental test program is a widely accepted practice. As a result, the effect of reversing the test sequence on spacecraft failure rates has not been quantified. However, evidence exists that many acoustic induced failures have not been detected until the spacecraft is exposed to the thermal-vacuum environment. These failures may not be detected during acoustics tests because of the short one-minute duration or a non-operating power condition. Typically, the identified failures that could be related to or caused by the dynamic acoustic environment were bad solder joints, intermittents, bad bearings, broken wires, poor welds, leaks, and foreign materials.

An example of a failure that might be induced by dynamic tests but not revealed until thermal vacuum, would be a broken wire or solder joint. This defect might be induced by acoustics but not be detected during the acoustic test due to the short duration of the test or to an unpowered or unmonitored state of the affected equipment. During post-acoustic functional testing, the wire or solder joint broken ends may be making adequate contact to show electrical continuity. In the subsequent thermal-vacuum test, the thermal distortions could cause loss of contact, allowing the failure to be detected. Reversing the test sequence could result in the defect not being induced until after thermal vacuum test and not detected until exposure to the flight thermal environment.

Even if all defects precipitated by the dynamics tests are revealed during the test or during post-test functional testing, performing dynamic tests first will nonetheless increase the probability of early defect detection, when correction of defects will have less impact on the flight program cost and schedule.

If the thermal-vacuum tests do not follow the dynamics tests, more intermittent or incipient discontinuity type failures may go undetected. If the defects are not detected during assembly level tests and are subsequently detected during the system level tests, redesign or rework at this late stage of the process could cause delays, increase costs, or make it necessary to accept additional risk that might have been avoided. If the defects are not detected at the system level, the defects may then cause hardware anomalies during the mission, and in the extreme could cause a mission failure.

B. Environmental Test Requirements

Appropriate in-situ environments must be determined and specified in order to effect a robust space microelectromechanical device design. These environmental design requirements depend upon factors ranging from the choice of launch vehicle to the type of spacecraft thermal control subsystem. Establishment of these requirements can be a time consuming task involving considerable research and analysis effort.

The requirements within this section encompass the basic launch environments, as well as those associated with ground operations and handling. They are offered here as generic baseline environmental levels, and should be used primarily as examples.

i) Launch Environment

The launch environment encompasses pre-launch operations, liftoff, and ascent. Typical requirements are provided here for both design and test of space micro- electromechanical devices, with environments including thermal conditions, deep space vacuum and insertion pressure decay, random and sinusoidal vibration, pyrotechnic shock and acoustic noise.

ii) Thermal

Spacecraft microelectromechanical systems should be designed to operate within specification over the temperature range of $-55^{\circ}C$ to $+70^{\circ}C$, or flight allowable $\pm 20^{\circ}C$, whichever is more extreme.

iii) Definitions

Terms used in thermal design and test of space microelectromechanical systems are defined as follows:

Operating Allowable Flight Temperatures: The temperature ranges of MEMS devices when powered-on in a worst case operational mode (hot or cold). In-spec operation is required.

Non-Operating Allowable Flight Temperatures: The temperature ranges of MEMS devices when powered-off in a worst case non-operational mode (hot or cold). MEMS devices must be capable of returning to in-spec operation as temperatures return to Operating Allowable Flight levels.

Design Temperature Limits: Temperature limits to which all MEMS devices should be designed to meet all functional and performance specifications.

Stabilization Temperature: In specification of test conditions, an assembly is defined to have attained a stabilization temperature when the rate of temperature change of its largest centrally located thermal mass is less than $2^{\circ}C$ per hour.

Control Temperature - Conductive Heat Transfer Tests: The control temperature for a thermal/vacuum conductive heat transfer test is defined to be the temperature of the heat exchanger plate midway between input and output of heat exchange fluid.

Control Temperature - Radiative Heat Transfer Tests: The control temperature for a thermal/vacuum radiative heat transfer test is defined to be the temperature of the major temperature control surface of the assembly (e.g. radiator).

iv) Thermal Radiation

Assembly allowable flight temperatures should not be exceeded during the mission under exposure to the applicable worst case expected thermal radiation levels in the accompanying table.

Mission Phase	Direct Solar	Reflected Solar (Albedo)	Planetary IR (LW Radiation)
Earth Orbit:	0 to 1400 W/m^2 (5770K effective blackbody temperature)	0 to 0.32 0 to 450 W/m^2 (global annual mean) 0 to 0.70 W/m^2 (polar regions)	100 to 270 W/m^2 (206K to 262K effective blackbody temperature)
Deep Space Cruise:			
Near Earth	0 to 1400 W/m^2 (at earth perihelion)	Negligible beyond 4 earth radii	Negligible beyond 4 earth radii

Table A-1: Thermal radiation levels.

v) Vacuum Pressure Decay

The design pressure for a typical mission can be expected to decrease from 101325 N/m^2 (760 Torr) on Earth to 1.33×10^{-3} N/m^2 (1×10^{-5} Torr) in deep space. A typical launch pressure decay rate, showing launch vehicle internal fairing pressure versus time, is provided in the figure below.

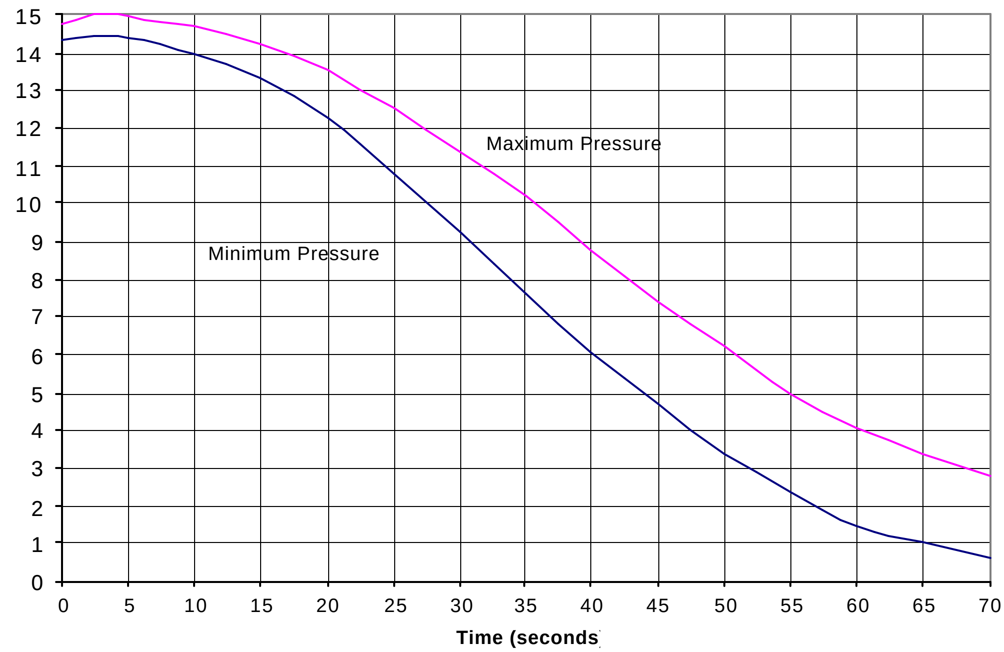

Figure A-1: Launch pressure decay rate.

Assemblies affected by launch pressure decay should be designed with a recommended structural design factor of 1.0 on yield and 1.4 on ultimate if tested, or 1.6 on yield and 2.0 on ultimate if not tested.

vi) Dynamics

Assembly-level vibration and shock tests, simulating launch vibroacoustics and upper-stage pyrotechnic separation events, represent the most severe dynamic environments for spacecraft hardware. Components of a spacecraft, at various levels of assembly, should generally be subjected to the following environments:

(1) Definitions

Sinusoidal vibration requirements are imposed to cover the various mid-frequency (5-100 Hz) launch vehicle-induced transient loading events.

Random vibration requirements are derived from launch vehicle induced acoustic excitations during liftoff, transonic and maximum dynamic pressure (e.g. “max q”) events.

Acoustic requirements are based on maximum internal payload fairing sound pressure level spectra.

Pyroshock requirements are intended to represent the structurally transmitted transients from explosive separation devices, including pyrotechnic fasteners utilized to effect spacecraft separation from the upper stage.

