Environmental Conditions for Space Flight Hardware – A Survey

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Introduction

Interest in generalization of the physical environment experienced by NASA hardware from the natural Earth environment (on the launch pad), man-made environment on Earth (storage acceptance and qualification testing), the launch environment, and the space environment, is driven by the need to find commonality among our hardware in an effort to reduce cost and complexity. NASA is entering a period of increase in its number of planetary missions and it is important to understand how our qualification requirements will evolve with and track these new environments.

Environmental conditions are described for NASA projects in several ways for the different periods of the mission life cycle. At the beginning, the mission manager defines survivability requirements based on the mission length, orbit, launch date, launch vehicle, and other factors such as the use of reactor engines. Margins are then applied to these values (temperature extremes, vibration extremes, radiation tolerances, etc.) and a new set of conditions is generalized for design requirements. Mission assurance documents will then assign an additional margin for reliability, and a third set of values is provided for during qualification testing. A fourth set of environmental condition values may evolve intermittently from heritage hardware that has been tested to a level beyond the actual mission requirement. These various sets of environment figures can make it quite confusing and difficult to capture common hardware environmental requirements.

Environmental requirement information can be found in a wide variety of places. The most obvious is with the individual projects. We can easily get answers to questions about temperature extremes being used and radiation tolerance goals, but it is more difficult to map the answers to the process that created these requirements: for design, for qualification, and for actual environment with no margin applied. Not everyone assigned to a NASA project may have that kind of insight, as many have only the environmental requirement numbers needed to do their jobs but do not necessarily have a programmatic-level understanding of how all of the environmental requirements fit together.

For this project, we sought environmental requirement numbers from a variety of sources. These were the project parts engineers (PPEs), safety assurance managers (SAMs), project documentation, and published papers and standards. Our primary resource for space environment information was NASA Memorandum 106655, Operational Environments for Electrical Power Wiring on NASA Space Systems, June 1994 [1]. Vibration and thermal-vacuum test conditions were excerpted from General Environmental Verification Specification (GEVS) for STS and ELV Payloads, Subsystems and Components (http://arioch.gsfc.nasa.gov/302/gevs-se/toc.htm) [2]. The final principal reference was an environments review performed for NEPP in 1992 titled The Influence of Space Environmental Factors on NASA Electrical, Electronic, and Electromechanical Part Selection and Application, A. Garrison, J. Barrows, December 1992 [3].

NASA Memorandum 106655 breaks down all applications into six different types: pressurized modules, low-Earth orbit (LEO), geosynchronous earth orbit (GEO), trans-atmospheric vehicle (orbiter), lunar surface, and Martian surface. (A useful addition to these categories may be deep space as a broad application area). The environmental conditions listed in this document are shown in Table 1 below.

Table 1. List of Attributes Making Up the Environment Suite Experienced by Flight Hardware

Vibration, Sinusoidal Sweep and Random
Shock and Acoustics
Particle Impact (In Space)
Temperature: Extremes and Cycling
Atmospheric Change and Vacuum
Humidity
Ultra Violet Radiation
Ionizing Particle Radiation
Atomic Oxygen
Gravity
Charged Plasma (Or Space Charging)
Combined Environment Tests

Each environmental attribute will be discussed individually in this paper in order to identify the range of stresses that flight hardware is exposed to throughout build and launch and in the actual use environments.

1. Vibration

Vibration is only significant during qualification testing and during launch since there is very little vibration in space (though temperature cycling and current heating has been known to cause bending and work-hardening stress that can mimic mechanical vibration). Vibration requirements are generally defined by the launch vehicle starting quite low at around 3 g's for acceptance testing of payloads on the Pegasus rocket to over 12 g's for payloads on the Shuttle (which experiences both the launch and return vibration periods). Vibration environments are known to subassembly designers and manufacturers in the test requirements. Two types of tests are done for vibration: random and sine directions (x, y, and z), and levels specified for acceptance testing and others for qualification testing. The frequencies sweep range during these tests also varies among launch vehicles in order to model actual frequencies that might stimulate damaging mechanical resonances experienced. Tables 2 and 3 show the test levels used for acceptance and qualification as described in GEVS.

