NASA Technical Memorandum 4322

NASA Reliability Preferred Practices for Design and Test

NASA Reliability and Maintainability
Steering Committee

SEPTEMBER 1991

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NASA Office of Safety and Mission Quality
Washington, D.C.

NASA

National Aeronautics and
Space Administration

Office of Management

Scientific and Technical
Information Program

1991

PREFACE

This manual summarizes reliability experience from both NASA and industry, and is intended to reflect engineering principles to support current and future civil space programs.

Reliability must be an integral part of the systems engineering process. Although both disciplines must be weighed equally with other technical and programmatic demands, the application of sound reliability principles will be the key to the effectiveness and affordability of America's space program. Experience with our space programs has shown that reliability efforts must focus on the design characteristics that affect frequency of failure. This manual emphasizes that these identified design characteristics must be controlled through the application of conservative engineering principles.

I strongly encourage the use of this manual to assess your current reliability techniques. The manual should promote an active technical interchange between reliability and design engineering that focuses on the design margins, and their potential impact on maintenance and logistics requirements. By applying these practices and guidelines, reliability organizations throughout NASA and the aerospace community, will continue to contribute to a systems development process that assures that:

- Operating environments are well defined and independently verified.
- Design criteria drive a conservative design approach.
- Design weaknesses evident by test or analysis are identified and tracked.

I intend that this manual should be a dynamic medium for technical communication. Additional practices and guidelines will be published periodically. This manual should be considered a series of technical memoranda for promoting a systematic approach to the reliability discipline. Selective use of these practices and guidelines provides the engineering community with the necessary tools to assure the highest possible degree of success in the Nation's civil space program.

George A. Rodney
Associate Administrator for
Safety and Mission Quality

CENTER CONTACTS

In the preparation of this manual, the dedication, time, and technical contributions of the following individuals are appreciated. Without the support of their individual centers, and their enthusiastic personal support and willingness to serve on the NASA Reliability and Maintainability Steering Committee, the practices and guidelines contained in this manual would not be possible.

All of the NASA Centers were invited to contribute to this manual. The people listed below may be contacted for more information about these practices and guidelines.

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DOCUMENT REFERENCING

The following shows the document numbering system applicable to these practices and guidelines. The example illustrated here is "Part Junction Temperature", Practice No. PD-ED-1204.

P D ED 12 04

1. **Practice**
2. **Design Factors**
3. **Engineering Design**
4. Series **12**
5. Practice **04**

Key to Nomenclature

Position	Code
1.	G - Guideline P - Practice
2.	D - Design Factors T - Test Elements
3.	EC - Environmental Considerations ED - Engineering Design AP - Analytical Procedures TE - Test Considerations & Procedures
4.	x Series Number
5.	xx Practice Number within Series

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I. OVERVIEW

A. PURPOSE

This manual is produced to communicate within the aerospace community design practices that have contributed to NASA mission success. The information presented has been collected from various NASA field centers and reviewed by a committee consisting of senior technical representatives from the participating centers.

B. APPLICABILITY

The information presented in this manual represents the "best technical advice" that NASA has to offer on reliability design and test practices. The practices contained in this manual should not be interpreted as requirements but rather as proven technical approaches that can enhance system reliability. Application of the practices and guidelines is strongly encouraged, but the final decisions regarding applicability resides with the particular program or project office.

The manual is divided into two technical sections. Section II contains reliability practices, including design criteria, test procedures, or analytical techniques that have been successfully applied on previous space flight programs. Section III contains reliability guidelines, including techniques currently applied to space flight projects, where insufficient information exists to certify that the technique will contribute to mission success.

C. DISCUSSION

Experience with NASA's successful extended duration space missions shows that four fundamental elements contribute to high reliability: 1) understanding stress factors imposed on flight hardware by its operating environment; 2) controlling the stress factors through selection of conservative design criteria; 3) appropriate analysis to identify and track high stress points in the design (prior to qualification testing or flight use); and 4) careful selection of redundancy alternatives that will provide the necessary function(s) should failure occur.

This manual is provided to encourage design, test, and reliability engineers to give careful attention to both redundancy and stress management during the design and development of space flight systems.

D. CONTROL/CONTRIBUTIONS

The practices and guidelines contained in this manual serve as a mechanism for communicating the latest techniques that contribute to high reliability. This publication will be revised periodically to include additional practices/guidelines, or revisions to information (as additional technical data become available). Contributions from aerospace contractors and NASA field centers is encouraged. Any practice, guideline, or technique that appears appropriate for inclusion in this manual should be submitted for review. Submissions should be formatted identically to the practices and guidelines in this manual and sent to the address below for consideration:

National Aeronautics and Space Administration
Code QR
600 Independence Avenue, S.W.
Washington, D.C. 20546

Organizations submitting practices/guidelines that are selected for inclusion in this manual will be recognized in the lower right-hand corner of the published item.

II. RELIABILITY PRACTICES

A. INTRODUCTION

This section contains Reliability Design Practices that have contributed to the success of previous space flight programs. The information presented in this section is for use throughout NASA and the aerospace community to assist in the design and development of highly reliable equipment and assemblies. The practices include recommended analysis procedures, redundancy considerations, parts selection, environmental requirements considerations, and test requirements and procedures.

B. RELIABILITY DESIGN PRACTICE FORMAT DEFINITIONS

The format for the reliability practices is shown in Figure 1.

PRACTICE FORMAT DEFINITIONS

Practice: *A brief statement of the practice.*

Benefit: *A concise statement of the technical improvement realized from implementing the practice.*

Programs That Certified Usage: *Identifiable programs or projects that have applied the practice.*

Center to Contact for More Information: *Source of additional information, usually sponsoring NASA Center. See "CENTER CONTACTS", page ii.*

Implementation Method: *A brief technical discussion that is not intended to give the full details of the process, but rather to provide a design engineer with adequate information to understand how the practice should be used.*

Technical Rationale: *A brief technical justification for the use of the practice.*

Impact of Nonpractice: *A brief statement of what can be expected if use of the practice is avoided.*

Related Practices: *Identification of other topic areas in the manual that contain related information.*

References: *Publications that contain additional information about the practice.*

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Figure 1

PRACTICES AS OF APRIL, 1991

PD-EC-1101 Environmental Factors

PD-ED-1201 EEE Parts Derating
PD-ED-1202 High Voltage Power Supply Design and Manufacturing Practices
PD-ED-1203 Class S Parts in High Reliability Applications
PD-ED-1204 Part Junction Temperature

PD-AP-1301 Surface Charging and Electrostatic Discharge Analysis

PT-TE-1401 EEE Parts Screening
PT-TE-1402 Thermal Cycling
PT-TE-1403 Thermographic Mapping of PC Boards
PT-TE-1404 Thermal Test Levels
PT-TE-1405 Powered-On Vibration
PT-TE-1406 Sinusoidal Vibration

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RELIABILITY
PRACTICES

ENVIRONMENTAL FACTORS

Practice:

At the onset of the design process, identify the operating conditions that will be encountered during the life of the equipment.

Benefits:

Each of the identified environmental factors requires consideration in the design process. This assures that adequate environmental strength is incorporated into the design to ensure reliability.

Programs That Certified Usage:

Space Electronic Rocket Test (SERT) I and II, Communication Technology Satellite (CTS), ACTS, Space Experiments, Launch Vehicles, Space Power Systems, and Space Station Freedom.

Center to Contact for More Information:

Lewis Research Center (LeRC)

Implementation Method:

To ensure a reliability-oriented design, determine the needed environmental resistance of the equipment. The initial requirement is to define the operating environment for the equipment. A Life-Cycle Environment Profile, containing this information, should be developed.

A Life-Cycle Environment Profile is a forecast of events and associated environmental conditions that an item experiences from manufacturing to retirement. The life cycle includes the phases that an item will encounter such as: handling, shipping, or storage prior to use; disposition between missions (storage, standby, or transfer to/from repair sites); geographical locations of expected deployment; and platform environments. The environment or combination of environments the equipment will encounter at each phase should be determined. All deployment scenarios should be described as a baseline to identify the environments most likely to be associated with each life cycle phase. The following factors should also be taken into account:

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a. Hardware configuration.

b. Environment(s) that will be encountered.

c. Platform/hardware interfaces.

d. Interfaces with other equipment.

e. Absolute and relative duration of exposure phase.

f. Probability that environmental condition(s) will occur.

g. Geographical locations.

h. Any other information that will help identify environmental conditions that may impact the item.

The steps in developing a Life-Cycle Environment Profile are as follows:

1) Describe anticipated events for an item of equipment, from final factory acceptance through terminal expenditure or removal from inventory.

2) Identify significant natural and induced environments or combination of environments for each anticipated shipping, storage, and logistic event (such as transportation, dormant storage, stand-by, bench handling, and ready modes, etc.).

3) Describe environmental and stress conditions (in narrative and statistical form) to which equipment will be subjected during the life cycle. Data may be derived by calculation, laboratory tests, or operational measurements. Estimated data should be replaced with actual values as determined. The profile should show the number of measurements used to obtain the average value of these stresses and design achievements as well as their variability (expressed as standard deviation).

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This analysis can be used to: develop environmental design criteria consistent with anticipated operating conditions, evaluate possible effects of change in environmental conditions, and provide traceability for the rationale applied in criteria selection for future use on the same program or other programs.

A listing of typical environmental factors is included in Table 1.

TABLE 1: ENVIRONMENTAL COVERAGE CHECKLIST (TYPICAL)

NATURAL	INDUCED
Albedo, Planetary IR	Acceleration
Clouds	Chemicals
Electromagnetic Radiation	Corona
Electrostatic Discharge	Electromagnetic, Laser
Fog	Electromagnetic Radiation
Freezing Rain	Electrostatic Discharge
Frost	Explosion
Fungus	Icing
Gravity, Low	Magnetics
Hail	Moisture
Humidity, High	Nuclear Radiation
Humidity, Low	Shock, Pyro, Thermal
Ice	Space Debris
Ionized Gases	Temperature, High, Aero. Heating, Fire
Lightning	Temperature, Low, Aero. Cooling
Magnetics, Geo	Turbulence
Meteoroids	Vapor Trails
Pollution, Air	Vibration, Mechanical, Microphonics
Pressure, High	Vibration, Acoustic
Pressure, Low, Vacuum	
Radiation, Cosmic, Solar	
Rain	
Salt Spray	
Sand and Dust	
Sleet	
Snow	
Temperature, High	
Temperature, Low	
Wind	

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Technical Rationale:

Given the dependence of equipment reliability on the operating conditions encountered during the life cycle, it is important that such conditions be identified accurately at the beginning of the design process. Environmental factors that strongly influence equipment reliability are included in Table 1, which provides a checklist for environmental coverage (typical).

Concurrent (combined) environments may be more detrimental to reliability than the effects of a single environment. In characterizing the design process, design/test criteria must consider both single and/or combined environments in anticipation of providing the hardware capability to withstand the hazards identified in the system profile. The effects of typical combined environments are illustrated in a matrix relationship in Figure 1, which shows combinations where the total effect is more damaging than the cumulative effect of each environment acting independently. For example, an item may be exposed to a combination such as temperature, humidity, altitude, shock, and vibration while it is being transported. The acceptance to end-of-life history of an item must be examined for these effects. Table 2 provides reliability considerations for pairs of environmental factors.

Each environmental factor that is present requires a determination of its impact on the operational and reliability characteristics of the materials and parts comprising the equipment being designed. Packaging techniques should be identified that afford the necessary protection against the degrading factors.

In the environmental stress identification process that precedes selection of environmental strength techniques, it is essential to consider stresses associated with all life intervals of the equipment. This includes operational and maintenance environments as well as the pre-operational environments, when stresses imposed on the parts during manufacturing assembly, inspection, testing, shipping, and installation may have significant impact on equipment reliability. Stresses imposed during the pre-operational phase often are overlooked; however, they may represent a particularly harsh environment that the equipment must withstand. Often, the environments to which systems are exposed during shipping and installation are more severe than those encountered during normal operating conditions. It is probable that some of the environmental strength features that are contained in a system design pertain to conditions that will be encountered in the pre-operational phase rather than during actual operation.