Quasi-Static Accelerations are associated with quasi-steady flight events generated by rocket motor-induced forces and other external forces which change slowly with time and for which the elastic responses are relatively small. Typical assembly design requirements for quasi-static acceleration environments are specified in the table below.

Axis	Acceleration (g)
Thrust	+14 ± 0.7
Lateral	+3 ± 0.3

Table A-2: Quasi-static accelerations.

Qualification testing of microelectromechanical systems for the quasi-static acceleration environment can be performed in a centrifuge. However, a low frequency sine vibration test, conducted on an electrodynamic shaker, can often be substituted for the relatively expensive centrifuge trial. If a microelectromechanical system is subjected to sine testing at a frequency sufficiently below its fundamental resonance, it will not vibrate, but will instead move as a rigid body under the input sine acceleration. To ensure pure rigid body behavior of the assembly under test, the frequency of sine excitation should generally be less than the microelectromechanical system resonance by a factor of two. More detail on each of the other test environments is provided below.

(2) Dynamics Test Tolerances

Tolerances for dynamics testing are provided below. The indicated tolerances are derived from space vehicle hardware test experience, and may be facility-, equipment-, and personnel-dependent.

a. Time: ±5 percent

b. Vibration Frequency: ±5 percent or 1 Hz, whichever is greater.

c. Acoustic Spectral Shape: Match to spectral shape of the specified Sound Pressure Level (SPL) in 1/3 octave bands.

d. Acoustic Overall Level: ± 1 dB of the specified level.

e. Random Vibration Spectral Shape: The Acceleration Spectral Density (ASD) shall be within ±3 dB when measured in frequency bands no wider than 25 Hz.

f. Random Vibration Wideband RMS Acceleration: Within $\pm$1.0 dB of that specified.

g. Pyro Shock : $\pm$ 3 dB 20 to 2000 Hz

h. Static Acceleration: $\pm$5%

(3) Sinusoidal Vibration

Sinusoidal vibration is employed to simulate the effects of significant flight environment launch transients. These transients typically produce the dominant loading on primary and secondary structure and many of the larger subsystems and assemblies. Sinusoidal vibration is the only widespread current method of adequately exciting the lower frequency dynamic modes, particularly those below 40 Hz.

Sweeping at a log rate between 1 octave/minute and 6 octaves/minute should avoid application of excessive fatigue cycles. The higher rate is near the upper limit which most control systems can accommodate without experiencing some instability. The use of logarithmic sweep rates has the advantage in that a nearly equal time is spent at resonance for a given Q, independent of frequency. Sinusoidal vibration levels can be derived as illustrated in the following example:

Step 1. Create analytically derived transient waveforms from various flight events:

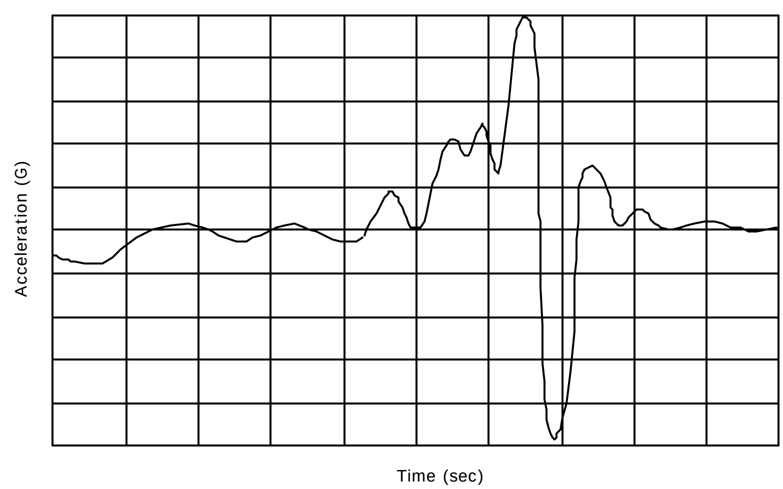

Figure A-2a: Creation of a sinusoidal vibration test profile (see 2b-e).

Step 2. Compute the shock spectra for each of the waveforms in Step 1:

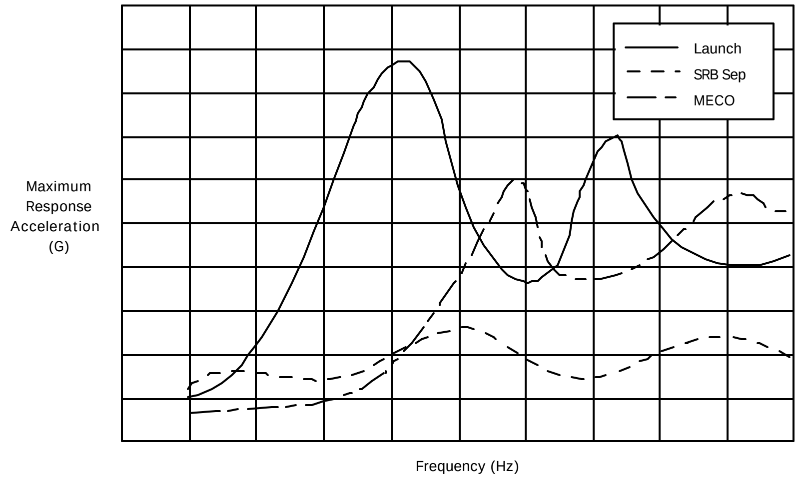

Figure A-2b

Step 3. Take data from previous flight measurements:

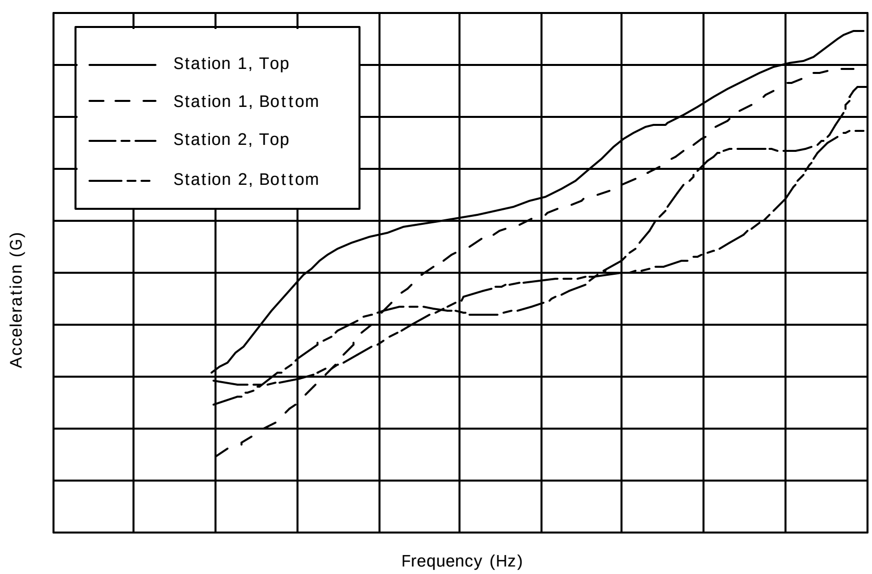

Figure A-2c

Step 4. Combine results from steps 2, and 3 and envelope:

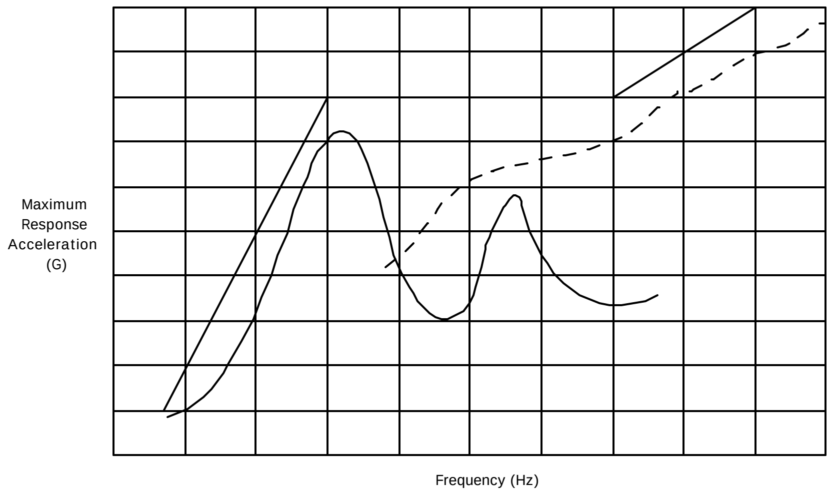

Figure A-2d

Step 5. Convert to a sine amplitude equivalent vs. frequency by dividing Shock Response Spectrum envelope in Step 4 by Q:

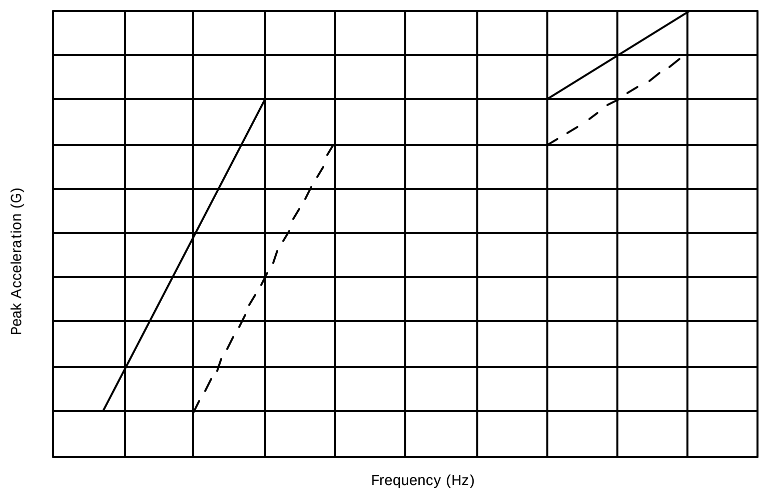

Figure A-2e

Alternatives to the use of swept sine vibration testing are currently under development which address several of the objections to this method. In particular, the problem of excessive resonance build-up in a sinusoidal vibration sweep relative to the flight transient environment may be alleviated by any of the following tests:

- Narrow band swept random.

- Discrete frequency sinusoidal pulses applied at regular frequency intervals.
- Complex waveform pulses representative of a composite of the various launch transient events.

Space microelectromechanical systems should be subjected to a set of swept sinusoidal vibration requirements similar to those specified in the table below. The sine vibration should be applied to the test item by sweeping over a frequency range beginning at 10 Hz (± one octave) up to 100 Hz (± one octave). The frequency range should be swept at a logarithmic rate, such that $\Delta f/f$ is constant. This testing may generally be performed with the same fixturing as a random test, and is often run concurrent with the random vibration trial.

For all tests, these conditions should be applied at interface or mounting surfaces. For structure-like assemblies such as antennas and some large microelectromechanical systems, the input forces may be limited or notches may be applied to the acceleration levels, such that forces at the interface do not exceed spacecraft structural design loads.

Spacecraft-Level		**Assembly-Level**	
Frequency (Hz)	**Level (Gs)**	**Frequency (Hz)**	**Level (Gs)**
5 - 10	1.0 cm DA[1]	5 - 20	1.9 cm DA
10 - 100	2.0 (0 - peak)	20 - 100	12.0 (0 - peak)
100 - 200	1.0 (0 - peak)	100 - 200	3.0 (0 - peak)

SWEEP RATE:
QUAL: 1 OCTAVE PER MINUTE, ONCE UP OR DOWN IN EACH OF THREE ORTHOGONAL AXES.
PF TEST: 2 OCTAVES PER MINUTE, ONCE UP OR DOWN IN EACH OF THREE ORTHOGONAL AXES.
ACCEPTANCE: SAME AS PF.

Table A-3: Sinusoidal vibration.

(4) Random Vibration

The random vibration environment consists of stochastic instantaneous accelerations which are input to a microelectromechanical system or other assembly, transmitted via

[1] DA: Double Amplitude Displacement.

spacecraft structure under launch dynamic excitation conditions. Random vibration input occurs over a broad frequency range, from about 10 Hz to 2000 Hz. In the space vehicle launch environment, random vibration is caused primarily by acoustic noise in the payload fairing, which is in turn induced by external aerodynamic forces due to dynamic pressure and reflection of rocket exhaust from the ground.

For microelectromechanical systems, random vibration can induce a number of failure modes, including fretting in gear trains and breakage of lead-wires in drive electronics. Brinnelling in recirculating bearings can also occur, as the random environment produces the equivalent of micro-shocks in these assemblies.

Random vibration criteria should be developed by the process described in the following four steps:

1. Determine the Power Spectral Density (PSD) of the random vibration directly transmitted into the flight article through its mounts from the launch vehicle sources such as engine firing, turbopumps, etc., as illustrated in the following figure. These vibration conditions at the launch vehicle-to-payload interface are typically available from the launch vehicle developer.

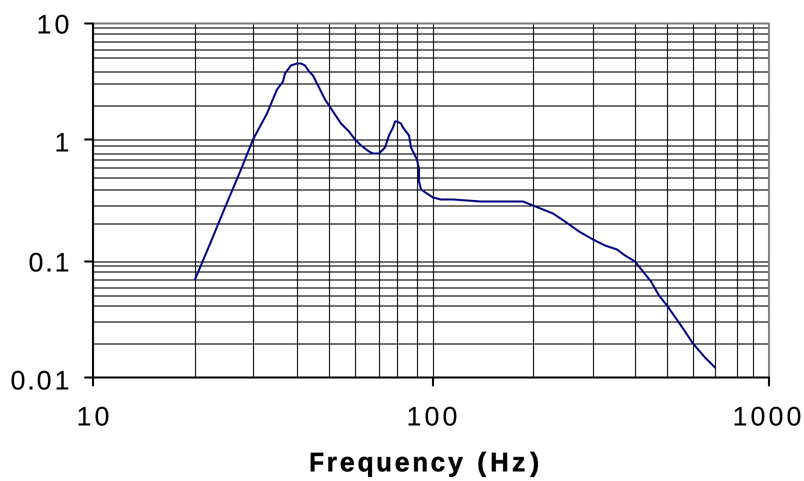

Figure A-3: Vibration levels transmitted to flight article through mounts.

2. Perform an analysis to predict the payload/flight article's vibration response to the launch vibroacoustic environment, as illustrated in the figure below. Statistical energy analysis (SEA) methods such as the VAPEPS (VibroAcoustic Payload Environment Prediction System) program are effective predictors in the higher frequencies. The VAPEPS program can also effectively extrapolate from a database using SEA techniques to provide predictions for a similar

configuration. If random vibration predictions are needed for the lower frequencies, finite element analysis methods, such as NASTRAN, are commonly used. The vibration is induced into the test article both directly and indirectly through its mounting.

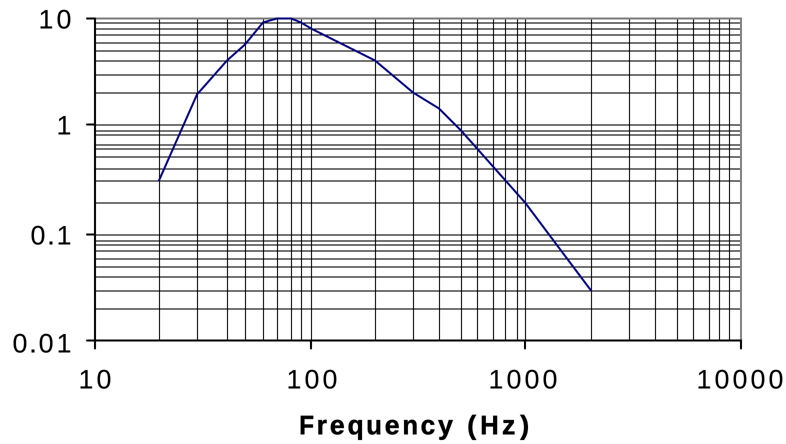

Figure A-4: Payload/flight article response to vibroacoustic environment.

3. Establish a minimum level of vibration which is necessary to ferret out existing workmanship defects and potential failures. The figure below provides such a workmanship vibe level, as specified in MIL-STD-1540.