Current NASA GSFC projects are specifying to the higher random vibration requirement levels, between 8.7 g's and 15.8 g's for qualification. Very little information was available on sine vibration requirements actually being used by the projects. Reference [4] cites that if the random vibration test does not show a self-resonance above 140 Hz, then the sinusoidal test can be skipped. Reference [4] also states that the shock and vibration expected in shipping and handling should be controlled to be less than the worst case for the "launch, separation, or mission dynamics environment."

Table 2. Random Vibration Conditions Requirements From GEVS [2]

Launch Vehicle	*Frequency (Hz)*	*ASD Level (G^2/Hz)*	
		Qualification	*Acceptance*
STS	20 to 2,000	12.9 G_{rms}	9.1 G_{rms}
ATLAS	20 to 2,000	2.7 G_{rms}	N/A
DELTA	20 to 2,000	8.7 G_{rms}	N/A
TITAN III	20 to 2,000	4.2 G_{rms}	
SCOUT	20 to 2,000	8.2 G_{rms}	5.8 G_{rms}
PEGASUS	20 to 2,000	4.5 G_{rms}	3.2 G_{rms}
ARIANE	20 to 2,000	7.3 G_{rms}	N/A
TAURUS	20 to 2,000	5.4 G_{rms}	N/A
TAURUS XL	20 to 2,000	5.4 G_{rms}	N/A
TAURUS XLS	20 to 2,000	6.5 G_{rms}	N/A
CONESTOGA	20 to 2,000	4.8 G_{rms}	3.4 G_{rms}
H II (JAXA)	N/A	N/A	N/A
LMLV	20 to 2,000	7.2 G_{rms}	5.1 G_{rms}

Table 3. Sine Vibration Conditions Requirements From GEVS [2]

Vehicle Name	*Thrust Axis*			*Lateral Axis*		
	Frequency (Hz)	*Sine Vibration (G_{0-p})*		*Frequency (Hz)*	*Sine Vibration (G_{0-p})*	
		Qualification	*Acceptance*		*Qualification*	*Acceptance*
DELTA	5~6.2	12.5 mm (0.5/in) DA		5-100	0.7	
	6.2~100	1.00				
TITAN				5~25	7.9 cm/s	
SCOUT	5~43	24 cm/s		25~80	1.25	
	43~100	6.25		80~200	1.5	
	100~200	5				
ARIANE	5~6	1	1	5~6	1	1
	6~100	1.25	1	6~100	1.25	1
H II	5~100	1		5~100	0.8	

2. <u>Shock and Acoustics</u>

Shock requirements originate from conditions experienced in shipping and handling as well as launch and separation of a payload from the launch vehicle. The acoustic stress comes from the cavity geometry between the payload or subsystem and the spacecraft or launch vehicle's fairing walls. GEVS specifies the amount of acoustic blanketing to be used in the cavity and qualification and acceptance test limits. The shock condition is assumed to be very high in energy and frequency but short in duration, and the damage is assumed to occur when a part of the test sample begins to resonate. Without the resonating response, the shock vibration frequency dissipates quickly. Table 4 shows shock and Table 5 shows acoustic vibration requirements that are given in GEVS by launch vehicle. The data collected on project requirements tends towards the levels associated with the Delta rocket.