1 *Combine to intensify mechanical deterioration
2 *Combine to intensify operational deterioration
3 Interdependent (one environment dependent on other)
4 Coexist with no significant combined effect
5 Weakened effect (one environment weakens effects of others)
6 Incompatible
7 Unknown (unlikely combination or indeterminate combined effect)
☐ (Blank) Combination not considered (independent environments)

* A minus sign (-) following number indicates that intensification through combination is weak or doubtful

	Clouds	Fog	Freezing Rain	Frost	Fungus	Geomagnetism	Hail	Humidity	Lightning	Pollution, Air	Rain	Salt Spray	Sand and Dust	Sleet	Snow	Radiation, Solar	Temperature, High	Temperature, Low	Wind	Gravity, Low	Ionized Gases	Meteoroids	Pressure, Low	Radiation, Cosmic	Radiation, Electromag.	Radiation, Van Allen	Acceleration	Explosion	Icing	Radiation, Nuclear	Shock	Temp., High Aero, Heat	Temp., Low, Cryogenic	Turbulence	Vapor Trails	Vibration, Mechanical	Vibration, Acoustic
Clouds																																					
Fog	5																																				
Freezing Rain	3																																				
Frost	5																																				
Fungus																																					
Geomagnetism																																					
Hail	2,1	6																																			
Humidity	3	3	3	3	3																																
Lightning	3	6				2-	2,1																														
Pollution, Air	4	1						1																													
Rain	3	2		6	3	6	1,2	3	4	4																											
Salt Spray		1-	5					5			5																										
Sand and Dust		1-						1-		1-		1-																									
Sleet	3	7					6	3	6																												
Snow	3	7						3	6		2-																										
Radiation, Solar		5		5	5			1-			6	1	1-																								
Temperature, High	5	5	6	6	3			1				1	5	6	6	1																					
Temperature, Low		3	3	3	5			1			1-	5	1-	3	3	5	6																				
Wind	2-	5	7				1,2	7	1,2	3	1,2	1	1,2	1,2	1,2	1	1-	2,1																			
Gravity, Low																4	4	4																			
Ionized Gases					5			1								1	1	5		4																	
Meteoroids																																					
Pressure, Low					7			1								1-	1	1	5	4	3																
Radiation, Cosmic						3										3				7	1-																
Radiation, Electromag.						3														7	1-			3													
Radiation, Van Allen						3										3				4				3													
Acceleration								7								7		1																			
Explosion								1-								7	7	7					1-														
Icing	3	1-	2,1								1-	5				5	6	3	2-																		
Radiation, Nuclear																1-	1-				1		7	7	7	7	7	7									
Shock																		1					1-							7							
Temp., High Aero, Heat																		6											6	7	1						
Temp., Low, Cryogenic																	6													7	1	6					
Turbulence	2-	6					2,1		2,1		2								2,1										2,1			7					
Vapor Trails	5		6				6	3	6	7	6			6	6	4	6	3	5																		
Vibration, Mechanical								1,2				1,2	1,2			1	1	1					1-				1			5		1	1	1			
Vibration, Acoustic																														7		1	1-				

Rows/columns Clouds through Wind: Earth and Lower Atmosphere; Gravity, Low through Radiation, Van Allen: Hyper and Space (both Natural). Acceleration through Vibration, Acoustic: Induced.

Figure 1. Effects of Combined Environments

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TABLE 2: VARIOUS ENVIRONMENTAL PAIRS

High Temperature And Humidity	High Temperature And Low Pressure	High Temperature And Salt Spray
High temperature tends to increase the rate of moisture penetration. The general deterioration effects of humidity are increased by high temperatures.	Each of these environments depends on the other. For example, as pressure decreases, outgassing of constituents of materials increases; as temperature increases, outgassing increases. Hence, each tends to intensify the effects of the other.	High temperature tends to increase the rate of corrosion caused by salt spray.
High Temperature and Solar Radiation	**High Temperature and Fungus**	**High Temperature and Sand and Dust**
This is a man-independent combination that causes increasing effects on organic materials.	A certain degree of high temperature is necessary to permit fungus and microorganisms to grow. However, fungus and microorganisms cannot develop above 160°F (71°C).	The erosion rate of sand may be accelerated by high temperature. However, high temperature reduces sand and dust penetration.
High Temperature and Shock and Vibration	**High Temperature and Acceleration**	**High Temperature and Explosive Atmosphere**
Since both environments affect common material properties, they will intensify each other's effects. The degree to which the effects are intensified depends on the magnitude of each environment in the combination. Plastics and polymers are more susceptible to this combination than metals, unless extremely high temperatures are involved.	This combination produces the same effect as high temperature and shock and vibration.	Temperature has minimal effect on the ignition of an explosive atmosphere but does affect the air-vapor ratio, which is an important consideration.
Low Temperature and Humidity	**High Temperature and Ozone**	
Relative humidity increases as temperature decreases, and lower temperature may induce moisture condensation. If the temperature is low enough, frost or ice may result.	Starting at about 300°F (150°C) temperature starts to reduce ozone. Above about 520°F (270°C), ozone cannot exist at pressures normally encountered.	

ENVIRONMENTAL FACTORS

TABLE 2: VARIOUS ENVIRONMENTAL PAIRS

Low Temperature and Solar Radiation	**Low Temperature and Low Pressure**	**Low Temperature and Salt Spray**
Low temperature tends to reduce the effects of solar radiation and vice versa.	This combination can accelerate leakage through seals, etc.	Low temperature reduces the corrosion rate of salt spray.
	Low Temperature and Sand and Dust	**Low Temperature and Fungus**
	Low temperature increases dust penetration.	Low temperature reduces fungus growth. At sub-zero temperatures, fungi remain in suspended animation.
Low Temperature and Shock and Vibration	**Low Temperature and Acceleration**	**Low Temperature and Explosive Atmosphere**
Low temperature tends to intensify the effects of shock and vibration. However, it is a consideration only at very low temperatures.	This combination produces the same effect as low temperature and shock and vibration.	Temperature has minimal effect on the ignition of an explosive atmosphere but does affect the air-vapor ratio, which is an important consideration.
Low Temperature and Ozone	**Humidity and Low Pressure**	**Humidity and Salt Spray**
Ozone effects are reduced at lower temperatures but ozone concentration increases with lower temperatures.	Humidity increases the effects of low pressure, particularly in relation to electronic or electrical equipment. However, the actual effectiveness of this combination is determined primarily by the temperature.	High humidity may dilute the salt concentration and could affect the corrosive action of the salt by increasing the coverage, thereby increasing the conductivity.
Humidity and Fungus	**Humidity and Sand and Dust**	**Humidity and Solar Radiation**
Humidity helps the growth of fungus and microorganisms but adds nothing to their effects.	Sand and dust have a natural affinity for water and this combination increases deterioration.	Humidity intensifies the deteriorating effects of solar radiation on organic materials.

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TABLE 2: VARIOUS ENVIRONMENTAL PAIRS

Humidity and Vibration	**Humidity and Shock and Acceleration**	**Humidity and Explosive Atmosphere**
This combination tends to increase the rate of breakdown of electrical material.	The periods of shock and acceleration are considered too short for these environments to be affected by humidity.	Humidity has no effect on the ignition of an explosive atmosphere but a high humidity will reduce the pressure of an explosion.
Humidity and Ozone	**Low Pressure and Salt Spray**	**Low Pressure and Solar Radiation**
Ozone meets with moisture to form hydrogen peroxide, which has a greater deteriorating effect on plastics and elastomers than the additive effects of moisture and ozone.	This combination is not expected to occur.	This combination does not add to the overall effects.
	Low Pressure and Fungus	
	This combination does not add to the overall effects.	
Low Pressure and Sand and Dust	**Low Pressure and Vibration**	**Low Pressure and Shock or Acceleration**
This combination only occurs in extreme storms during which small dust particles are carried to high altitudes.	This combination intensifies effects in all equipment categories but mostly with electronic and electrical equipment.	These combinations only become important at the hyperenvironmental levels, in combination with high temperature.
Low Pressure and Explosive Atmosphere	**Salt Spray and Fungus**	**Salt Spray and Dust**
At low pressures, an electrical discharge is easier to develop but the explosive atmosphere is harder to ignite.	This is considered an incompatible combination.	This will have the same combined effect as humidity and sand and dust.

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TABLE 2: VARIOUS ENVIRONMENTAL PAIRS

Salt Spray and Vibration	**Salt Spray and Shock or Acceleration**	**Salt Spray and Explosive Atmosphere**
This will have the same combined effect as humidity and vibration.	These combinations produce no added effects.	This is considered an incompatible combination.
Salt Spray and Ozone	**Solar Radiation and Fungus**	**Solar Radiation and Sand and Dust**
This combination is similar to but more corrosive than humidity and ozone.	Because of the resulting heat from solar radiation, this combination probably produces the same combined effect as high temperature and fungus. Further, the ultraviolet in unfiltered radiation is an effective fungicide.	It is suspected that this combination will produce high temperatures.
Solar Radiation and Ozone	**Fungus and Ozone**	**Solar Radiation and Shock or Acceleration**
This combination increases the rate of oxidation of materials.	Fungus is destroyed by ozone.	These combinations produce no added effects.
Solar Radiation and Vibration		**Sand and Dust and Vibration**
Under vibration conditions, solar radiation deteriorates plastics, elastomers, oils, etc., at a higher rate.		Vibration might possibly increase the wearing effects of sand and dust.
Shock and Vibration	**Vibration and Acceleration**	
This combination produces no added effects.	This combination produces increased effects when encountered with high temperatures and low pressure in the hyperenvironmental ranges.	
Solar Radiation and Explosive Atmosphere		
This combination produces no added effects.		

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Environmental stresses affect parts in different ways. Table 3 illustrates the principal effects of typical environments on system parts and materials.

High temperatures impose a severe stress on most electronic items, since it can cause catastrophic failure (such as melting of solder joints and burnout of solid-state devices). High temperature also causes progressive deterioration of reliability due primarily to chemical degradation effects.[1] It is often stated that excessive temperature is the primary cause of poor reliability in electronic equipment.

In electronic systems design, great emphasis is placed on small size and high part densities. This generally requires a cooling system to provide a path of low thermal resistance from heat-producing elements to an ultimate heat sink of reasonably low temperature.

Solid-state parts are rated in terms of maximum junction temperatures. The thermal resistance is usually specified from this point to either case or to free air. Specification of the maximum ambient temperature for which a part is suitable generally is not a sufficient method for part selection, since the surface temperature of a particular part can be greatly influenced by heat radiation or heat conduction effects from nearby parts. These effects can lead to overheating, even though an ambient temperature rating appears not to be exceeded. It is preferable to specify thermal environment ratings such as equipment surface temperatures, thermal resistance paths associated with conduction, convection, and radiation effects, and cooling provisions such as air temperature, pressure, and velocity. In this manner, the true thermal state of the internal components of temperature-sensitive components can be determined. Reliability improvement techniques for high temperature stress include the use of heat dissipation devices, cooling systems, thermal insulation, and heat-withstanding materials.

Low temperatures experienced by electronic equipment can cause reliability problems. These problems usually are associated with mechanical system elements. They include mechanical stresses produced by differences in the coefficients of expansion (contraction) of metallic and nonmetallic materials, embrittlement of nonmetallic components, mechanical forces caused by freezing of entrapped moisture, stiffening of liquid constituents, etc. Typical examples include cracking of seams, binding of mechanical linkages, and excessive viscosity of lubricants. Reliability improvement techniques for low temperature stress include the use of heating devices, thermal insulation, and cold-withstanding materials.

[1]See Practice No. PT-TE-1404, "Thermal Test Levels/Durations."

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TABLE 3: ENVIRONMENTAL EFFECTS

ENVIRONMENT	PRINCIPAL EFFECTS	TYPICAL FAILURES INDUCED
High temperature	Thermal aging: Oxidation Structural change Chemical reaction Softening, melting, and sublimation Viscosity reduction/evaporation Physical expansion	Insulation failure Alteration of electrical properties. Structural failure. Loss of lubrication properties. Structural failure; increased mechanical stress; increased wear on moving parts.
Low temperature	Increased viscosity and solidification Ice formation Embrittlement Physical contraction	Loss of lubrication properties. Alteration of electrical properties. Loss of mechanical strength; cracking, fracture. Structural failure; increased wear on moving parts.
High relative humidity	Moisture absorption Chemical reaction Corrosion Electrolysis	Swelling, rupture of container; Physical breakdown; Loss of electrical strength; Loss of mechanical strength; Interference with function; Loss of electrical properties; Increased conductivity of insulators.
Low relative humidity	Desiccation Embrittlement Granulation	Loss of mechanical strength; Structural collapse; Alteration of electrical properties, "dusting".
High pressure	Compression	Structural collapse; Penetration of sealing; Interference with function.

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TABLE 3: ENVIRONMENTAL EFFECTS

Low pressure	Expansion Outgassing Reduced dielectrical strength of air	Fracture of container; Explosive expansion. Alteration of electrical properties; Loss of mechanical strength. Insulation breakdown and arc-over; Corona and ozone formation.
Solar radiation	Actinic and physicochemical reactions: Embrittlement	Surface deterioration; Alteration of electrical properties; Discoloration of materials; Ozone formation.
Sand and dust	Abrasion Clogging	Increased wear. Interference with function; Alteration of electrical properties.
Salt spray	Chemical reactions: Corrosion Electrolysis	Increased wear. Loss of mechanical strength; Alteration of electrical properties; Interference with function. Surface deterioration; Structural weakening; Increased conductivity.
Wind	Force application Deposition of materials Heat loss (low velocity) Heat gain (high velocity)	Structural collapse; Interference with function Loss of mechanical strength. Mechanical Interference and clogging; Abrasion accelerated. Accelerates low-temperature effects. Accelerates high-temperature effects.

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TABLE 3: ENVIRONMENTAL EFFECTS

Rain	Physical Stress Water absorption and immersion Erosion Corrosion	Structural collapse. Increase in weight; Electrical failure; Structural weakening. Removes protective coatings; Structural weakening; Surface deterioration. Enhances chemical reactions.
Temperature shock	Mechanical stress	Structural collapse or weakening; Seal damage.
High-speed particles (nuclear irradiation)	Heating Transmutation and ionization	Thermal aging; Oxidation. Alteration of chemical, physical, and electrical properties; Production of gases and secondary particles.
Zero gravity	Mechanical stress Absence of convection cooling	Interruption of gravity-dependent functions. Aggravation of high-temperature effects.
Ozone	Chemical reactions: Crazing, cracking Embrittlement Granulation Reduced dielectrical strength of air	Rapid oxidation; Alteration of electrical properties; Loss of mechanical strength; Interference with function. Insulation breakdown and arc-over.
Explosive decompression	Severe mechanical stress	Rupture and cracking; Structural collapse.
Dissociated gases	Chemical reactions: Contamination Reduced dielectric strength	Alteration of physical and electrical properties. Insulation breakdown and arc-over.
Acceleration	Mechanical stress	Structural collapse.