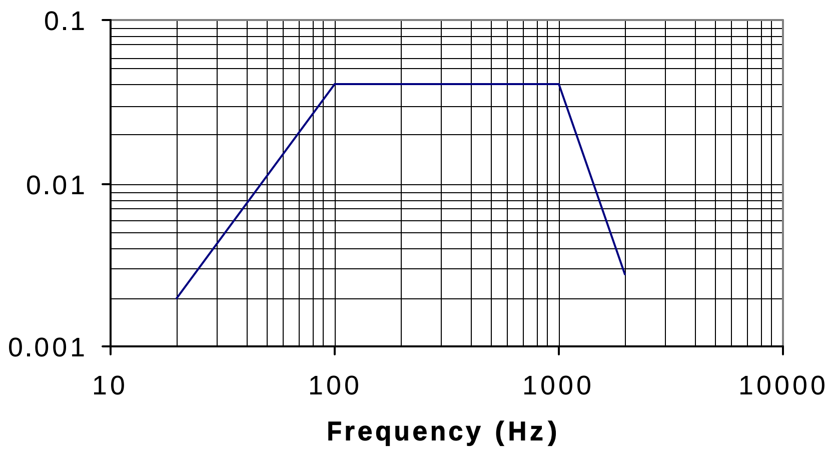

Figure A-5: Minimum vibration levels for workmanship defect detection

4. Envelope the curves from steps 1-3 to produce a composite random vibration specification for the test article, as illustrated below.

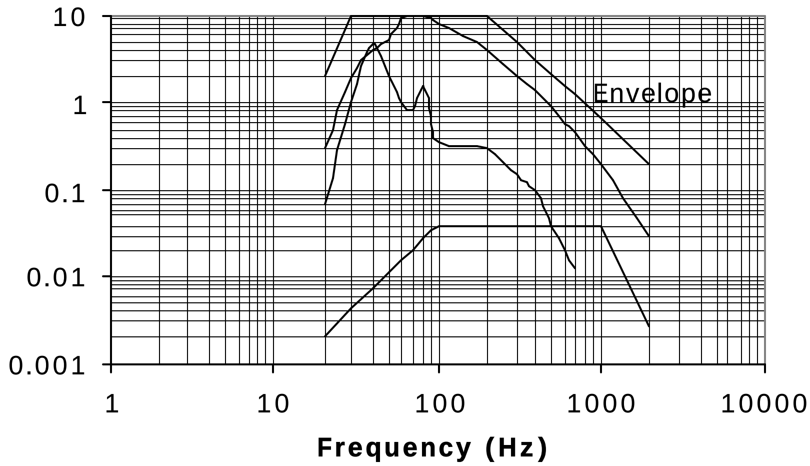

Figure A-6: Composite random vibration envelope.

This resultant random vibration specification, which is employed as the flight acceptance test level, covers the two primary sources of this vibration while also providing an effective process for uncovering workmanship defects. Qualification and Protoflight test levels are increased typically 3 to 6 dB above flight acceptance to verify that the design is not marginal.

Recommended random vibration environments for both spacecraft and assembly-level testing are specified in the accompanying table. Instantaneous accelerations are assumed to exhibit a gaussian distribution. For structure-like assemblies such as antennas and some large instruments, force limit criteria should be used in testing to mitigate the problem of impedance mismatch between the test article and rigid shaker fixture.

Typically, microelectromechanical systems and similar assemblies are mounted to spacecraft structure which is somewhat flexible. If, during a launch event, the MEMS is excited into a state of mechanical resonance, the relatively low stiffness spacecraft mount will serve to limit interface forces. On the other hand, if a microelectromechanical system resonates during a vibration test, the interface forces between shaker and test article can become artificially high, as the infinite impedance shaker continues to drive the resonating mechanical structure to the specification acceleration power spectral density level. To mitigate this problem, the input vibration specification can be notched at resonances or force limiting can be effected. Either way, the interface forces will be limited to more realistic levels, and an unnecessary overtest will be avoided.

Spacecraft-Level		Assembly-Level	
Frequency	**Level**	**Frequency**	**Level**
(Hz)		(Hz)	
20 - 45	+10 dB/octave	20 - 80	+6 dB/octave
45 - 600	0.06 g^2/Hz	80 - 1000	0.25 g^2/Hz
600 - 2000	6 dB/octave	1000 - 2000	-12 dB/octave
Overall	**7.7 grms**	**Overall**	**17.6 grms**

DURATION:
DESIGN: 3 MINUTES IN EACH OF 3 ORTHOGONAL AXES
PF TEST: 2 MINUTES IN EACH OF 3 ORTHOGONAL AXES
ACCEPTANCE SAME AS PF

Table A-4: Random vibration specifications.

Launch Random Vibration Tests are generally applied in each of three orthogonal axes, and have a gaussian distribution of the instantaneous acceleration. Both the Acceleration Spectral Density and wideband acceleration are test parameters and should be within specified tolerances. Each assembly or subsystem should be in its launch configuration. Powered-on vibration of MEMS support electronics, with attendant functional monitoring during testing, should be considered as an effective defect screening tool. All microelectromechanical systems or subsystems should be attached to vibration test fixtures at their normal flight structural interfaces.

Test Control accelerometers should be located at fixture-to-test article interfaces. When more than one control accelerometer is specified, the test should be controlled by averaging the accelerometer signals. Automatic, closed-loop servo control should always be implemented with an electrodynamic vibration exciter.

Vibration Instrumentation for microelectromechanical system testing should include appropriately located accelerometers and strain gages. The accelerometers, strain gages and data acquisition system should have flat frequency response characteristics within $\pm$1 dB from 5 Hz to 2 kHz. Visual data available on site during actual execution of the test should include paper oscillograph recordings of the time histories of the control and selected response channels. Additional quick-look analysis data in the form of Acceleration PSD plots should be available during testing as needed.

(5) Acoustic Noise

Acoustic noise results from the propagation of sound pressure waves through air or other media. During the launch of a rocket, such noise is generated by the release of high velocity engine exhaust gases, by the resonant motion of internal engine components, and by the aerodynamic flow field associated with high speed vehicle movement through the atmosphere.

The fluctuating pressures associated with acoustic energy can cause vibration of structural components over a broad frequency band, ranging from about 20 Hz to 10,000 Hz and above. Such high frequency vibration can lead to rapid structural fatigue. Thus, the objective of a spacecraft acoustic noise requirement is to ensure structural integrity of the vehicle and its components in the vibroacoustic environment. A typical acoustic specification is provided in the figure below.

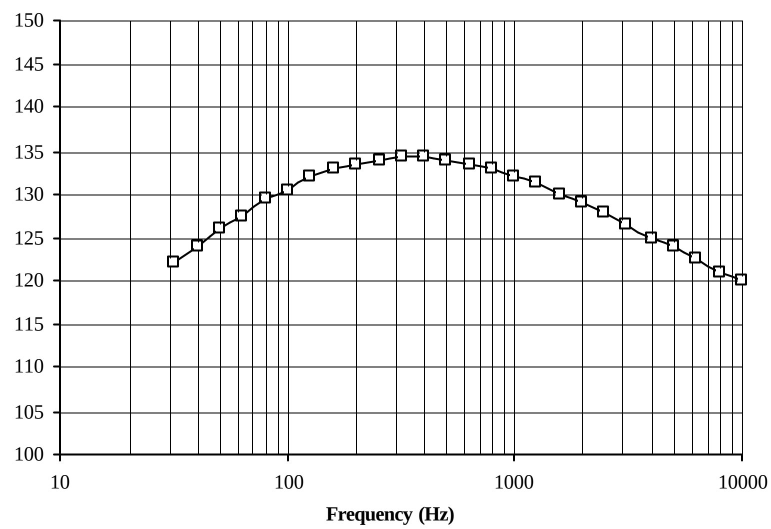

Figure A-7: Typical acoustic noise requirement.

Such a figure specifies the level of input sound pressure over the spectrum of frequencies at which the pressure can fluctuate. The pressure P is measured in decibels, defined as

$$dB = 20\log\left(\frac{P}{P_{ref}}\right)$$

where the reference pressure $P_{ref} = 2 \times 10^{-5}$ Pa is ostensibly the audible limit of the human ear.