Table 4. Shock Levels Defined by GEVS [2]

Vehicle	*Frequency (Hz)*	*Shock Levels (G)*	
		Qualification	*Acceptance*
DELTA	350	140	100
	1,700	3500	2500
	4,000 – 5,000	7700	5500
	10,000	2730	1950
TITAN	100	100	70
	100 – 1,250	280	200
	5,000	90	65
SCOUT	100	14	10
	600 – 2,000	280	200
ARIANE[1]	100	25 – 28	18 – 20
	10,000	2800 – 7000	2000 – 5000
H II	100	56	40
	1,500	56	56
	3,000	580	4100

Note:

1. Adapter dependent. See GEVS for intermediate test frequencies and limits.

Table 5. Acoustic Vibration Requirements Defined by GEVS [2]

Vehicle	*Fundamental Frequency Maximum (Hz)*		*Test Range (Hz)*	*Noise Level (dB)*	
	Thrust Axis	*Lateral Axis*		*Qualification*	*Acceptance*
DELTA	35	15	25 – 10,000	142.6	139.6
TITAN	26	15	25 – 10,000	130	136
SCOUT	Not defined	Not defined	50 – 10,000	140	137
ARIANE	31	10	25 – 10,000	145	142
H II	30	10	25 – 10,000	144	141

3. Particle Impact

Particle impact refers to micrometeorite hits during orbit or space travel. The long-duration exposure failing (LDEF) experimental testbed measured micrometeorite impacts in low-Earth orbit in the early 1980s and found that during its 5.75-year exposure, LDEF saw one impact of about 0.7 mm in diameter, per 7 m^2 of exposed surface area in the RAM direction (estimated rate of travel of 10 km/s). This sort of impact is expected to be able to penetrate approximately 2.5 mm of aluminum. Another example of micrometeorite impact observed on the Hubble Space Telescope (HST) solar cell array is shown in Figure 1.

The lunar surface is more exposed to micrometeorites because there is no atmosphere to interfere with the oncoming matter. Impact velocities range from 24 to 72 km/s on the moon.

Table 6 shows particle impact expectations for LEO and other types of missions.

Table 6. Particle Impact by Mission Type [2]

Mission Type	***Conditions***
LEO	11 to 26 impacts/m^2/year.
GEO	Lower than LEO.
Trans-Atmospheric Vehicle	Altitude dependent.
Lunar Surface	0.01 to 1.2×10^{-4} impacts/m^2/year for small- to medium-sized matter, low for larger meteoroids.
Martian Surface	Electrically charged dust particles delivered by dust devils. [8,9] Their charge (created by friction) makes them attach to and seek surfaces such as visors and the inside of non-hermetic cavities to the degree that visibility can be reduced or that operation of the system could be impacted. The flux of micrometeorite hits is insignificant.

No data was collected for actual projects for particle impact requirements, as that requirement is not usually available to the parts engineer and generally parts are not qualified with respect to this environmental condition.

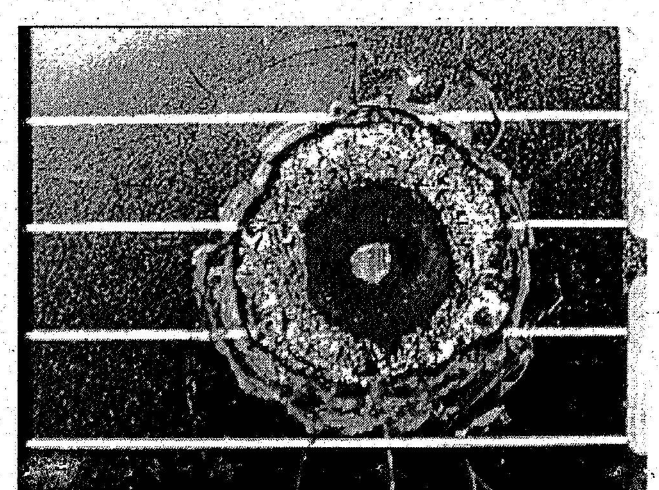

Figure 1. HST Solar Cell Impact Crater (Size 4 mm). The shown crater was observed on the solar arrays of the Hubble Space Telescope that were retrieved in March 2002 from a 600 km orbit. It is likely caused by a micro-meteoroid or space debris particle of 0.5 mm diameter impacting with a velocity of 10-20 km/s. Thousands of such impacts are visible with the naked eye on the 20 m^2 of retrieved solar cells. ESA has initiated a study to systematically analyze the impact features with the aim to identify potential damage resulting from impacts, gain new knowledge on the meteoroids and debris populations, and validate or update the existing flux models.[10]