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TABLE 3: ENVIRONMENTAL EFFECTS

Vibration	Mechanical Stress Fatigue	Loss of mechanical strength; Interference with function; Increased wear. Structural collapse.
Magnetic fields	Induced magnetization	Interference with function; Alteration of electrical properties; Induced heating.

Additional stresses are produced when electronic equipment is exposed to sudden changes of temperature or rapidly changing thermal cycling conditions. These conditions generate large internal mechanical stresses in structural elements, particularly when dissimilar materials are involved. Effects of thermal shock-induced stresses include cracking of seams, delamination, loss of hermeticity, leakage of fill gases, separation of encapsulating materials from components and enclosure surface leading to the creation of voids, and distortion of support members.

A thermal shock test may be specified to determine the integrity of solder joints since such a test creates large internal forces due to differential expansion effects. Such a test also has been found to be instrumental in creating segregation effects in solder alloys leading to the formation of lead-rich zones, which are susceptible to cracking effects.

Electronic equipment often is subjected to environmental shock and vibration during both normal use and testing. Such environments can cause physical damage to parts and structural members when deflections produced cause mechanical stresses which exceed the allowable working stress of the constituent parts.

Natural frequencies of items comprising the equipment are important parameters that must be considered in the design process since a resonant condition can be produced if a natural frequency is within the vibration frequency range. The resonance condition will greatly amplify subsystem deflection and may increase stresses beyond the safe limit.

The vibration environment can be particularly severe for electrical connectors, since it may cause relative motion between members of the connector. In combination with other environmental stresses, this motion can produce fret corrosion. This generates wear debris and causes large variation in contact resistance. Reliability improvement

ENVIRONMENTAL FACTORS

techniques for vibrational stress include the use of stiffening, control of resonance, and reduced freedom of movement.

Humidity and salt air environments can cause degradation of equipment performance since they promote corrosion effects in metallic components. They also can foster the creation of galvanic cells, particularly when dissimilar metals are in contact. Another deleterious effect of humidity and salt air atmosphere is the formation of surface films on nonmetallic parts. These films cause leakage paths and degrade the insulation and dielectric properties of these materials. Moisture absorption by insulating materials also can cause a significant increase in volume conductivity and the dissipation factor of these materials. Reliability improvement techniques for humidity and salt environments include use of hermetic sealing, moisture-resistant material, dehumidifiers, protective coatings/covers, and reduced use of dissimilar metals.

Electromagnetic and nuclear radiation can disrupt performance levels and, in some cases, cause permanent damage to exposed equipment. Therefore, it is important that such effects be considered in determining the environmental strength for electronic equipment that must achieve a specified reliability goal.

Electromagnetic radiation often produces interference and noise effects within electronic circuity, which can impair system performance. Sources of these effects include corona or lightning discharges, sparking, and arcing phenomena. These may be associated with high voltage transmission lines, ignition systems, brush type motors, and even the equipment itself. Generally, the reduction of interference effects requires incorporating filtering and shielding features or specifying less susceptible components and circuity.

Nuclear radiation can cause permanent damage by alteration of the atomic or molecular structure of dielectric and semiconductor materials. High energy radiation also can cause ionization effects that degrade the insulation levels of dielectric materials. The migration of nuclear radiation effects typically involves materials and parts possessing a higher degree of radiation resistance, and the incorporation of shielding and hardening techniques.

Each environmental factor experienced by an item during its life cycle requires consideration in the design process. This ensures that adequate environmental strength is incorporated into the design for reliability.

ENVIRONMENTAL FACTORS

Impact of Nonpractice:

Failure to perform a detailed life cycle environment profile can lead to overlooking environmental factors whose effect is critical to equipment reliability. If these factors are not included in the environmental design criteria and test program, environment-induced failures may occur during space flight operations.

References:

Government

1. MIL-HDBK-217E Notice 1, "Reliability Prediction of Electronic Equipment," January 1990.

2. MIL-HDBK-251, "Reliability/Design Thermal Applications," January 1978.

3. MIL-HDBK-338-1A, "Electronic Reliability Design Handbook," October 1988.

4. MIL-STD-810E, "Environmental Test Methods and Engineering Guidelines," July 1989.

Industry

5. EID-00866, Rocketdyne Division, Rockwell International, "Space Station Freedom Electric Power System Reliability and Maintainability Guidelines Document," 1990.

6. SAE G-11, Society of Automotive Engineers, Reliability, Maintainability, and Supportability Guidebook, 1990.

NASA
PREFERRED RELIABILITY PRACTICES

EEE PARTS DERATING

Practice:

Derate applied stress levels for electrical, electronic, and electromechanical (EEE) part characteristics and parameters with respect to the maximum stress level ratings of the part. The allowed stress levels are established as the maximum levels in circuit applications.

Benefits:

Derating lowers the probability of failures occurring during assembly, test, and flight. Decreasing mechanical, thermal, and electrical stresses lowers the possibility of degradation or catastrophic failure.

Program That Certified Usage:

All Goddard Space Flight Center (GSFC) flight programs

Center to Contact for More Information:

GSFC

Implementation:

EEE parts derating can be established as either design policies or from reliability requirements. In general, NASA has taken the approach of establishing derating policies that cover all applications of the various part types in space flight equipment. These policies are available in MIL-STD-975, "NASA Standard Parts List." Table 1 provides typical derating factors from that document. If derating is to be determined from a reliability requirement, the reference document is MIL-HDBK-217, "Reliability Prediction of Electronic Equipment." MIL-HDBK-217 contains the information necessary to quantitatively estimate the effects of stress levels on reliability.

EEE PARTS DERATING

TABLE 1. TYPICAL PART DERATING GUIDELINES

PART TYPE	RECOMMENDED DERATING LEVEL
Capacitors	Max. of 60% of rated voltage
Resistors	Max. of 60% of rated power
Semiconductor Devices	Max. of 50% of rated power
	Max. of 75% of rated voltage
	Max. junction temperature of 110°C
Microcircuits	Max. supply voltage of 80% of rated voltage
	Max. of 75% of rated power
	Max. junction temperature of 100°C
Inductive Devices	Max. of 50% of rated voltage
	Max. of 60% of rated temperature
Relays and Connectors	Max. of 50% of rated current

NOTE: Maximum junction temperature levels should not be exceeded at any time or during any ground, test, or flight exposure. Thermal design characteristics should preclude exceeding the stated temperature levels.

Technical Rationale:

The reliability of a EEE part is directly related to the stresses caused by the application, including both the environment and the circuit operation. MIL-HDBK-217 contains specific part failure rate models for a wide variety of part types. The models include factors for calculating the effects of various stresses on the failure rate and thus on part reliability. The types of factors include (for example): environment, quality levels, voltage, frequency, and temperature. Given the extensive tables of factors in MIL-HDBK-217, one can formulate reliability predictions for piece parts.

As shown in Figure 1, the plot of piece part failures versus an application stress level such as temperature, voltage, or current indicates decreasing failure rates for lower levels of stress. Therefore, a part's reliability in an application can be increased by

EEE PARTS DERATING

decreasing the maximum allowed stress levels from the absolute maximum for which a part is rated.

Derating policy documents such as those prepared by NASA and DoD, and generally required in their contracts, allow the designer to avoid lengthy and involved calculations by mandating the derating of specific characteristics and parameters.

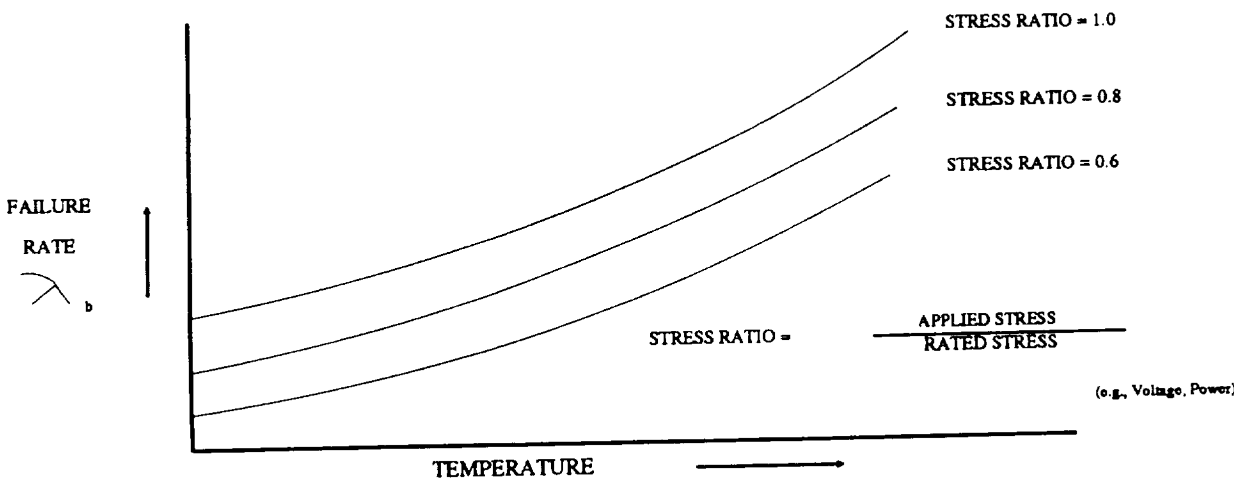


Figure 1. Piece-part Failure Rate vs. Temperature

NASA

PREFERRED RELIABILITY PRACTICES

HIGH VOLTAGE POWER SUPPLY DESIGN AND MANUFACTURING PRACTICES

Practice:

Thoroughly test high voltage power supply packaging on flight configured engineering models, in a simulated space flight environment, to evaluate corona effects.

Benefits:

Process controls on design, manufacturing, and testing operations reduce component failure rates and improve reliability. The goal is production of power supplies that will operate in space for the mission duration.

Programs That Certified Usage:

Space Electronic Rocket (SERT) Tests I and II, Communication Technology Satellite (CTS), 30 cm Thruster Bi-module.

Center to Contact for More Information:

Lewis Research Center (LeRC)

Implementation Method:

There are special requirements in packaging HV power supplies for space use. The power processor should be voltage-partitioned and the low voltage circuits should be separated from the high voltage circuits. This is usually done with a metal wall. There still will be signals transmitted between the sections. All grounds should be isolated to provide a means to predict the currents when transients or arcs occur. When capacitors discharge, there can be current flows of several hundred amperes. The low voltage section should be protected from these current and voltage surges.

Table 1 shows recommended design practices used for an 11 kV CTS TWT power supply. All volumes must be vented. The pressure in any unvented volume will decrease gradually and result in corona or arcing. Allow for screens, RF traps, etc.; and count only the holes in the screens. Interior volumes, down to the capped nut plates, must also be vented.

LEWIS RESEARCH CENTER

TABLE 1. HIGH VOLTAGE POWER SUPPLY DESIGN GUIDELINES

PHYSICAL LAYOUT	
Voltage Partitioning	(Separate high and low voltage components)
Isolated Grounds	(Provide known current path for transients)
Voltage Suppression	(Suppress signal from high voltage to low voltage circuits)
ELECTRIC FIELDS	
Solid Dielectric:	
DC Stress	50 Volts/MIL
AC Stress	10 Volts/MIL
Surface Creepage	8 Volts/MIL
Air or Vacuum Gap	20 Volts/MIL
VENTING	
>2 cm^2 per 1000cc of enclosed volume (screens and RF traps reduce vent size), including:	
	Capped nut plates
	Dielectric spacers
	Polyolefin shrinkable tubing
	High voltage connectors

Figure 1 shows the fabrication methods used to build this supply. Round off all edges on metal as well as dielectric materials. Use anti-corona spheres. Void-free encapsulation is important. Remove excess RTV from bolts to keep vent paths open. Use shrink tubing in strips for hold downs, to avoid trapped air. Dielectric separators must be sized correctly for surface creepage. Anti-corona spheres should have a vent hole to eliminate voids in the solder. Dielectric inserts should be slotted to vent the interior volume.