The decibel pressure levels in acoustic noise spectra are not generally provided at each and every frequency. Instead, they are often specified over discreet bands of width Δf, which span 1/3 of a frequency octave. With this method, three sound pressure levels will be provided over any interval in which the frequency doubles. The table below is an example of such a 1/3 octave band specification, for the curve data above.

Acoustic Specification	
Center Frequency	SPL (dB)
31.5	122.0
40.0	124.0
50.0	126.0
63.0	127.5
80.0	129.5
100.0	130.5
125.0	132.0
160.0	133.0
200.0	133.5
250.0	134.0
315.0	134.5
400.0	134.5
500.0	134.0
630.0	133.5
800.0	133.0
1000.0	132.0
1250.0	131.5
1600.0	130.0
2000.0	129.0
2500.0	128.0
3150.0	126.5
4000.0	125.0
5000.0	124.0
6300.0	122.5
8000.0	121.0
10000.0	120.0

Table A-5: Acoustic specification table.

When pressure levels are defined with these methods, it is convenient to provide a measure of the overall acoustic noise intensity. The overall sound pressure level, or OASPL, provides just such a measure and, for 1/3 octave band specifications, can be calculated as the decibel equivalent of the root sum square, or RSS, pressure. The table below illustrates such a calculation for the data of the previous example, and shows that the OASPL is 144.9 dB. It should be noted that this OASPL exceeds any individual sound pressure level in the specification, because it represents an intensity of the spectrum as a whole.

Center Frequency	SPL (dB)	Pressure P (Pa)	Squared Pressure
31.5	122.0	25.2	633.9
40.0	124.0	31.7	1004.6
50.0	126.0	39.9	1592.2
63.0	127.5	47.4	2249.1
80.0	129.5	59.7	3564.5
100.0	130.5	67.0	4487.5
125.0	132.0	79.6	6338.7
160.0	133.0	89.3	7979.9
200.0	133.5	94.6	8953.6
250.0	134.0	100.2	10046.2
315.0	134.5	106.2	11272.0
400.0	134.5	106.2	11272.0
500.0	134.0	100.2	10046.2
630.0	133.5	94.6	8953.6
800.0	133.0	89.3	7979.9
1000.0	132.0	79.6	6338.7
1250.0	131.5	75.2	5649.4
1600.0	130.0	63.2	3999.4
2000.0	129.0	56.4	3176.9
2500.0	128.0	50.2	2523.5
3150.0	126.5	42.3	1786.5
4000.0	125.0	35.6	1264.7
5000.0	124.0	31.7	1004.6
6300.0	122.5	26.7	711.2
8000.0	121.0	22.4	503.5
10000.0	120.0	20.0	399.9
		RSS Pressure = 351.8 Pa	
		20 log(351.8/2E-5) = 144.9 dB	

Table A-6: Calculation of overall sound pressure level.

To quantify the acoustic environment, launch vehicles are often instrumented with internal microphones, which measure noise levels within the rocket fairing. This data is telemetered to the ground for processing, and ultimately plotted in the form of a sound pressure level versus frequency spectrum. Since the acoustic forcing function is stochastic, depending on many atmospheric and other variables, data from a number of such flights are generally gathered, and an envelope, such as that of the previous figure, is developed to encompass the historical record of microphone data.

This process can be extended and applied to data from a number of launch vehicles. If a launch platform has not yet been manifested for a particular payload, acoustic profiles from a number of candidate rockets can be enveloped, producing an aggressive specification which will ensure design adequacy for the spacecraft. The figure below reflects such a process, providing an envelope which encompasses the acoustic environments from three launch vehicles.

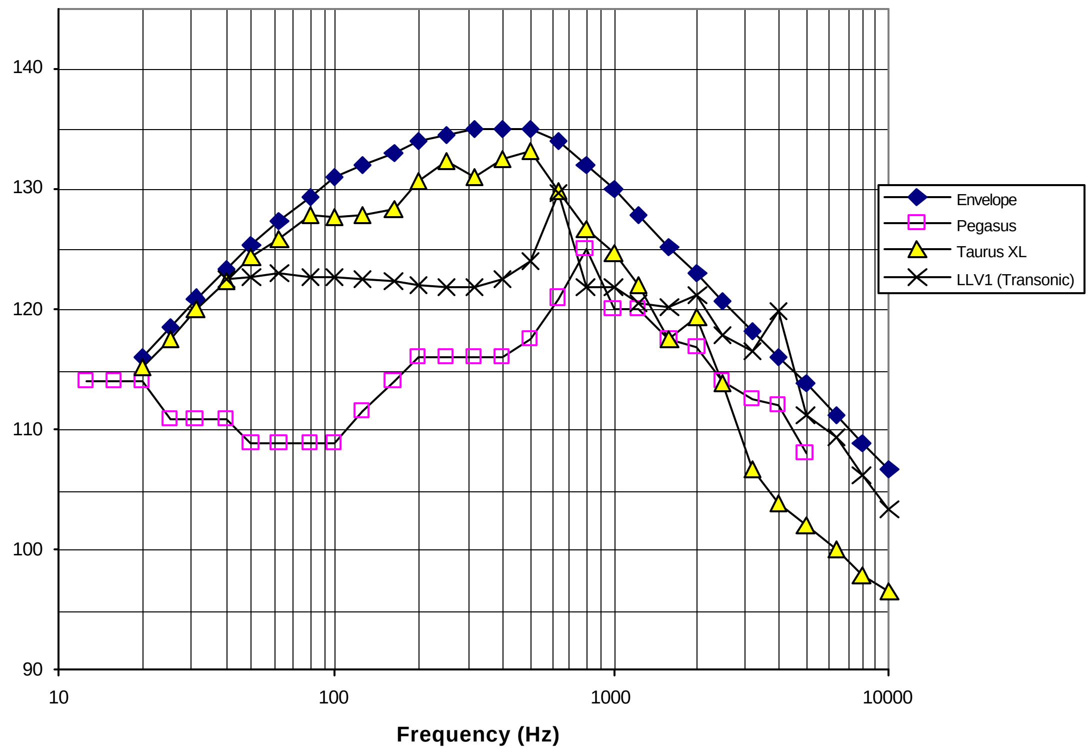

Figure A-8: Envelope of acoustic flight data.

The rationale for acoustic noise testing is straightforward, as acoustic energy is the primary source of vibration input to a space launch vehicle. During the initial phases of a rocket launch, high velocity gases are ejected from motor nozzles and reflected from the ground, creating turbulence in the surrounding air and inducing a vibratory response of the rocket structure. During the subsequent ascent phase of a launch, as the vehicle accelerates through the atmosphere to high velocity, aerodynamic turbulence induces pressure fluctuations which again cause structural vibration. These pressure fluctuations increase in severity as the vehicle approaches and passes through the speed of sound, due to the development and instability of local shock waves. The high-level acoustic noise environment continues during supersonic flight, generally until the maximum dynamic pressure, or max Q, condition is reached.

Acoustic energy gets transmitted to the mission payload in two ways. First, fluctuating pressures within the payload fairing impinge directly on exposed spacecraft surfaces, inducing vibration in high gain antennae, solar panels and other components having a large ratio of area-to-mass. Secondarily, the fluctuating external pressure field causes an oscillatory response of the rocket structure, which is ultimately transmitted through the spacecraft attachment ring in the form of random vibration. From the spacecraft perspective, this random input is generally lowest at the launch vehicle attachment plane, and increases upward along the payload axis.

At the integrated spacecraft level acoustic noise is a primary source of vibration excitation. It should be included in virtually any space vehicle test program. At the subsystem level, however, and particularly in the context of space MEMS, acoustic testing is generally not conducted due to the obvious low ratio of area-to-mass exhibited by a microelectromechanical system.

The failure modes produced by acoustic noise excitation are generally identical to those associated with other types of vibratory structural fatigue. These include failures due to excessive displacement, in which one deflecting component makes contact with another, as well as fractured structural members and loose fasteners. Broken solder joints, cracked PC boards and wave guides can also occur. Electronic components whose function depends on the motion of structural parts, such as relays and pressure switches, are particularly susceptible.