4. <u>Temperature Extremes and Cycling</u>

Temperature extremes experienced by flight hardware are a combination of their location with respect to the sun or a shadow (planet or spacecraft) or in engineered environments (e.g., a dewar), and the overall thermal management of the spacecraft, which might involve the use of thermal radiators, heaters, or coolers. Table 7 shows temperature extremes and the number of cycles expected on a yearly basis for the different mission categories. [2]

The temperature range in LEO goes from –65 °C to +125 °C with thermal cycling dependent on the orbit height. Temperature conditions in GEO are similar to LEO on the hot side and colder when the spacecraft enters the Earth's shadow about twice a year over a period of 45 days (each time). During these 45-day periods, the spacecraft is in shadow daily for a maximum of 70 minutes. Trans-atmospheric vehicles will experience temperature extremes from –200 °C to +260 °C; however, temperature cycling is not an issue because operation duration in orbit is for a few minutes or hours. The temperature of the lunar surface changes greatly during the day and night, rising up to +111 °C and dropping to –171 °C. Twenty-eight days of the lunar day/night cycle results in 13 cycles in 1 Earth year.

Table 7. Temperature Extremes and Cycling by Mission Type [2]

Mission Type	*Conditions*
Pressurized Module	+18.3 °C to 26.7 °C, cycling is not significant.
LEO	-65 °C to +125 °C, cycles/yr. depending on orbit height (6,000 for a height of 2 km, 780 for 20,000 km).
GEO	-196 °C to +128 °C, 90 cycles/yr.
Trans-Atmospheric Vehicle	-200 °C to +260 °C, cycles altitude dependent and fairly minimum due to short mission duration.
Lunar Surface	-171 °C to +111 °C, 13 cycles/yr.
Martian Surface	-143 °C to +27 °C, 356 cycles/yr.

Mars has four seasons as on Earth. The temperature varies, depending on location, but generally during summer high temperature at the equator ranges from -12 °C to 4 °C, and during the winter polar temperature goes as low as -126 °C. [5] Temperatures recorded by past missions are shown in Table 8. Large changes in temperature in a particular location are believed to be the source of the "wind," which creates dust devils.
Figure 2 shows thermal blanket covering the Cassini spacecraft.

Table 8. Temperatures Recorded by Past Mars Missions [6, 7]

Mission	*Date*	*Recorded min/ max Temperature*[1]
Mariner 4	July 1965	-133 °C to +27 °C
Viking 1 Lander	July 1976	-94 °C to -17 ° C [6]
Mars Pathfinder	July 1997	-80 °C to 0 °C [6]
Spirit Rover	January 2004	-101 °C to +5 °C [7]

Note:
1. Temperatures from references converted to metric units for consistency.

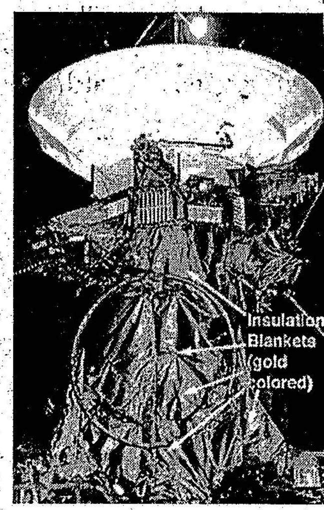

Figure 2. Thermal Blankets Covering the Cassini Spacecraft

The values in Table 7 represent unmanaged environmental temperatures. The values would be much less for electronics in a temperature-managed area, perhaps by as much as half. The temperature requirements increase towards these numbers when standard military specification requirements are used for electronics, namely –55 °C to +125 °C.