HIGH VOLTAGE POWER SUPPLY DESIGN AND MANUFACTURING PRACTICES

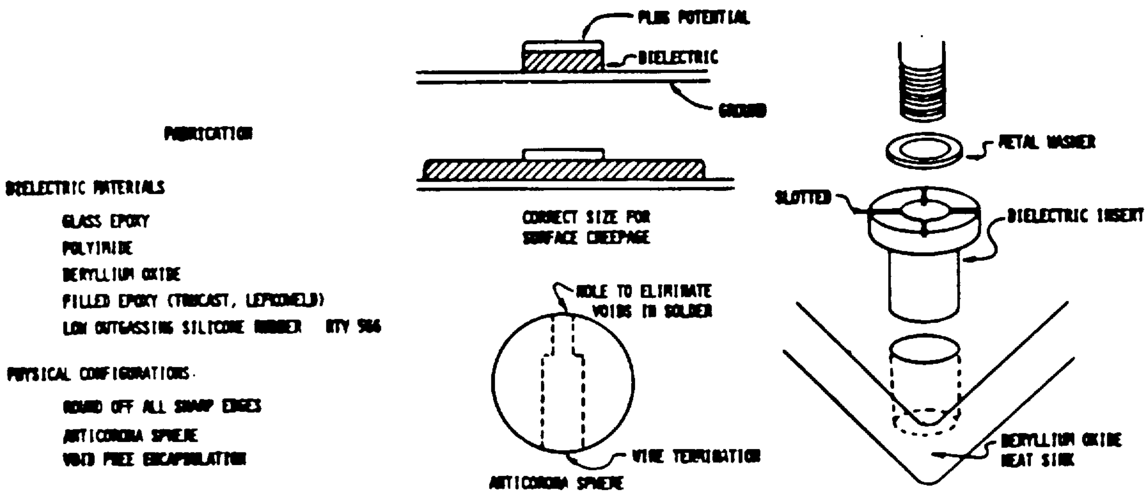
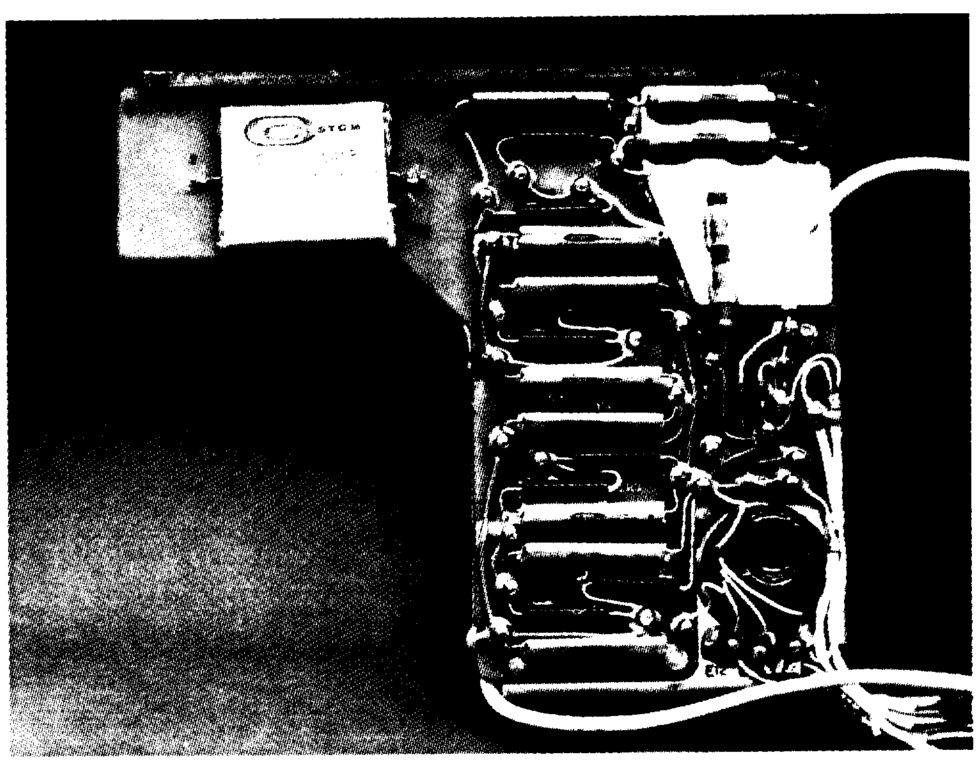


Figure 1. High Voltage Power Supply Fabrication

Figure 2 illustrates the special construction methods used in the HV compartment for the equipment to operate in the thermal and vacuum environment of space.

Figure 2.
HV Compartment
Construction



HIGH VOLTAGE POWER SUPPLY DESIGN AND MANUFACTURING PRACTICES

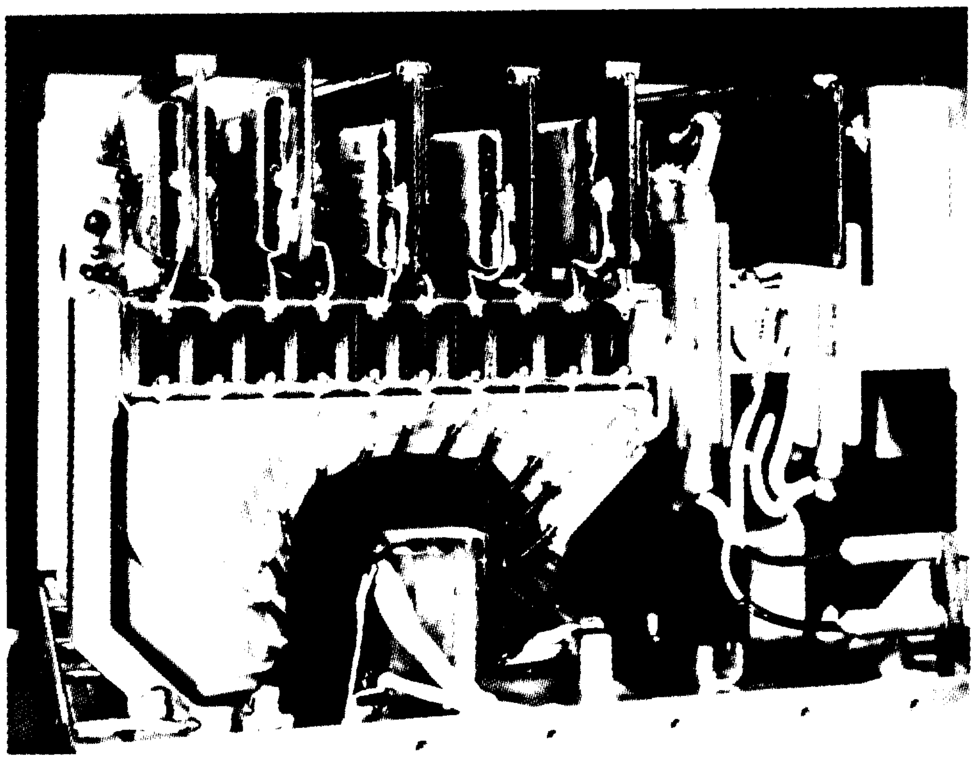


Figure 2. HV Compartment Construction (Continued)

Table 2 shows the testing methods that should be used to check out the HV power supply. The glass epoxy boards should be scanned ultrasonically to check for density differences. The transformers and components mounted on the boards should be corona-tested. Corona discharges of less than 5 picocoulombs are allowed. Induced voltage in the dielectric testing should be done in vacuum at temperature per MIL-T-27. The corona tests should be repeated to detect internal degradation from the high voltage stress. Be careful to bake out the components at 65°C for 72 hours in the vacuum chamber and cool the components down before the power supply is turned on.

HIGH VOLTAGE POWER SUPPLY DESIGN AND MANUFACTURING PRACTICES

TABLE 2. TESTING

Corona Testing:	Transformers Component Configuration on Boards <5 Picocoulombs
Electric Testing:	Induced voltage (twice-rated voltage) Dielectrical withstanding (2.5 times rated AC and DC) Important to perform in vacuum at temperature
Thermal Testing:	Minimum of 10 temperature cycles at component level Minimum of 3 temperature cycles at box level
Initial thermal-vacuum test preceded by bakeout of 65°C for 72 hours	
Ultrasonic Scanning of Glass Epoxy Boards (NASA TM X-73432)	

Technical Rationale:

These design criteria were developed experimentally. The various component configurations, board layouts, and component assemblies were tested to 125% of expected working voltage in air, vacuum, and full operating temperature with a requirement that the corona inception measured less than 5 picocoulombs.

An example of an early flight failure caused by corona was a short that developed between two pins of a high voltage connector. Gas trapped inside connector voids gradually decreased in pressure until corona discharge began to decompose the insulating material. When the insulating material thickness was reduced to the point that leakage started increasing, a carbon tree formed and a short occurred, disabling the experiment. This can be easily avoided by running corona tests on all high voltage parts to ensure that no gases are trapped in high voltage circuits.

Impact of Nonpractice:

Allowing High Voltage power supplies that have not been thoroughly tested for corona to operate in space has resulted in corona-caused shorts that disabled the power supply.

HIGH VOLTAGE POWER SUPPLY DESIGN AND MANUFACTURING PRACTICES

References:

1. MIL-T-27, "Transformers and Inductors (Audio, Power and High-Power Pulse), General Specification for," August 08, 1987.

2. NASA CP 2159, "Spacecraft Transmitter Reliability," September 1979.

3. NASA TMX-3287, Lalli, Vincent R., Nueller, Larry A., and Koutnik, Ernest A., "System Reliability Analysis Through Corona Testing," September 1975. Presented at Power Electronics Specialist Conference (sponsored by IEEE), Culver City, CA (June 9-11, 1975).

4. NASA TMX-73432, Klima, S. J. and P. J. Riley, "Ultrasonic Evaluation of High Voltage Circuit Boards," June 1976.

5. NAS3-17782, Cronin, D. L., "Modeling and Analysis of Power Circuits," TRW Systems Group, June 1975.

6. NAVMAT P4855-1, "Navy Power Supply Reliability Design and Manufacturing Guidelines," December 1982.

7. Foster, W.M., "Thermal Test Report for the Space Acceleration Measurement System Circuit Boards", NASA Lewis Code 6730 Internal Report, November 1987

NASA

PREFERRED RELIABILITY PRACTICES

CLASS S PARTS IN HIGH RELIABILITY APPLICATIONS

Practice:

Use Class S and Grade 1 or equivalent parts in all applications requiring high reliability or long life[1] to yield the lowest possible failure rates.

Benefits:

Low parts failure rates in typical circuit applications result in significant system reliability enhancement. For space systems involving serviceability, the Mean-Time-Between-Failure (MTBF) is greatly extended, which significantly reduces maintenance requirements and crew time demands.

Implementation Method:

Redundancy is an appropriate usage of resources - especially in critical applications to protect against random failures - but is not a justification for using less than Class S or "equivalent" parts. Establish a policy that Class S parts will be used without exception or that limited exceptions are only permitted with extensive testing and inspections for upgrading of Class B to an acceptable level (approximately Class S or Grade 1).

Programs That Certified Usage:

Viking, VGR, and GLL

Center to Contact for Information:

Jet Propulsion Laboratory (JPL)

Technical Rationale:

Basic reliability is a function of parts failure rates. In any analytical calculations of reliability, the usage conditions of parts (derating, temperature, stress, etc.) are expressed as a failure rate that integrates these conditions from empirical or analytical considerations. High reliability parts (Class S or Grade 1) are screened and tested to yield the lowest failure rate parts producible in large quantities (refer to Table 1 for the relationship of Class S to Class B). The failure rates of Class S parts are

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[1] Long life is defined as a requirement to perform the defined function without sacrifice to the primary mission objectives for a period longer than 3 years. Criticality of a function may require high reliability for any period of time and is not necessarily coupled to long life. However, when high reliability is coupled with long life, increased attention to the best reliability design practices is appropriate.

CLASS S PARTS IN HIGH RELIABILITY APPLICATIONS

generally about one fourth of those for Class B. When parts failure rates are coupled into circuit applications, the effects can magnify significantly depending on the circuit configuration.

When the space equipment is not serviceable in a system requiring high reliability and long life, the lowest possible failure rate parts should be selected. This is especially true when considering the economics associated with the launch costs. For example, when changing from Class S to Class B parts, the parts cost decreases by a factor of 4x → 10x but the reliability of the system decreases significantly (by 20 → 50 times in the typical 5-year mission example provided). When total system, mission operations, and launch costs are considered, the delta between the parts costs for Class S and Class B is a minute percentage of total cost. This is especially true for Space Shuttle payloads.

On systems that are serviceable, the MTBF of an assembly is extended in proportion to the basic failure expectation. This significantly longer MTBF reduces on-orbit service requirements with less time demand on the crew, less risk associated with extravehicular activity (EVA), fewer spares required, and fewer launches to transport spares.

Redundancy has a much lower reliability payoff than does parts class – until it is needed. Maverick parts, workmanship flaws, and other uncertainties justify redundancy for critical circuits in high reliability, long life applications to protect against random failures. For long life, the use of high reliability hardware, Class S (or Grade 1) parts, and redundancy in critical applications, provide an optimum and cost-effective approach.

<u>Impact of Noncompliance</u>:

A typical radio and digital subsystem, for a flight instrument with a 3-year mission, has been analyzed considering no redundancy (except TWTs) and partial redundancy in critical circuits for both Class S and B parts.

The parts count method provided in MIL-HDBK-217E was applied. These calculations are not considered accurate for any usage in an absolute sense, due to other design and test factors the data base cannot estimate. However, relative comparisons are very useful and accurate for tradeoff studies of effects of redundancy and parts classifications.

The data are presented in graphical format for ease of understanding. On each plot, the basic reliability for the assumed conditions is plotted on the left ordinate, years are plotted on the abscissa, and the ratio of the analyzed condition is plotted on the right ordinate.

In a single-string (nonredundant) design (Figure 1), the decline in system reliability over time is much less for a system built entirely of Class S parts than if it were built of Class B parts. The ratio of the two reliabilities for a 5-year mission indicates the system built of Class S parts is 50 times more reliable than the system built of Class B parts.

When critical system circuits are made redundant, the time dependent reliability with both Class S and B parts is improved, but the improvement for the system built with B parts is greater (Figure 2). However, the 5-year mission reliability for the system built with B parts is still 20 times less than for the system built with Class S parts.

A correlation is made between the single string (nonredundant) system built with Class S parts and the system with redundant critical circuits and Class B parts (Figure 3). In this correlation, it is clear that for a 5-year mission the single-string system with Class S parts was still 10 times more reliable than the system with redundancy made from Class B parts.

This example reflects that the payoff in reliability is significant for Class S parts compared to Class B parts (for a 5-year mission, Class S is 20 → 50 times more reliable depending on redundancy). Additionally, the return on reliability, addressing non-random failures, is higher for Class S parts than for redundancy used with Class B parts. The highest reliability is obtained with Class S parts with redundancy in the critical circuits.