Large flat panels are most easily influenced by, and therefore damaged by, acoustic energy, as they can undergo large displacements while oscillating at low frequency. For a typical spacecraft, this means that a fixed high gain antenna must be carefully designed and stiffened to avoid bending failures, debonding of composite members and related problems. In general, any structure with a high ratio of surface area to mass can be expected to experience potential problems in the acoustic noise environment.

Supporting data for acoustic noise design, analysis and testing can be found in the literature, as well as in various launch vehicle user manuals. The acoustic test has traditionally been severe, with the qualification environment generally established at 4 dB above the expected launch noise profile. The table below provides a sampling of problems detected during acoustic tests on several large programs.

Acoustic Test Problem/Failure History			
Program	**Year**	**Subsystem**	**Failure Mode**
Viking	1973	S/X Band Antenna	Cracked Epoxy
Viking	1973	S/X Band Antenna	Spacers Loosened
Viking	1973	S/X Band Antenna	Studs Loosened
Viking	1973	Infrared Mapper	Wire Shorted
Viking	1973	Radio Antenna	Screw Sheared
Voyager	1977	S/X Band Antenna	Magnetic Coil Debonded
Galileo	1983	Dust Detector	Sensor Cover Buckled
Mars Observer	1991	Telecom Subsystem	HGA Screws Backed Out
Mars Observer	1991	High Gain Antenna	HGA Struts Debonded
Mars Observer	1991	High Gain Antenna	Waveguide Broke
Topex	1992	Instrument Module	I/C Lead Wire Broke
Cassini	1995	High Gain Antenna	HGA Screws Backed Out
Cassini	1995	High Gain Antenna	HGA Struts Debonded

Table A-7: Acoustic test problem/failure history.

The testing has clearly identified improperly designed components. It is interesting to note that a majority of these problems have occurred in high gain antennas and related subsystems, which have the previously identified characteristics of large surface areas, low mass and bonded attachments.

Failure mode sensitivities and cost tradeoffs for the acoustic noise environment are illustrated in the table below. The primary test variables are acoustic noise input level, time duration for the test, frequency of noise input and whether or not power is on in the test article.

Each test parameter in an acoustic noise trial is generally a cost driver. This is primarily due to the fact that the test requires a large chamber, many support personnel and a significant amount of equipment.

Requirement	Control Parameters	Failure Modes	Sensitivity to Increase				Cost	
			dB	tdur	power	f		
Acoustic Noise	dB peak	intermittents	+	+	+	+	dB increase = more N2, etc.	+
	t duration	broken solder joints	+	+	0	-	t duration change	+
	power on	opens	+	+	0	+	power on = extra equipt	+
	frequency	shorts	+	+	0	+	f increase = better modulator	+
		broken connectors	+	+	0	-		
		broken wave guides	+	+	0	-		
		broken crystals	+	+	0	+		
		cracked diodes	+	+	0	+		
		relay chatter	+	+	+	+		
		fastener loosening	+	+	0	+		
		potentiometer slippage	+	+	0	+		

Table A-8: Control parameter sensitivity and cost.

Due to their typically low ratios of area to mass, space MEMS do not require independent testing in the acoustic environment. Instead, such devices are usually subjected only to random vibration, shock and possibly sine testing, with acoustic qualification deferred to the spacecraft level. Nonetheless, the acoustic environment drives many related dynamic specifications, and the informed reader should have some knowledge of common acoustic requirements.

The acoustic noise environment for a typical spacecraft and subassemblies is a reverberant random-incident acoustic field specified in 1/3 octave bands. The cumulative test duration should be no less than 1 minute in any acoustic trial, of which a minimum 35 seconds must be contiguous.

All test items should be in their launch/ascent mechanical and electrical configuration and should be suspended or otherwise positioned within the acoustic chamber such that no major surfaces are parallel to the chamber walls, floor or ceiling, with a minimum of 0.6 m (2 ft) of clearance from any chamber surface. A functional test should be performed before and after the acoustic trial to verify operational performance. Tolerances for SPLs should be as delineated in the table. The OASPL should be controlled to within ±1dB (true RMS) of the specification nominal.

The test should be controlled so that the square root of the average mean-square sound pressure at several locations surrounding the test object meets the test levels specified in the table, in 1/3 octave bands centered on the specified frequency. Test time should commence when the overall control SPL is raised to within 1 dB of that required and should terminate when the level is reduced to more than 1 dB below that required. The control microphone locations should be 12-18 inches from major exterior surfaces of the assembly or subsystem. The control microphones and their data acquisition systems should have flat frequency response characteristics within ±1 dB from 30Hz to 10 kHz.

Frequency	F.A. SPL	Qual SPL	Tolerance
(Hz)	(dB ref 20 mPa)	(dB ref 20 mPa)	(dB)
31.5	129.0	132.0	+6, -3
40	131.0	134.0	+5, -3
50	132.5	135.5	+5, -3
63	134.0	137.0	+5, -3
80	135.0	138.0	+4, -3
100	135.5	138.5	±3
125	136.0	139.0	±3
160	136.0	139.0	±3
200	135.5	138.5	±3
250	135.3	138.3	±3
315	135.0	138.0	±3
400	134.0	137.0	±3
500	132.0	135.0	±3
630	130.5	133.5	±3
800	129.0	132.0	±3
1000	126.5	129.5	±3
1250	125.0	128.0	±3
1600	123.0	126.0	±3
2000	121.0	124.0	±3
2500	119.0	122.0	±3
3200	117.0	120.0	±3
4000	115.0	118.0	±3
5000	113.0	116.0	±3
6400	111.0	114.0	±3
8000	109.0	112.0	±3
10000	107.0	110.0	±3
OASPL	***145.8***	***148.8***	***±1***

Table A-9: Acoustic noise spectra.

The tested assembly or subsystem should be appropriately instrumented with response accelerometers. The accelerometers, in turn, should have flat frequency response characteristics within ±1 dB from 5 Hz to 2 kHz, as should associated data acquisition electronics.

(6) Pyrotechnic Shock

Pyrotechnic Shock is a design and test condition under which flight hardware is subjected to a rapid transfer of energy. The energy transfer is associated with the firing of an explosive device, usually for the purpose of initiating or performing a mechanical action. Spacecraft separation events or the release of propulsion system safing devices are typical such mechanical actions.

A typical pyrotechnic shock requirement is illustrated in the figure below.

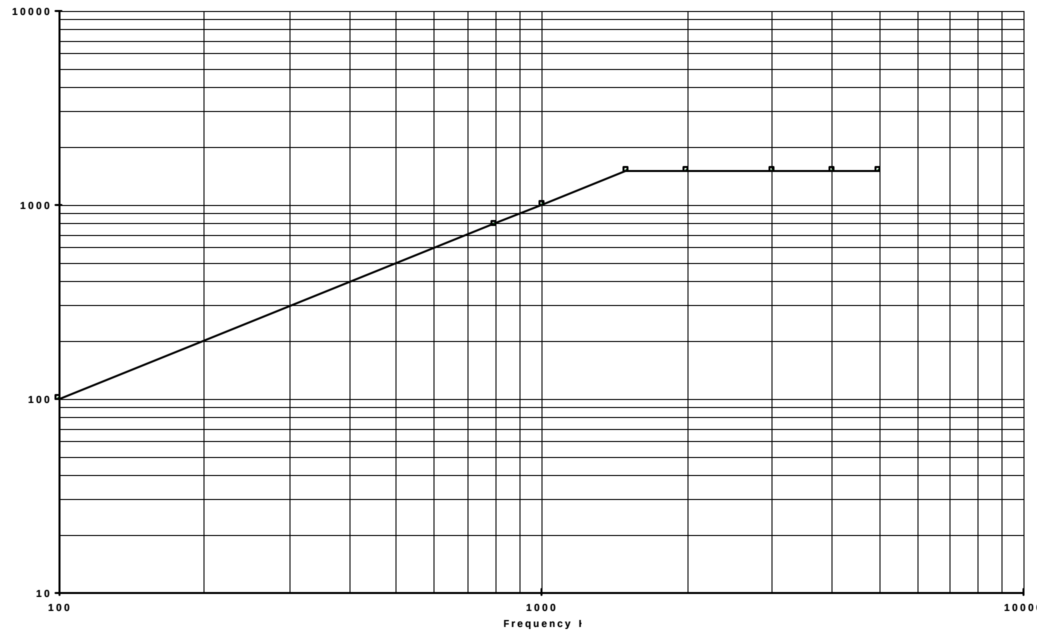

Figure A-9: Typical pyrotechnic shock requirement.