PPEs surveyed for the temperature extremes and cycling reported temperature numbers in the range of –50 °C to +50 °C for standard missions and up to –165 °C to +93 °C for an external arm on a GOES spacecraft. In these cases, 12 to 20 cycles were defined.

Reference [3] provides examples of typical operating temperature ranges for mission allowable (actual use) acceptance and qualification testing (Table 9) for several electrical subsystems. These temperatures show that though the mission type may indicate rigorous environmental conditions, spacecraft and subsystem design will tend to provide actual conditions, which are much less severe, even with the margin included in the qualification values.

Table 9. Example Operating Temperature Limits for Selected Electrical Subsystems

Component	*Qualification (°C)*			*Acceptance (°C)*			*Mission (°C)*		
	Min.	*to*	*Max.*	*Min.*	*to*	*Max.*	*Min.*	*to*	*Max.*
Power Control Unit	-23	to	+50	-13	to	+40	-8	to	+35
Battery	-5	to	+35	-5	to	+25	0	to	+20
Solar Cells and Panels	-180	to	+85	-170	to	+75	-165	to	+7
Heater Control Electronics	-35	to	+65	-25	to	+55	-20	to	+50
S-Band Pre-Amp	-20	to	+55	-15	to	+43	-10	to	+45
S-Band Output Mux	-15	to	+50	-5	to	+45	0	to	+4

5. Atmospheric Change and Vacuum

The flight hardware encounters changes in atmospheric pressure during ground testing and launch and operates in a vacuum in space. Though pressure changes and exposure to vacuum have been directly related, on their own, to failure mechanisms (such as discharging between closely spaced conductors during launch and oil-canning of sealed containers), NASA focuses heavily on the combination of temperature and vacuum together, and specifies operational and test requirements closely linking the two. Table 10 shows the in-flight pressure expected for flight hardware in various mission types [2].

Table 10. Pressure Conditions [2]

Mission Type	*Conditions*	*Composition*
Pressurized Module	517 Torr	21% oxygen[1]
LEO	10^{-10} Torr	
GEO	7.5 x10^{14} Torr	
Trans-Atmospheric Vehicle	760 Torr (sea level) to 7.5 x 10^{-14} Torr (interplanetary)	
Lunar Surface	10^{-8} to 10^{-12} Torr	Gases (mainly neon, helium, and hydrogen) exist due to seismic activity but quickly dissipate due to lack of gravity and UV radiation. [11]
Martian Surface	4.4 to 11.4 Torr	95% carbon dioxide, 3% nitrogen, and trace amounts of oxygen and water. [12]

Note:

1. Prior to EVA activity, the vestibule area's atmosphere is enriched to 30% oxygen at 69 kPa.

6. Humidity

Humidity is considered for the "conditions of transportation, storage, the orbiter cargo bay, and the orbiter manned spaces." Humidity is also used as a specialty condition that accelerates certain failure mechanisms in certain types of parts.

The specifications found to date, which explain operational requirements or qualification test criteria for stresses due to humidity during ground or space operations, are shown in Table 11.

Table 11. Requirements for Humidity [2]

Requirement	*Location*	*Value*
Minimum and maximum in flight	Pressurized module	25% to 70%
Assumed maximum for uncontrolled environments	Assembly, storage, and transport to launch facility	100%
Maximum on launch pad	Payload	≤ 50%

7. Ultraviolet Radiation

The ultraviolet spectrum (specifically, the range 0.12 μm and 10 μm) is singled out from the other wavelengths of sunlight irradiated on spacecraft because it contains 99.5% of the total energy of all electromagnetic radiation coming from the Sun. The energy intensity of UV radiation depends on the constitution of the ambient media and is at its highest in a vacuum. Equivalent Sun hours (ESH) per year is used to compare the stress due to UV radiation between various mission types in Table 12 [2]. The energy of UV radiation has been found to alter the composition of materials such as polymers.