CLASS S PARTS IN HIGH RELIABILITY APPLICATIONS

TABLE 1. THE DIFFERENCE BETWEEN CLASS S AND CLASS B PARTS

ISSUE	CLASS S	CLASS B	IMPACT
Wafer lot acceptance	Required	--------------	Uniformity and pedigree traceability
Certification of production facilities	To specific assembly lines	To technologies and general facilities only	Burn-in and screening value relates to consistency of original product
Precap internal inspection	100%	Sampled	Significant driver on level of reliability - criteria much more stringent in MIL-M-38510H
PIND for loose particle detection	Required	--------------	Loose metallics in zero g field can cause failures
Serialization	Required	--------------	Traceability lost
Interim electrical test between test phases	Required	--------------	Potential of passing over problems and their causes
Burn-in	240 hours	160 hours	Later problem discovery
Reverse bias burn-in	Required	--------------	Impurity migration not detected
Interim electrical test after reverse bias burn-in	Required	--------------	Effects of reverse bias burn-in may be masked by subsequent actions
Radiographic inspection	Required	--------------	Observation of latent defects
Nondestructive 100% bond pull test	100%	Sampled	Parts with mechanical deficiencies get into equipment

CLASS S PARTS IN HIGH RELIABILITY APPLICATIONS

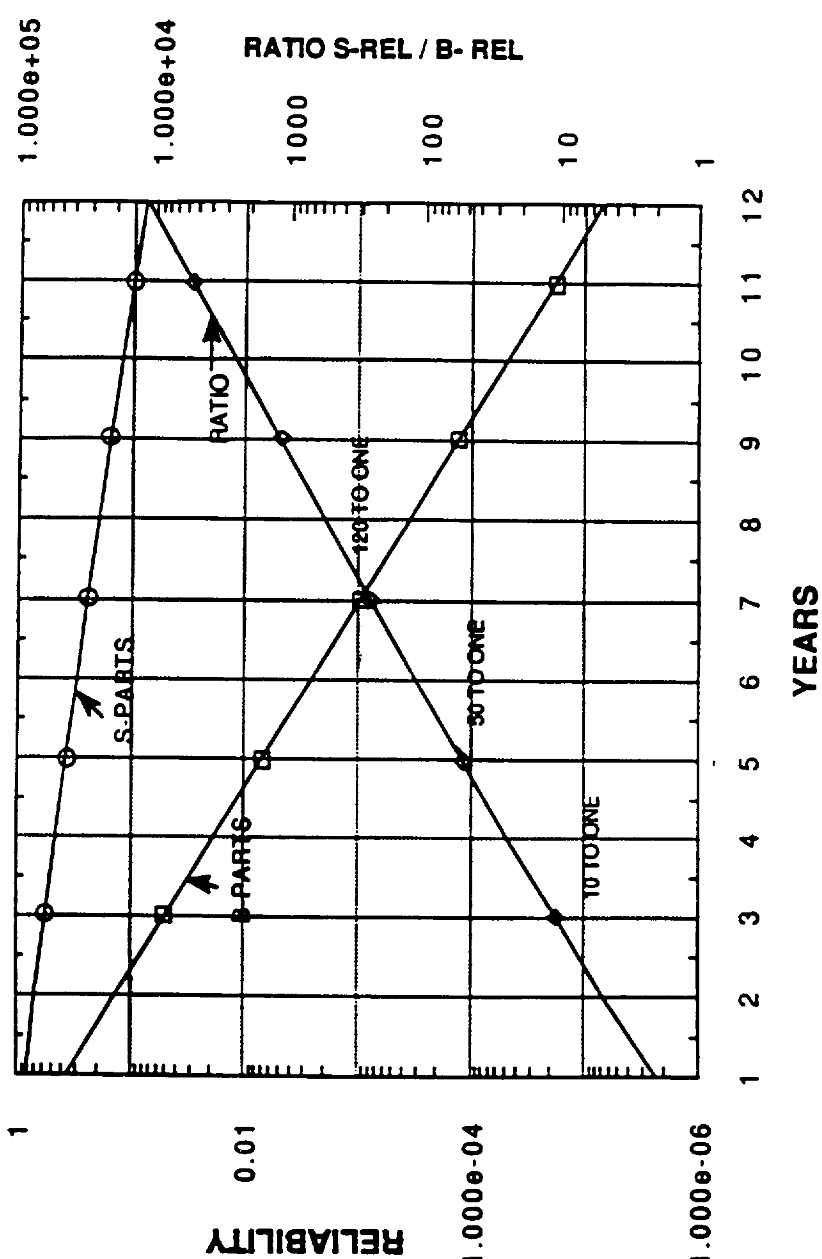


Figure 1 Reliability of Nonredundant System (Except for TWTA)

CLASS S PARTS IN HIGH RELIABILITY APPLICATIONS

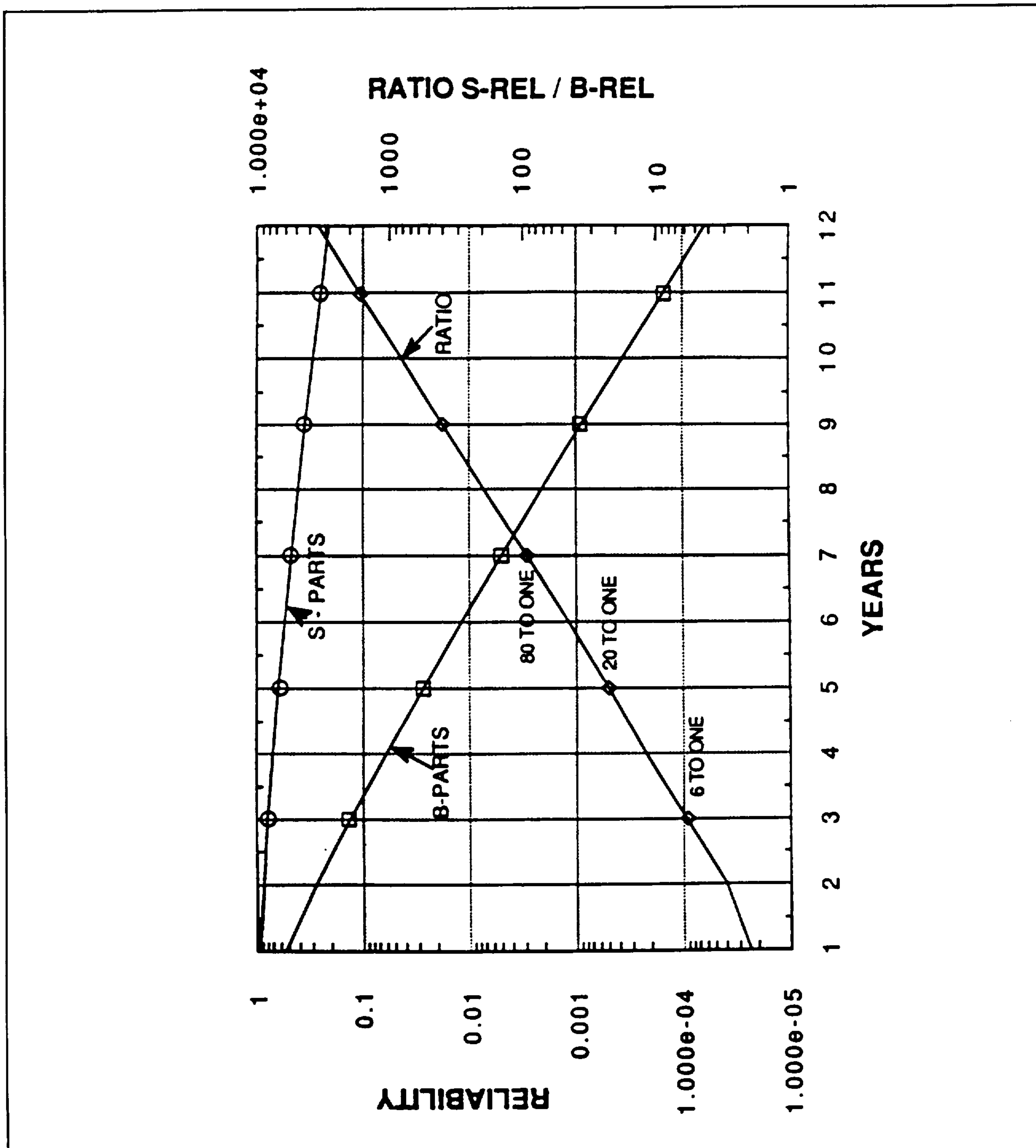


Figure 2 Reliability for Selectively Redundant Systems (for Critical Circuits)

CLASS S PARTS IN HIGH RELIABILITY APPLICATIONS

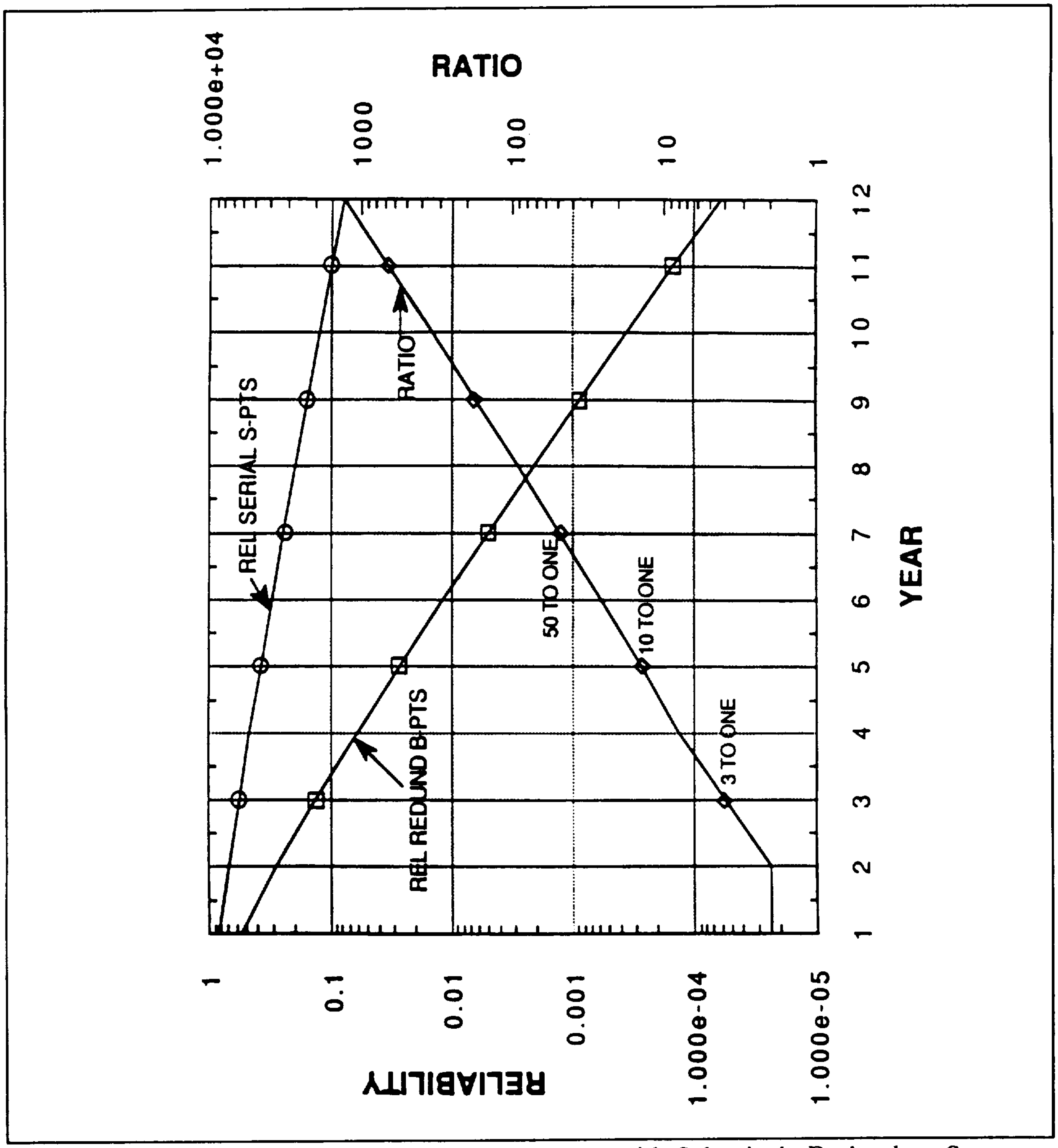


Figure 3 Comparison of Serial System with S Parts with Selectively Redundant System with B Parts (for Critical Circuits)

NASA
PREFERRED
RELIABILITY
PRACTICES

PART JUNCTION TEMPERATURE

Practice:
Maintain part junction temperatures during flight below 60^0C. (Short-term mission excursions associated with transient mission events are permissible.)

Benefit:

Reliability is greatly increased because the failure rate is directly related to the long-term flight temperature.

Programs That Certified Usage:

VGR, GLL, Viking, Mariner series

Center to Contact for Information:

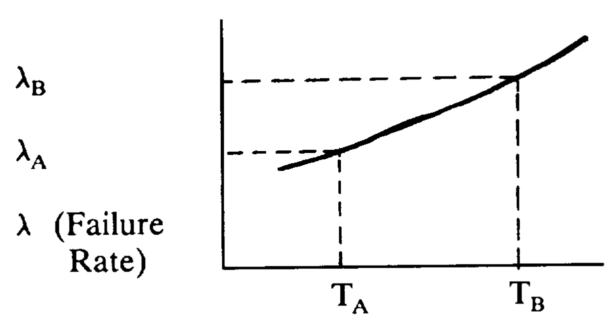
Jet Propulsion Laboratory (JPL)

Implementation Method:

Establish in-specification design (and test) temperatures $\geq 75^0$C and limit part junction temperatures (J_T) to $\leq 110^0$C[1] which constrains permissible part junction temperature rise (ΔJ_T) to $\leq 35^0$C.