Another possible pyrotechnic shock environment requirement is presented in the following figure. The shock input is applied at the assembly mounting points in each of 3 orthogonal axes.

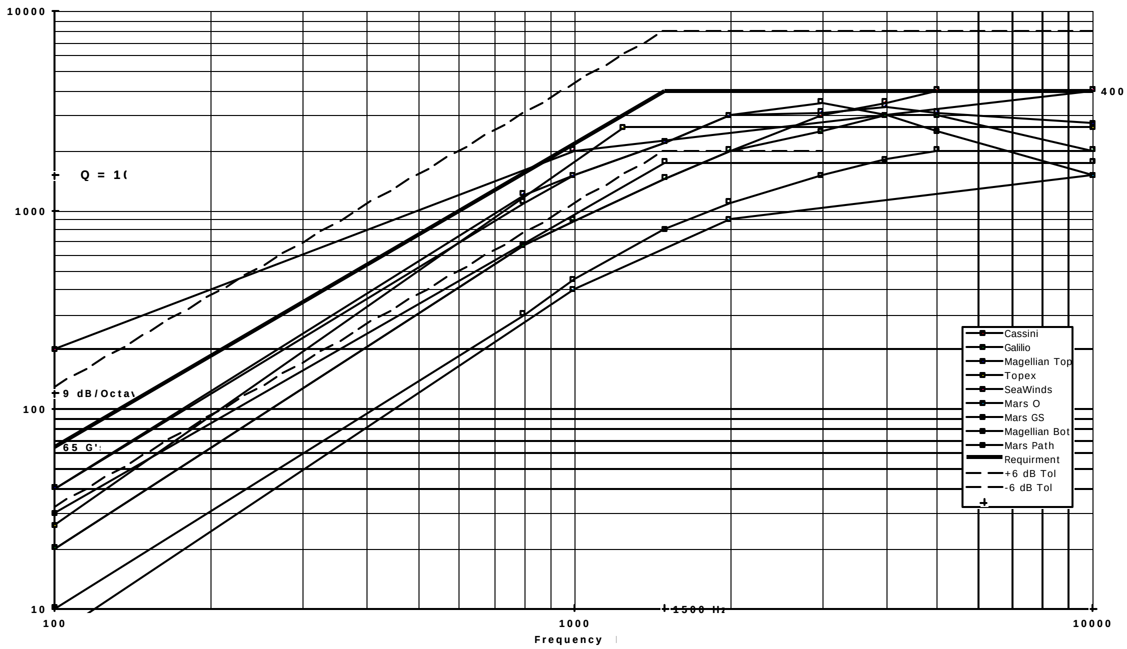

Figure A-10: Subassembly pyrotechnic shock design requirement.

This spectrum represents a 2σ environmental level. It is intended to encompass 95% of all expected shock environments for all available launch vehicles. For reference, shock levels from a number of previous programs are also indicated in the figure.

For test purposes, this environment should be considered a qualification level. Equipment should be exposed to the shock spectrum 3 times in each axis. For devices with self-contained ordinance, 3 self-induced shocks should also be applied.

The release of energy from an ordnance-containing device and the subsequent transfer to the surrounding structure represents a very complex event. As a result, it is difficult to describe the actual shape of the applied shock wave; it is generally not a simple time-based pulse such as a square or triangular wave. The figure below illustrates a typical acceleration versus time trace from an actual pyrotechnic shock event.

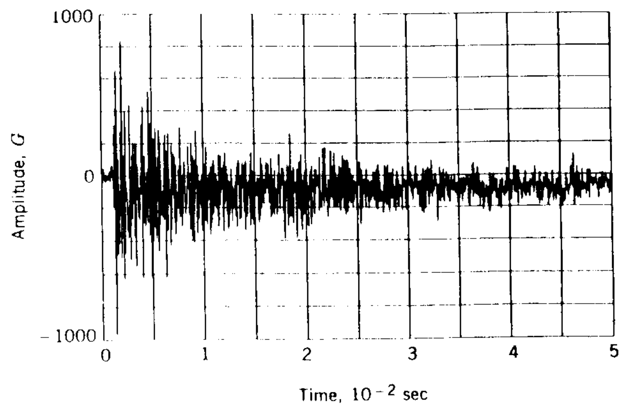

Figure A-11: Pyro shock acceleration time history.

Thus, in establishing a pyro shock requirement, no attempt is made to describe the input pulse, but the frequency-domain response of the structure subjected to the pulse is described instead. The figure below illustrates a typical measurement of this response.

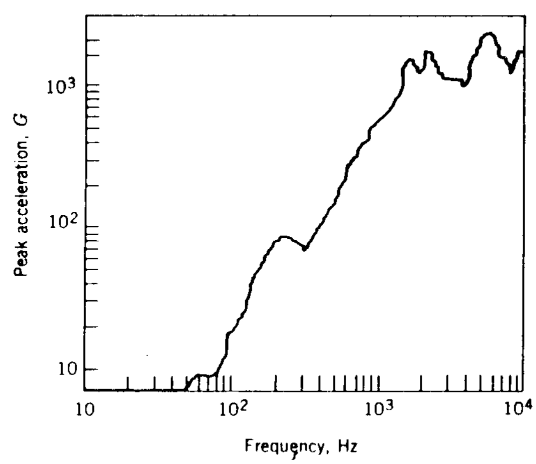

Figure A-12: Frequency response to pyro shock.

The failure modes produced by shock excitation can be broadly grouped into four categories. First are those failures associated with high stresses, such as buckling of long and slender structures, plastic deformation of structures or fracture in brittle components. Next are failures due to high acceleration levels, which can cause relays to chatter, potentiometers to slip and bolts to loosen. Third are problems associated with excessive displacement, which include broken solder joints, cracked PC boards and wave guides, or general problems associated with the impact of one structural component into another. The final category consists of transient electrical malfunctions, which occur only during application of the shock environment. Such malfunctions occur in capacitors, crystal oscillators and hybrids, the latter of which can temporarily short circuit during a shock event due to contact between the device package and internal die bond wires.

Many studies regarding the effects of pyrotechnic shock have been conducted during the life span of the aerospace industry, but one of the best is perhaps that of Moening.[73] Conducted by the Aerospace Corporation under contract to the Air Force Systems Command Space Division, the study examined and summarized ordnance-related shock failures over a period spanning some 20 years, dating from the first missile-related pyro shock failures in the early 1960s to about 1982 when the study was concluded. A total of 85 flight failure events are summarized in the paper, reflecting events ranging from relay chatter, broken electrical wires and leads, cracked glass diodes or fracture of brittle ceramic components and a number of others.

Failure mode sensitivities and cost tradeoffs for the pyrotechnic shock environment need to be discussed in the context of a particular test technique. The three principal methods for shock testing include shaker synthesis, resonant plate testing and actual firing of pyro devices.

In the shaker synthesis technique, the article to be shock tested is mounted to an electrodynamic vibration shaker using an appropriate fixture. A function generator is connected to the shaker, and a series of complex sinusoids or similar time-based pulses are input to the test article in an attempt to generate the desired frequency response spectrum.

Generally, this is a trouble-prone and ineffective exercise because, as stated above, a pyro shock pulse rarely manifests itself as a simple function. Furthermore, the shaker synthesis technique tends to input excessive energy to the structure at low frequencies and insufficient energy at high frequencies. As a result, hardware subjected to such tests is often overtested in the low frequency regime and undertested elsewhere.

In an attempt to improve upon the synthesis method, many environmental test engineers have attempted to modify the input to the shaker using chirp techniques. In this case, output from the function generator is passed through a graphic equalizer before being routed to the shaker. The shaker input spectrum is then tuned through an increase in the gain of high frequency signals, and through an attendant gain reduction at low frequencies. Unfortunately, such efforts offer marginal improvements at best, due to the inherent low-pass filter characteristics of a mechanical shaker.