Table 12. Level of UV Radiation by Mission Profile [2]

Mission Type	*Conditions*
Pressurized Module	None (shielded)
LEO	2220 to 5800 ESH/yr.[1], energy of 118 W/m^2
GEO	8760 ESH/yr., energy of 118 W/m^2
Trans-Atmospheric Vehicle	See LEO and GEO
Lunar Surface	8760 ESH/yr., energy of 1371 W/m^2
Martian Surface	1656 ESH/yr., energy of 649 W/m^2

Note:

1. ESH = Equivalent Sun Hours

8. Ionizing Particle Radiation

All types of spacecraft will be exposed to ionizing particle radiation consisting of atomic and sub-atomic particles such as protons, heavy ions, alpha particles, and electrons. The origin of these particles is the Sun, via the solar wind, and they are either encountered directly from the Sun's flux or in the Van Allen belts surrounding the Earth where they are trapped in large densities. The number, flux, and energies associated with a mission profile vary highly and are the subject of a large part of the discipline of radiation physics and engineering for space (see http://radhome.gsfc.nasa.gov/radhome/rpo.htm). Information and products related to radiation tolerance levels of 300 kRads and higher are considered sensitive information and are subject to International Traffic in Arms (ITAR) regulations. NASA programs were researched to find current typical mission requirements and few representative examples are shown below. Table 13 lists the results.

Table 13. Radiation Requirements for Current Missions

Project	*TID (kRads)*	*SEU*	*SEL (MeV/mg/cm2)*	*Displacement Damage*
GPM	13.6			
EOS-AURA	10		100	
JWST	50			
AIM	10		80	
THEMIS	66		40	

9. Atomic Oxygen

Atomic (monatomic) oxygen is encountered in low-Earth orbit where the last low-density residual amounts of our atmosphere reside. Monatomic oxygen, created by UV excitation of the 0_2 molecule, is highly reactive with many space hardware materials and can cause erosion due to impact with the fast moving spacecraft. Table 14 shows atomic oxygen concern based on mission profile [2].

Table 14. Atomic Oxygen Concerns [2]

Mission Profile	*Level*
Pressurized Module	N/A
LEO	$10^{11} – 10^{12}$ atoms/m^3, atomic energy from 4.3 to 4.4 eV[1]
GEO	Same as LEO
Trans-Atmospheric Vehicle	See LEO
Lunar Surface	N/A
Martian Surface	N/A

Note:
1. To account for variations in the solar max, testing is done from 10^{20} to 10^{22} atoms/cm^2/year.

10. Gravity

Although we think of the space environment as generally uniform with respect to gravity and at zero gravity, we know that there are non-zero-gravity conditions in forward-moving spacecraft and on planet surfaces. Zero gravity, or reduced value of gravity, has a significant effect on mechanical systems such as spacecraft and instruments and dominates our strategies for thermal management. It is important to recognize that terrestrial applications leverage heavily from the heat conduction through the ambient media of air and the "free" work that gravity provides when it pulls cold air toward hot spots and lighter hot air floats away (convection). A much reduced gravity environment makes these passive thermal management removal methods unavailable. Table 15 shows gravity conditions for various mission types [2].

Table 15. Gravity [2]

Mission Type	*Conditions*
Pressurized Modules	Same as LEO with exceptions due to thrust conditions
LEO	< 10^{-6} to 10^{-3} g
GEO	Same as LEO
Trans-Atmospheric Vehicle	1 to 10^{-6}g
Lunar Surface	0.165 g
Martian Surface	0.38 g

11. Charged Plasma

Ionized particles reside in various levels of the Earth's orbit and can create a host of problems when they interact with the spacecraft, which can be both mechanically and electrically damaging. Careful consideration of charging and degradation due to these charged particles should be considered when designing spacecraft and subsystem grounding schemes and during material selection for exposed surfaces. Table 16 shows charged plasma environment for various mission types [2].