Technical Rationale:

Basic reliability is directly related to temperature and time, i.e., $\lambda = f(T,t)$. The following relationship is obtained either theoretically from the Arrhenius relationship ($\lambda = A\exp[-E_a/k\ (1/T - 1/T_0)]$) or empirically from the data in MIL-HDBK-217E.



Given:
. Specific part
. Specific derating factor
. Specific chemical activation energy

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[1]This practice has been verified on programs in place before the release of MIL-STD-975H. If the MIL-STD-975H junction temperature of 100^0C is used, junction temperature rise should be changed to assure that long-term flight junctions stay below 60^0C.

PART JUNCTION TEMPERATURE

The curve shape is representative of all electronic parts (and most mechanical processes) in the range of temperature typified by space exposure. Simply stated, the higher the long-term flight temperatures, the lower the reliability:

$$\frac{B\ failures}{A\ failures} = \frac{\lambda_B}{\lambda_A}$$

Assume that a design and test temperature of 75^{0}C is chosen. In the figure from MIL-STD-883B reproduced on page 4, observe that a 25^{0}C ΔT corresponds to a failure rate increase of more than an order of magnitude, i.e., >1000% difference. MIL-HDBK-217E has different values but the factor is up to approximately 3X on some parts (depends on derating criteria and parts qualification). The following example illustrates the effect of this relationship to design and test temperatures.

Assume the following conditions as an example:

Case A: T = 75^{0}C in-specification design temperature for baseplate

Case B: T = 50^{0}C in-specification design temperature for baseplate

Case A and Case B:

T = 25^{0}C long-duration flight temperature for baseplate

J_T = 110^{0}C limit for any exposure or analysis

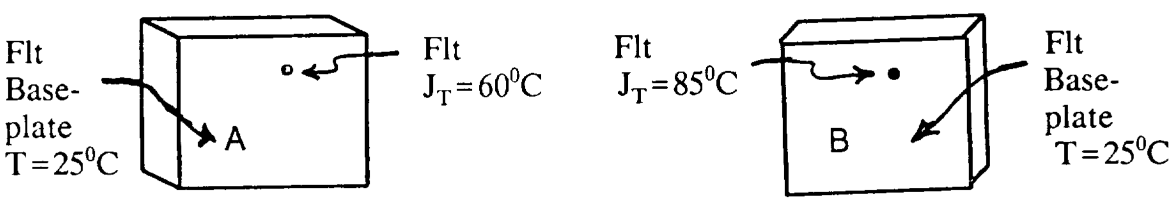
PART JUNCTION TEMPERATURE

Then:

Design/Test Parameters

	Case A	Case B
Design Baseplate	75^0C	50^0C
J_T limit	110^0C	110^0C
Permitted ΔJ_T rise	35^0C	60^0C

Flight Conditions



$\Delta T = 25^0C$

from Arrhenius

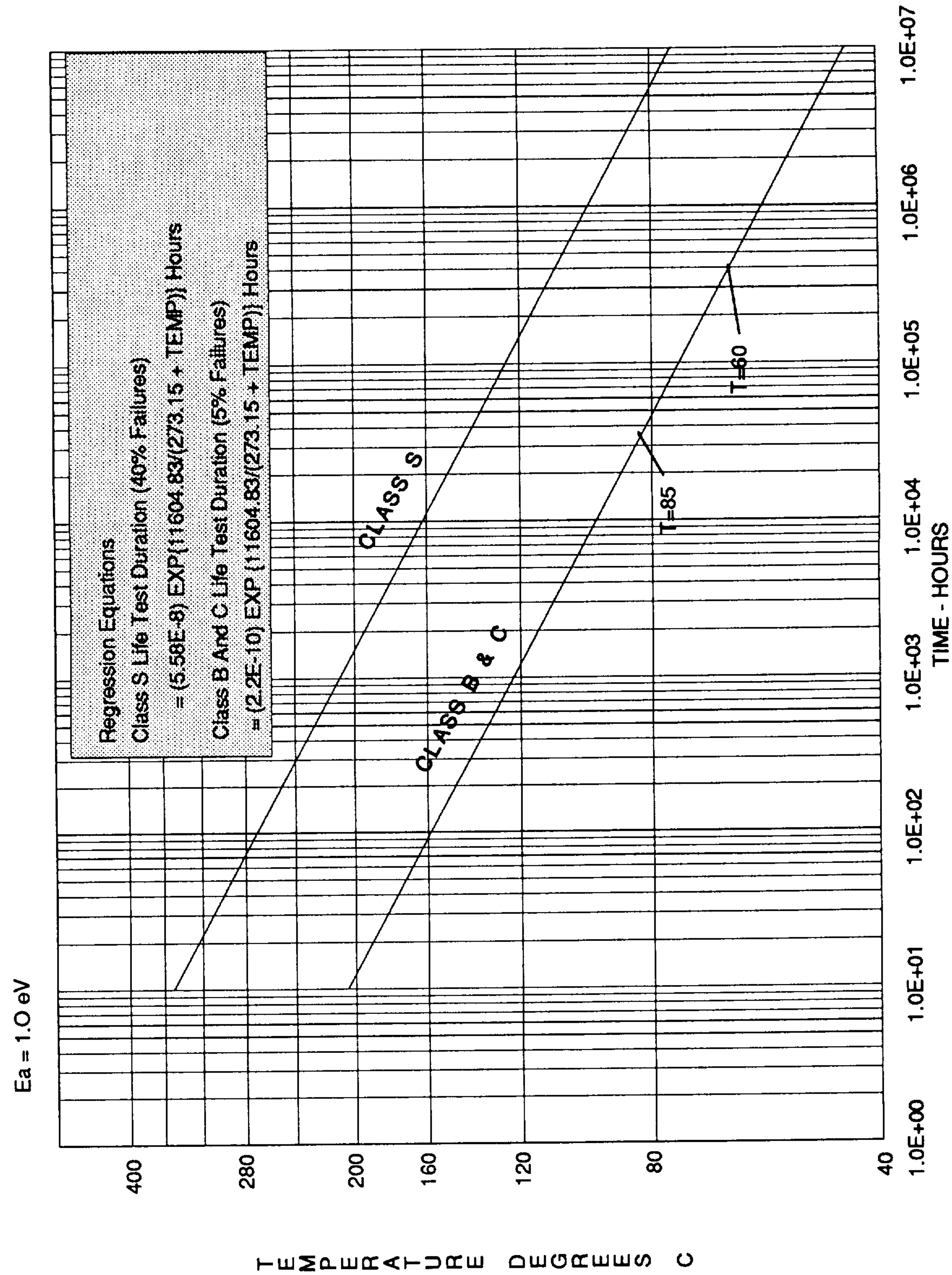
$$\frac{Reliability\ (Case\ A)}{Reliability\ (Case\ B)} \geq \frac{10}{1}$$

NOTE: In the example given, short-term ground test exposure on the order of 1-2 weeks will use an insignificant amount of life in hardware designed for long-life/high reliability, e.g., a 1-week thermal vacuum test at 75^0C provides a short-term high temperature screen in the actual circuit usage configuration to provide confidence for a long-term exposure under flight conditions ($J_T < 60^0C$), and uses only 0.018% of the parts capability. This demonstration is an important element in establishing prelaunch confidence in design adequacy.

Impact of Non-Practice:

Reliability of electronic parts will be reduced significantly.

PART JUNCTION TEMPERATURE



NASA
PREFERRED RELIABILITY PRACTICES

SURFACE CHARGING / ESD ANALYSIS

Practice:

Considering the natural environment, perform spacecraft charging analyses to determine that the energy that can be stored by each nonconductive surface is less than 3 mJ. Determine the feasibility of occurrence of electrostatic discharges (ESD). ESD should not be allowed to occur on surfaces near receivers/antenna operating at less than 8 GHz or on surfaces near sensitive circuits. For this practice to be effective, a test program to demonstrate the spacecraft's immunity to a 3 mJ ESD is required.

Benefit:

Surfaces that are conceivable ESD sources can be identified early in the program. Design changes such as application of a conductive coating and use of alternate materials can be implemented to eliminate or reduce the ESD risk. Preventive measures such as the installation of RC filters on sensitive circuits also can be implemented to control the adverse ESD effects.

Programs That Certified Usage:

Voyager, Galileo

Center to Contact for Information:

Jet Propulsion Laboratory (JPL)

Implementation Method:

Use a validated computer code (NASCAP or other appropriate computer code) to determine the maximum differential charging (V) of each nonconductive surface. When differential charging occurs, an electric field is developed within the dielectric material. The magnitude of the electric field (E) is given by:

$$E = V/d$$

where d is the thickness of the dielectric material. Usually, when this electric field is greater than 2×10^5 V/cm, ESD is likely to occur.

To determine the charging level, electrical properties of the nonconductive material must be known. These properties include

JET PROPULSION LABORATORY

SURFACE CHARGING / ESD ANALYSIS

(but are not limited to) surface resistivity, bulk resistivity, secondary and backscatter electron emission coefficient, and photoelectron yield. For materials with unknown electrical properties, the charging level must be determined by a ground test. In the ground test, the nonconductive surface is exposed to simulated charging environments (mission-dependent) and the resulting charging levels are measured.

ESD must not be allowed to occur on surfaces near sensitive RF receivers and on surfaces near sensitive circuits. For other surfaces, the energy of an ESD should be limited to 3 mJ. The ESD energy can be determined with the following equation:

$$W = 1/2CV^2$$

where C is the capacitance of the nonconductive surface with respect to spacecraft ground. The value C depends on the geometry (area and thickness) of the nonconductive surface. The ESD energy as a function of capacitance and charging level is displayed in Figure 1. Usually, the best way to reduce the ESD energy is to limit the value of V. This usually implies the use of a more conductive material. Since the charging current available in the space environment is relatively low, material with resistivity of 10^9 Ohm-cm is considered adequate for effective charge control.

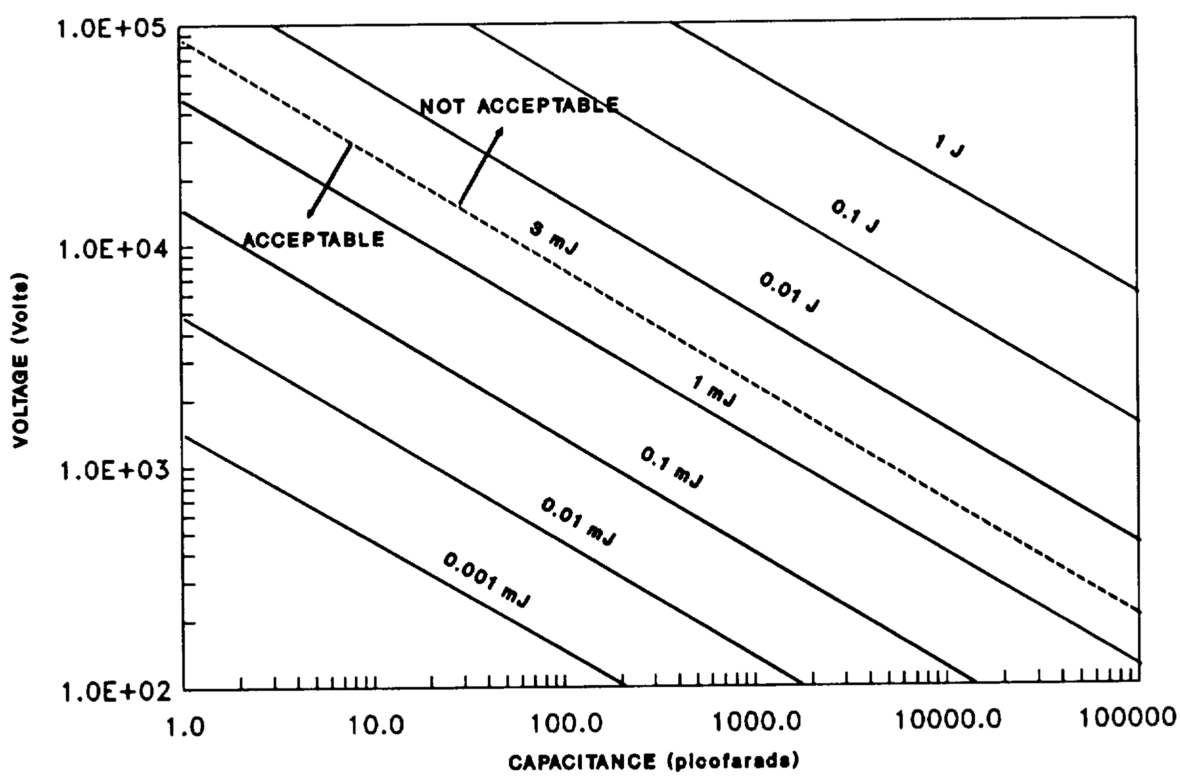


Figure 1. ESD Energy as a Function of Capacitance and Voltage

SURFACE CHARGING / ESD ANALYSIS

Technical Rationale:

In an environment of energetic electrons, spacecraft surface charging can occur. Due to their high resistivities, dielectric surfaces can be charged to different potentials than the metallic surfaces (which should be at spacecraft ground potential). When the electric field that results from differential charging is sufficiently high, an ESD would occur.

ESD is an intense source of electromagnetic interference (EMI). The EMI energies that can be capacitively and inductively coupled to electronic circuits are proportional to both the magnitude and rate of increase (dI/dt) of the discharge current, respectively. Under most conditions, the discharge current (I) is directly related to the energy (W) of a discharge. By minimizing the ESD energy, the magnitude of discharge current and the magnitude of ESD-induced EMI on circuits can be reduced.