In the resonant plate technique, advantage is taken of the fact that a stiff, free metal plate can exhibit very high frequency resonances. The article to be tested is mounted to an aluminum or steel plate, and the plate is subsequently suspended in mid-air. A metal pendulum is then swung into contact with the plate, inducing transient vibration. If the frequency response of the mounted test article is measured with an accelerometer, a plot such as that illustrated in the figure below can result.

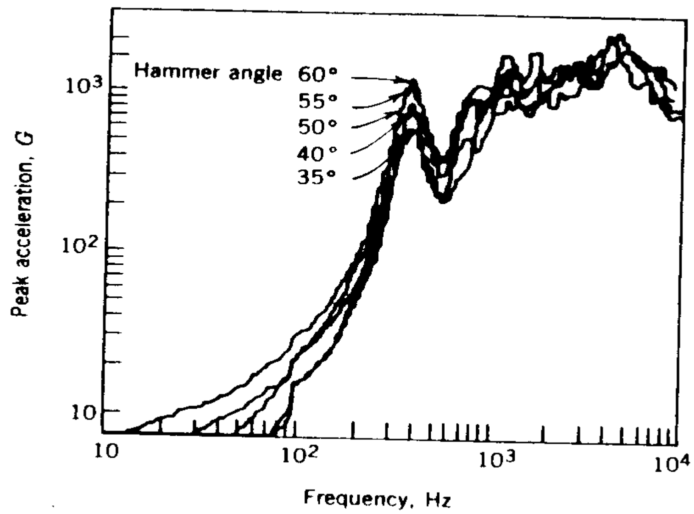

Figure A-13: Response spectrum in resonant plate test.

The Mechanical Impulse Pyro Shock, or MIPS, simulator is a test device which encapsulates the basic resonant plate shock test parameters in a single, relatively compact machine. In a MIPS simulator, an aluminum plate is fabricated and allowed to rest on a foam or plywood pad. The plate is then excited into resonance by the impact of a pneumatic actuator on a moveable bridge.

Shape of the resulting shock pulse is tailorable with a MIPS simulator, by way of experimentation. Dimensions of the resonant plate, the strike location of the hammer and the hammer actuation pressure all affect the resulting shock response spectrum. Interchangeable impactor heads, fabricated from lead, aluminum or steel, are used to alter the duration of the applied pulse.

The MIPS table produces a high fidelity simulation of a pyrotechnic event, in that it generates substantial energy at high frequency in an extremely repeatable manner. The figure below illustrates the basics of MIPS table construction.

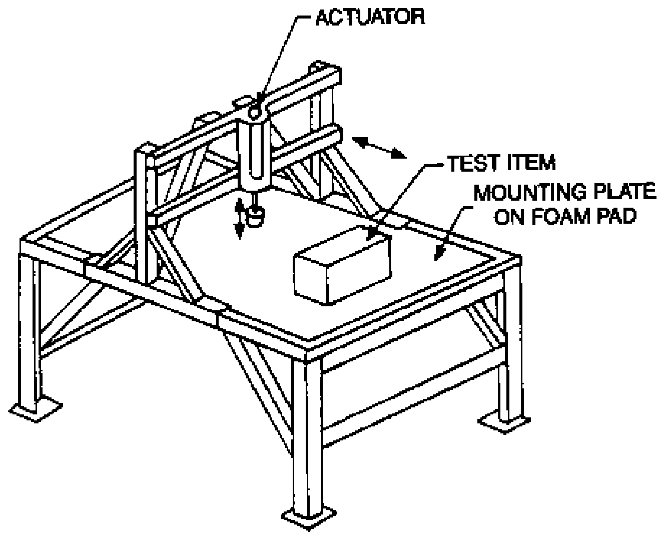

Figure A-14: Mechanical impulse pyro shock simulator.

Although resonant plate techniques can produce a response exhibiting the desired trend of increasing acceleration with increasing frequency, they are still less than ideal. Tuning of the response spectrum such that the correct accelerations occur at the desired frequencies is difficult, involving modification of the plate thickness, shape or suspension method, or modification of hammer characteristics. These activities are time consuming and generally based on trial and error, and do not guarantee generation of the correct response spectrum.

The best pyrotechnic shock test method, then, is one which utilizes pyrotechnic devices. Due to safety, facility and related requirements, this can be an expensive proposition. However, considering the time that might otherwise be wasted during the construct of a simulation, and considering the potential for overdesign or underdesign of hardware which could occur if the simulation is inaccurate, the pyro method may in fact be a bargain. It should be utilized if at all possible.

Armed with our vast knowledge of the primary shock testing methods, we can now present appropriate test control parameters, the sensitivity of failure modes to changes in these parameters, and cost tradeoffs associated with each. The table below provides a summary matrix of this information.

Control Parameters	Failure Modes	Sensitivity to Increase				Cost	
		g	tdur	trise	f	**Shaker Synthesis Method**	
g peak	intermittents	+	-	-	0	g increase = bigger shaker	+
t duration	broken solder joints	+	+	+	-	t duration change	0
t rise	opens	+	-	-	+	t rise redct = better fct gen	+
frequency	shorts	+	-	-	+	f increase = chirp test eqpt	+
	broken connectors	+	-	+	-		
	broken wave guides	+	-	+	-	**Resonant Plate Method**	
	broken crystals	+	-	-	+	g incr = plate/pendlm change	+
	cracked diodes	+	-	-	+	t duration change	0
	relay chatter	+	-	-	+	t rise reduction	0
	fastener loosening	+	-	-	+	f incr = plate/pendlm change	+
	potentiometer slippage	+	-	-	+		
						Pyro Device Method	
						g incr = charge change	+
						t duration change	0
						t rise reduction	0
						f increase	0

Table A-10: Control parameter sensitivity and cost.

Recommended Shock Test

Space microelectromechanical systems and related hardware should be tested to the shock spectrum (Q=10) provided in the table below, and plotted in the accompanying figure.

FREQUENCY (Hz)	ACCEPTANCE (G PK)	PROTOFLIGHT (G PK)
100	40	60
100-1500	9.2 dB per Octave	9.2 dB per Octave
10000	2500	3750

Table A-11: Shock response spectrum (Q=10).

The input shock pulse time history, applied to the base of the test item, should be oscillatory in nature and should decay to less than 10% of its peak value within 50 milliseconds. The spectrum shape should be controlled to within +6/-3 dB, and should be applied in each of three (3) orthogonal axes. At least 30% of spectrum amplitudes should exceed the nominal test specification. Components which are powered-on during spacecraft separation should be shock tested in the powered-on state.

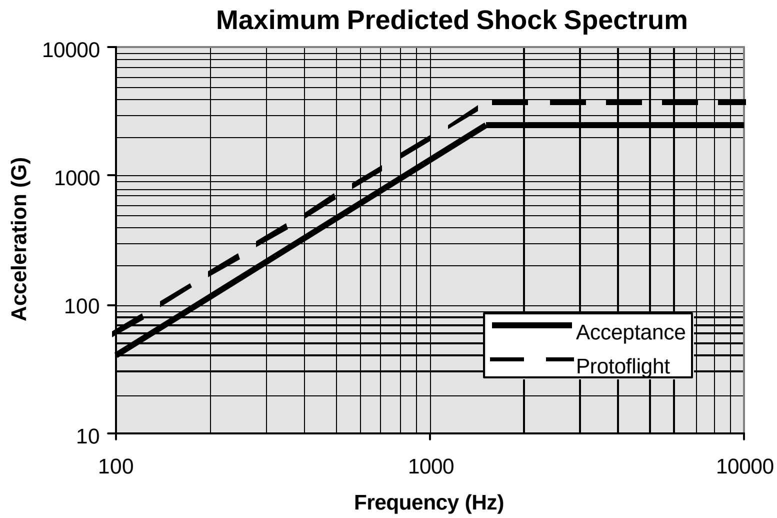

Figure A-15: Shock spectrum.

While dynamics testing is an integral part of preparing MEMS for the space environment, there are a number of other commonly used packaged parts screens as well.