Table 16. Charged Plasma Environment [2]

Mission Type	*Conditions*
Pressurized Module	N/A
LEO	3 x 10^4 to 9 x 10^5 cm^{-3}, energies of 0.1 to 0.2 eV (high-density, low-energy particles)
GEO	0.24 to 1.12 cm^{-3}, 120 to 295 keV (low-density, high-energy particles)
Trans-Atmospheric Vehicle	Will go through range of LEO and GEO (interplanetary travel space is considered the same as GEO for this category)
Lunar Surface	Mostly charged particles from the solar wind and galactic rays
Martian Surface	10^3 to 10^5 atoms/cm^3

12. Combined Environments Tests

In order to qualify assembled instruments and spacecraft for flight, combined environment tests that represent worst-case conditions expected in actual mission life are used. GEVS specifies several different combinations involving temperature, vacuum, and humidity (Table 13). The level of vacuum required is 10^{-5} Torr. Test temperatures to be used are defined by project use temperatures but must provide at least a ±10 °C margin (with an exception in temperature-controlled areas). The number of temperature cycles varies on the hardware level. Spacecraft and payloads are required to have four thermal-vacuum cycles performed with dwells at the temperature extremes for 24 hours. Instrument and subsystems are required to have four thermal-vacuum cycles performed, dwelling at temperature extremes for 12 hours. Eight cycles are required at the component or unit level with dwell times of 4 hours. Table 17 shows GEVS test requirements for combined environments [2].

Table 17. GEVS Test Requirements for Combined Environments

Requirement	Payload or Highest Practicable Level of Assembly	Subsystem including Instruments	Unit/ Component
Thermal-Vacuum	T	T	T
Thermal Balance	T and A	T,A	T,A
Temperature-Humidity (Manned Spaces)	T/A	T/A	T,A
Temperature-Humidity (Descent & Landing)	T/A	T/A	T,A
Temperature-Humidity (Transportation & Storage)	A	A	A
Leakage	T	T	T

T = Test, A = Analysis

Appendix A provides listing of environmental test requirements and test conditions for various projects.

References

[1] Operational Environments for Electrical Power Wiring on NASA Space Systems, M. Stavnes, A. Hammoud, and R. Bercaw, NASA Lewis Research Center, June 1994.

[2] General Environmental Verification Specification (GEVS) for STS and ELV Payloads, Subsystems and Components, http://arioch.gsfc.nasa.gov/302/gevs-se/toc.htm.

[3] The Influence of Space Environmental Factors on NASA Electrical, Electronic and Electromechanical Parts Selection and Application, A. Garrison and J. Barrows, December 1992.

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[7] Mars Daily, http://www.marsdaily.com/2004/040109195713.ov9bww5y.html.

[8] Detecting Dust Devils on Mars, June 24, 2004, http://www.jpl.nasa.gov/news/features.cfm?feature=519.

[9] Martian Dust Devils Could Hamper Future Manned Missions, Keay Davidson, *San Francisco Chronicle*, Jan. 12, 2004, http://sfgate.com/cgi-bin/article.cgi?f=/c/a/2004/01/12/MNGK947BRU1.DTL.

[10] ECSS Space Environmental Standard (ECSS E-10-04), March 2001, http://www.estec.esa.nl/wmwww/WMA/

[11] Apollo 17 Lunar Atmospheric Composition Experiment, http://www.lpi.usra.edu/expmoon/Apollo17/A17_Experiments_LACE.html.

[12] The Martian Atmosphere, http://www.starryskies.com/solar_system/mars_html/martian_atmosphere.html.