The typical energy required to damage a sensitive IC is an order of several μJ. The energy required to upset a circuit is approximately 10 times less. In a typical discharge, only a fraction of the stored electrostatic energy can be coupled to a circuit. The coupling efficiency is dependent on the shielding and geometry of the spacecraft. Restricting the energy of an ESD minimizes the amount of energy available for IC damage and circuit upset, resulting in a more reliable spacecraft. In the Voyager ESD system test program, a 30 mJ discharge did not disturb spacecraft operation. However, differences in spacecraft configurations and circuit protection devices (e.g., RC filters in sensitive circuits) means that the "safe" (maximum allowable) energy could be different for different spacecraft configurations. Thus, 3 mJ was chosen as the maximum allowable energy.

Impact of Nonpractice:

Unpredictable operational anomalies and electronic parts failure caused by in-flight ESD events. Consequences could be catastrophic.

NASA
PREFERRED RELIABILITY PRACTICES

EEE PARTS SCREENING

Practice:

Implement a 100% nondestructive screening test on EEE parts prior to assembly, which would prevent early-life failures (generally referred to as infant mortality).

Benefits:

A lower rework cost during manufacturing and lower incident of component failures during flight.

Program That Certified Usage:

All Goddard Space Flight Center (GSFC) flight programs.

Centers to Contact for More Information:

- GSFC
- Jet Propulsion Laboratory (JPL) for missions referencing long-life, high reliability, or more stringent requirements.

Implementation Method:

Screening for each part can be established as follows:

- Refer to NASA's compilation of screening criteria for use with various EEE part types. An example may be found in Appendix C of the GSFC Preferred Parts List.
- If Class S parts are purchased, the screening tests shown in Table 1 have already been conducted. When Class S parts are not available, the screens of Table 1 should be used.
- Failure criteria during screening should specify Percent Defectives Allowable (PDA) and allowable parameter drift. Typical PDA criterion is 5%.

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A sample listing of failure mechanisms, the associated distribution of failures, and related screening tests are provided in Table 2.

EEE PARTS SCREENING

Table 1. 100% Parts Screening Matrix

Test and Inspection Requirements / Part Type	Semi Conductors			Capacitors									Resistors									Miscellaneous						
	Transistors, MIL-S-19500, JANS	Diodes, MIL-S-19500, JANS	Microcircuits, MIL-M-38510, Class S	Ceramic, G.P., MIL-C-39014	Ceramic, T.C., MIL-C-20	Ceramic, Chip, MIL-C-55681	Film, MIL-C-83421	Film, MIL-C-19978	Glass, MIL-C-23269	Mica, MIL-C-39001	Tant., Solid, MIL-C-39003	Tant., Foil/Slug, MIL-C-39006	Carbon Comp., MIL-R-39008	Film, G.P., MIL-R-39017	Film, Hi-Stab., MIL-R-55182	Film, Chip, MIL-R-55342	WW, Accurate, MIL-R-39005	WW, Power, MIL-R-39007	WW, Chassis, MIL-R-39009	Variable, Non WW, MIL-R-39035	Variable, WW, MIL-R-39015	Inductors	Transformers	Relays	Crystals	Fuses	Thermistors	Sensistors
1. Supplier Precap Visual	●	●	●																					●	●			
2. SEM (Sample)			●																					●	●			
3. Stabilization Bake	●	●	●																								●	
4. Temperature Cycle	●	●	●	●	●		●	●			●				●	●						●	●		●	●	●	●
5. Constant Acceleration	●	●	●																									
6. PIND	●	●	●																					●				
7. Seal	●	●	●				●	●	●		●	●												●	●			
8. HTRB	●	●	●																									
9. Power Burn-In	●	●	●																									●
10. Radiographic	●	●	●	●				●			●														●			
11. Parameter Measurements	●	●	●	●	●	●	●	●	●	●	●	●		●	●	●	●	●	●	●	●	●	●	●	●	●	●	●
12. Vibration																								●	●			
13. Power Conditioning														●			●	●	●	●	●							
14. Voltage Conditioning				●	●	●	●	●	●	●	●	●																
15. Overload															●													
16. Aging																									●			
17. Room Temp. Miss Test																								●				
18. High Temp. Miss Test																								●				
19. Low Temp. Test																								●				
20. Resistance-Temp. Characteristics																											●	
21. Contact Resistance Variation																				●								
22. Immersion																				●	●							

EEE PARTS SCREENING

TABLE 2. FAILURE MECHANISMS/SCREENING METHODS

Failure Mechanism	Distribution of Failures	Screen Tests
Metallization	11%	Burn-in, Internal Visual, Temperature Cycling, Scanning Electron Microscope for Metallization
Diffusion	1%	Burn-in
Oxide Faults	14%	Burn-in
Bulk	3%	Stabilization Bake, Burn-in, Temperature Cycling
Surface	21%	Internal Visual, Radiography, PIND, Constant Acceleration, Stabilization Bake, Burn-in
Interconnect	9%	Temp. Cycling, Burn-in, Constant Acceleration
Wirebond	1%	Nondestructive Bond Pull, Stabilization Bake, Temperature Cycling, Constant Acceleration, Internal Visual, Burn-in, Radiography, PIND
Package	40%	PIND, Radiography, Seal, External Visual, Temperature Cycling, Constant Acceleration

<u>Technical Rationale</u>:

The EEE parts manufacturing is controlled by military specification requirements covering a variety of areas such as: starting materials, process controls, electrical or electromechanical performance characteristics, and periodic inspections of some

EEE PARTS SCREENING

characteristics of finished product. Despite these requirements, defects that cause early-life failures can be randomly built into a product. The screening tests are designed to be destructive to parts with particular defects but nondestructive to good parts.

As an example, integrated circuits such as CMOS are highly susceptible to electrical performance failures caused by ionic contamination on the die surface. The contamination can be introduced by any of several uncontrollable avenues during manufacture and cannot be ruled out as an occurrence in any given lot of parts. To avoid early-life failures at higher assembly levels, the lot of parts is subjected to a 100% static burn-in. The burn-in is designed to drive contamination into the die areas where it will interfere with proper circuit operation and cause electrical failures before parts are installed on boards.

Impact of Nonpractice:

Without screening, there could be latent failure mechanisms that could cause flight delays and/or failures. For example, two circuits on the Solar Maximum Mission (SMM) spacecraft failed. The failed parts were analyzed upon return from the repair mission and found to contain defects that would have been revealed through screening. In one case, the microcircuit had a metallization flaw; in the second, the CMOS microcircuit had contamination on the die. In another example, screening tests performed on microcircuits resulted in an 85% failure rate. Subsequent failure analysis revealed that improper parts had been used.

References:

1. NASA GSFC Preferred Parts List (NPPL) 18/19.

2. Seidl, Raymond H., Garry, William J., "Pi Factors Revisited," Proceedings of the Annual Reliability and Maintainability Symposium, 1990.

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PRACTICES

THERMAL CYCLING

Practice:

As a minimum, run eight thermal cycles over the approximate temperature range for hardware that cycles in flight over ranges greater than 20°C. The last three thermal cycles should be failure-free.

Benefit:

Demonstrates readiness of the hardware to operate in the intended cyclic environment. Precipitates defects from design or manufacturing processes that could result in flight failures.

Programs That Certified Usage:

ATLAS, CENTAUR, Space Electronic Rocket Tests (SERTs) 1 and 2, Communication Technology Satellite (CTS), GOES, COBE, NOAA, LANDSAT, Solar Maximum Mission

Centers to Contact for Information:

- Lewis Research Center (LeRC)
- Goddard Space Flight Center (GSFC)

Implementation Method:

As part of ATP, run at least eight thermal cycles over the temperature range experienced by the hardware during storage, shipping, launch, flight, and reentry. The maximum and minimum temperatures anticipated should be exceeded by 10°C. The last three thermal cycles should be failure-free.

Equipment must stabilize at these limits before cycling to the opposing limit. Equipment generally should be operated within the anticipated thermal range rather than at the thermal limits.

Thermal cycling should be conducted in a vacuum if the test item is designed to operate in a vacuum.

THERMAL CYCLING

Technical Rationale:

Thermal cycle modeling has shown that the general form of the thermal cycling test math model is given by Equation (1).

$$TE = F \times P_d [1 - \exp(-\lambda_0 N K^{\Delta T})] \tag{1}$$

Where:
- TE = Test Effectiveness
- F = Fraction of total failures that can be precipitated by a thermal cycle
- P_d = Probability of detection
- λ_0 = Failure rate at T_0
- N = Number of thermal cycles
- K = A constant
- ΔT = $T - T_0$
- T = Operating temperature for λ
- T_0 = Operating temperature for λ_0

Fig. 1 shows that the failures available are the sum of three parts:

1. Failures detected by thermal cycle tests
2. Undetected failures
3. Failures not precipitated

For single temperature range of 50°C, the test effectiveness equation reduces to Equation (2).

$$TE = 0.9 \times P_d (1 - e^{-0.0864N}) \tag{2}$$

Figure 2 shows a plot of Equation (2) based on a probability of detection, P_d, of 0.9. The equation is based on values of λ_0 and K that were found by solving two simultaneous equations derived from the data base provided in Table 1.

Printed circuit boards (PCBs) are especially prone to solder joint cracking. The design is required to minimize the mechanical forces, as generated by thermal mismatch of materials or vibration, in the solder joints.

THERMAL CYCLING

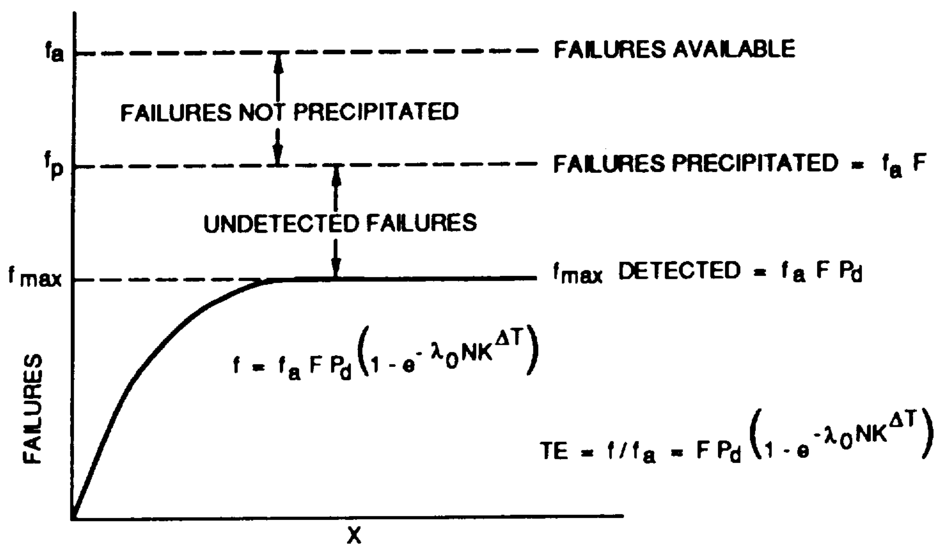

Fig. 1 General Form of TC Test Model

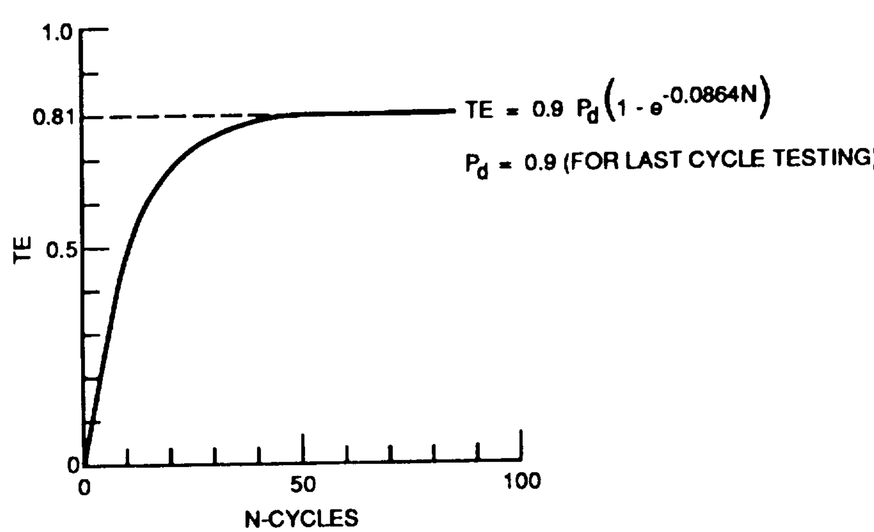

Fig. 2 Test Effectiveness Plot for $\Delta T = 50^0C$

THERMAL CYCLING

Impact of Nonpractice:

Design and manufacturing defects that could have been detected during ground testing manifest themselves during flight.

Related Practices:

Solder Joint Fatigue Cycles, Thermal Dwell Testing.

References

1. GDCD BNZ 69-007, Curssell, G. M., "Atlas and Centaur Component Acceptance Test Plan," 1984.

2. NASA TMX-53731, Van Orden, R. E., "Mounting of Components to Printed Wiring Boards," 1968.

3. Laube, R. B., "Space Vehicle Thermal Cycling Test Parameters," Proceeding of the Institute of Environmental Sciences, 1983.

4. Nelson, C. E., "System Level Reliability Thermal Cycling," Proceeding of the Institute of Environment Sciences, 1983.

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PREFERRED RELIABILITY PRACTICES

THERMOGRAPHIC MAPPING OF PC BOARDS

Practice:

Use thermographic mapping methods to locate hot spots on operating PC boards.

Benefit:

Quick find of electronic components operating at or above recommended temperatures. Also, this technique can validate the derating factors and thermal design via low cost testing versus analysis.