Appendix A

Table 1a: Environmental Conditions for Selected Flight Missions

Project	GOES NOP (extreme high)	GOES NOP (nominal)	GOES NOP (extreme low)	Mars Exploration Rovers	Phoenix	AIM	THEMIS
Launch Date				6/10/2003 and 07/07/03	Planned 2007		
Website	http://rsd.gsfc.nasa.gov/goes/	http://rsd.gsfc.nasa.gov/goes/	http://rsd.gsfc.nasa.gov/goes/	http://www.jpl.nasa.gov/missions/current/marsexplorationrovers.html	http://phoenix.lpl.arizona.edu/	http://aim.hamptonu.edu/	
Temperature Range (Parts) degrees C							
	+90	-20 to +50	-193 (80K)	-120 to 85C	-120C to 85C (Qual hardware)	-45 to +45	-50 to +65
Duration of Mission (yrs)							
		10		0.26		3	2
Vibration Random							
Freq. Range Hz		20 - 2000		MEFL + 3dB			20 - 2000
Total G's		12.3				15.8	8.7
Vibration Sine							
Freq. Range Hz		5 - 80					
Total G's		10					
Shock							
F in Hz				MEFL + 3dB		100 & 1000	
Max. G						50 & 3000	
Temp. Cycle							
Range of Temp. (spacecraft or payload level)	-165 to +93	-20 to +60	-165 to +93	-120C to 85C		-50 to +50	
Cycle Number (testing)	12	12	12	3X number of mission/ground		8	
Max Number Cycles (miss life)	20	20	20	300 cycles, 90 days			

Table 1b. Environmental Conditions for Selected Missions

Project	GOES NOP (extreme high)	GOES NOP (nominal)	GOES NOP (extreme low)	Mars Exploration Rovers	Phoenix	AIM	THEMIS
Radiation (w/o design margin)							
TID kRads		1M				10	66
SEL in MeV/mg/cm2				50		80	40
Pressure (Torr)							
				7.6			
Environmental Condition							
Temp							
Humidity							
POC							
	G. Kiernan	G. Kiernan	G. Kiernan	R. Ramesham		J. Osche	J. Osche

Table 2a. Environmental Conditions for Selected Missions

Project	**DAWN**	**EOS-AURA**	**GPM**	**LTMPF (Low Temperature Microgravity Physics Facility)**	**Outer Planets Program, Pluto, LaRC**	**Outer Planets Program, Solar Probe, LaRC**	**Galaxy Evolution Explorer**
Launch Date	Planned 2006						4/28/2003
Website	http://dawn.jpl.nasa.gov	http://aura.gsfc.nasa.gov	http://gpm.gsfc.nasa.gov	http://ltmpf.jpl.nasa.gov			http://www.jpl.nasa.gov/missions/current/galex.html
Temperature Range (Parts) degrees C							
	-79C to +170C						
Duration of Mission (yrs)							
	The Dawn mission will launch in May 2006. It will study Vesta beginning in July 2010, and Ceres beginning in August 2014.						
Vibration Random							
Freq. Range Hz	10-20	20 - 2000			20 - 2000	20 - 2000	
Total G's	0.0322 g2/Hz Qual/PF level	8.7	9.5		16.7	16.7	

Table 2b. Environmental Conditions for Selected Missions

Project	DAWN	EOS-AURA	GPM	LTMPF (Low Temperature Microgravity Physics Facility)	Outer Planets Program, Pluto, LaRC	Outer Planets Program, Solar Probe, LaRC	Galaxy Evolution Explorer
Vibration Sine							
Freq. Range Hz			5 - 50	2.5 G (5-30Hz) and 2.0G (30-100Hz)	5 - 100	5 - 100	
Total G's				9.1 Grms	7.5	7.5	
Shock							
F in Hz	100Hz	100 & 3000	1500-3000	None			
Max. G	15 g	57 & 5740	4100				
Temp. Cycle							
Range of Temp. (spacecraft or payload level)	-35 to 75C						
Cycle Number (testing)							
Max Number Cycles (miss life)							
Radiation (w/o design margin)							
TID kRads	0.3 Krads 9based on 6 months with 100 mils shielding and RDF=2	10	13.6		65	25	5krad(Si) with RDM=2
SEL in MeV/mg/cm2		100					
Pressure (Torr)							
Environmental Condition							
Temp	-79C to +170C			-35 to 75C			
Humidity	>or = 30% or <or = 70%						
POC							
		T. King	R. Williams				