Programs That Certified Usage:

Space Acceleration Measurement System (SAMS), Isothermal Dendritic Growth experiment (IDGE), and STDCE

Center to Contact for More Information:

Lewis Research Center (LeRC)

Implementation Method:

Using an infrared camera and the flight PC board, make thermographic pictures of the prototype PC boards in operation. Verify the thermograph and determine the delta T to the actual use environment with thermocouples. Shut down the equipment and prepare it for a vacuum test.

The board to be tested is placed in a mother board with the appropriate +5 V and ±12 V power supplies. Power is applied to the board, and after a short period, a video recording of the board is made with an infrared camera.

Technical Rationale:

The following procedure is used to determine the temperature of each component:

Junction temperature: $$T_J = T_A + T_{JA} \quad (1)$$

Where:
T_J = Junction Temperature
T_A = Ambient Temperature
T_{JA} = Junction to Ambient Temperature Rise

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THERMOGRAPHIC MAPPING OF PC BOARDS

T_J = 40°C for this example

The case temperature, T_C, which is measured on the bench at room ambient is given by Equation (2):

$$T_C = T_J - \theta_{CA}P \tag{2}$$

Where: T_C = case temperature
θ_{CA} = case to ambient thermal resistance
P = Power dissipated

For reliability purposes, it is necessary to keep junction temperatures for CMOS devices at or below 49°C. The case temperature to be measured on the bench comes out to be T_C = 34°C for this application.

Infrared pictures are made of the PCB mounted outside the package on extended connectors while the equipment is operating on the bench. The logic IC temperature is determined from the infrared picture. If less than or equal to 34°C, the junctions are at the desired operating temperature. If greater than 34°C, the reason for the higher temperature is determined. Corrective action is worked out and approved by the Engineering Review Board.

Figure 1 is a drawing of the component layout of the SCSI card, and Figure 2 is a thermographic photograph of the board. Thermographic pictures are usually in color, but in this monochrome reproduction, the cross hairs are at the hottest location (128°F), black represents 108°F, white is 98°F, dark grey is 88°F, and light grey is 78°F.

The operating temperature with the board back in the case is checked by several thermocouples attached to the hottest observed components. This is done in a simulated use environment, perhaps during the thermal environment tests. The resultant delta-T is added to the measured case temperature as a final check of the junction temperature, T_J, in the end-use environment. For sample logic IC, the delta T was 5°C so the resultant junction temperature is 45°C in the package. This is below the guideline of 49°C.

THERMOGRAPHIC MAPPING OF PC BOARDS

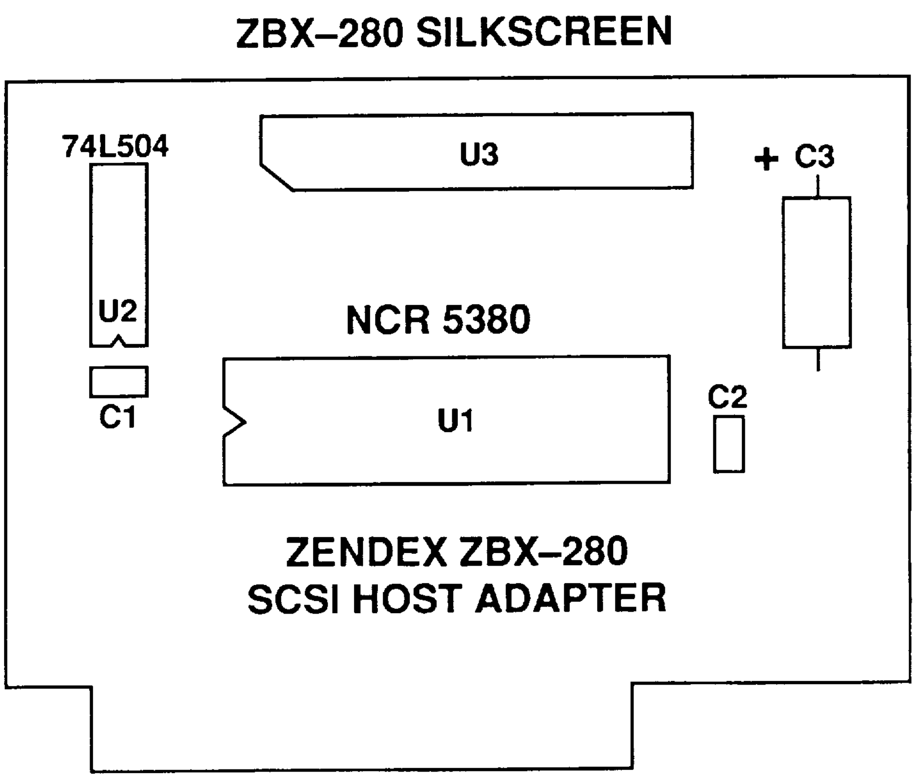

Figure 1 SCSI Card

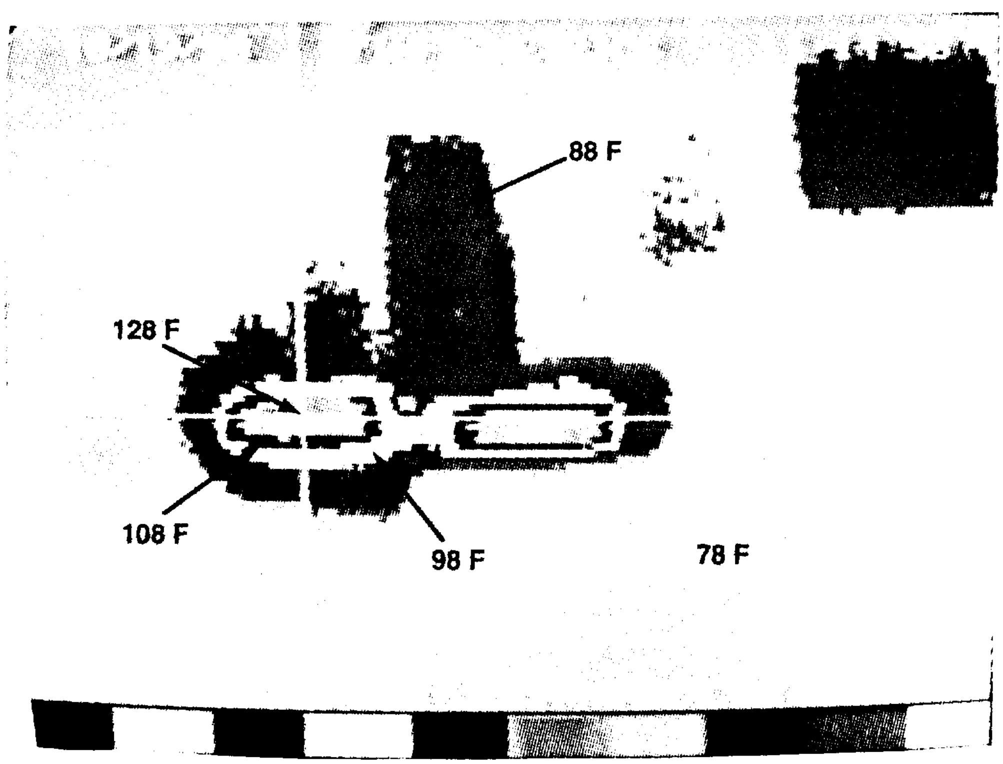

Figure 2 Thermograph

THERMOGRAPHIC MAPPING OF PC BOARDS

Impact of Nonpractice:

Allowing undetected hot spots to exist in flight hardware can be very expensive since the later a problem is detected in a flight program, the more it costs to repair. Using thermography to verify system engineering models is a fast, low-cost technique.

References:

1. Crall, R. F., "Thermal Imaging Benchtop Analysis for Reliability," Evaluation Engineering, December 1989.

2. Masi, C. G., "What Can Thermal Imaging Do for You?, " Test & Measurement World, May 1988.

3. MIL-HDBK-217E, "Reliability Prediction of Electronic Equipment," Rome Air Defense Center, October 27, 1986.

4. Foster, W. M., "Thermal Test Report for the Space Acceleration Measurement System Circuit Boards", NASA LeRC Code 6730 Internal Report, November 1987.

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PRACTICES

THERMAL TEST LEVELS / DURATIONS

Practice:

Perform thermal dwell test[1] on protoflight hardware over the temperature range of +75°C/-20°C (applied at the thermal control/mounting surface or shearplate) for 24 hours at the cold end and 144 to 288 hours at the hot end.

Benefit:

This test, coupled with rigorous design practices, provides high confidence that the hardware design is not marginal in its intended long life high reliability mission.

Programs Which Certified Usage:

Voyager, Galileo, Viking and Mariner Series

Center to Contact for Information:

JPL

Implementation Method:

Establish a minimum hardware test temperature level range of -20°C/+75°C and specify that a single cycle thermal dwell test be performed for the appropriate durations (24 hours cold and 144 to 288 hours hot).

Technical Rationale:

In the early 1960's, JPL adopted a conservative set of thermal design and test temperature levels to demonstrate hardware design adequacy. As a starting point, a reasonable short term flight temperature excursion (+5°C to +50°C) was established for thermal control surfaces (shearplates). The +5°C lower level is a few degrees Celsius above the freezing point of Hydrazine, thus integrated thermal control of bus electronics and propulsion systems is possible. The 50°C upper limit is the approximate level reached by a louvered bus electronics bay after about one hour of full (perpendicular) solar irradiance at one A. U. (astronomical unit) and accomodates near earth maneuvers. The long term desired thermal control range is typically 25±5°C, but this

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[1]Thermal dwell testing is the standard practice at JPL for systems/components which do not thermally cycle during flight. For systems/components that do thermally cycle (generally over a range > 20 °C) in flight, JPL practice is to cycle over a conservative range for three times the number of flight cycles.

THERMAL TEST LEVELS / DURATIONS

range may be broader depending on the tradeoffs of long term reliability and thermal control costs. This original approach reduced the overall complexity of the system thermal control design process: the wide range reduced the sensitivity to louver/radiator size, heater size, power variations, etc. A margin of ±25^{0}C was then applied to the Allowable Flight range for qualification/protoflight test levels of assemblies mounted to such thermal control surfaces. These levels accommodate thermal compromises in the design where the short term extremes may be approached during steady state operation and also has been demonstrated to provide an effective screen of assemblies. This resulted in the JPL standard minimum test range of -20^{0}C to +75^{0}C (for electronic assemblies in particular).

These conservative test level ranges lead to several desirable features. The conservative high temperature limit restricts the permitted temperature rise from the shearplate to the junction of electronic piece-parts. Thus junction temperatures during the bulk of a mission are much cooler than assemblies designed and tested at lower shearplate temperatures. The increase in theoretical reliability is on the order of a factor of 10 per 25^{0}C.[2]

There are at least two failure mechanisms for both design and workmanship that should be screened by an adequate thermal environmental test of any given assembly. The first is based on Arrhenius rate related physics where time at high temperature is the key to demonstrating reliability during testing. Electronic part life is a prime example of an Arrhenius mechanism, but so are other elements of assemblies including interactions between metal traces within Printed Wiring Boards (PWB's), certain component to board joints and even solder joints to a certain extent. The other identifiable mechanism is thermally induced mechanical stress (including fatigue) as between components and the board and especially solder joints.

Arrhenius Rate Physics:

Contrast the test level of 75^{0}C (shearplate) to 50^{0}C short term worst case transients during flight and 25^{0}C for the bulk of the mission. Based on Arrhenius reaction rate physics described and shown on page 6, the 75^{0}C test provides a demonstrated reliability some 2 to 8 times that of short transients to 50^{0}C, (typical of thermal cycling tests), and some 4 to 94 times that of long term mission shearplate temperatures (25^{0}C). These reliability ratios are based on activation energies of 0.3 eV to 1.0 eV which cover most assembly element reaction physics.

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[2]See "Part Junction Temperature", Practice No. PD-ED-1204

THERMAL TEST LEVELS / DURATIONS

The Mariner and Viking spacecraft performed a hot dwell test (75^{0}C) of 288 hours duration. This was reduced to 144 hours for the Voyager and Galileo spacecraft. The statistical data base supporting this shorter test is unique to the JPL design rules and processes, therefore the longer hot dwell duration of 288 hours is recommended for assemblies designed to non-equivalent or less conservative practices.

On page 7 we show the percentage of the screening test capability for Class S parts that is used by a JPL assembly test at 75^{0}C for 144 hours. A very conservative assumption here is that all parts in the assembly test have a 35^{0}C temperature rise and that they are at 110^{0}C for the entire test. Even given this over-conservative assumption, the JPL test uses only 0.018% of the class S parts minimum screened capability. Clearly less than 2/10000's of the minimum parts capability being dedicated to the assembly protoflight test is not a concern. The parts are not over-stressed by this test.

Thermally Induced Mechanical Stress (Fatigue):

JPL has historically done a thermal dwell test rather than a specific thermal cycle test. There are data that indicate thermal cycling uses up hardware life and therefore is degrading to the flight hardware. In practice, the JPL test approach is never really just a one-cycle dwell test. The assembly test program (plus any retest) and the systems test program (frequently two phases) result in a minimum of two cycles and as many as four (or more) are possible although they are not continuous and the transients are controlled to < 30^{0}C/hr to prevent thermal shock. The VGR hardware was tested as follows:

	Proof Test Assemblies Qualification Test	Flight Assemblies Acceptance Test
Cycles	1 Assembly (+ Retest)	1 Assembly (+ Retest)
	2 Systems	1 Systems
	3 Cycles (+ Retest)	2 Cycles (+ Retest)

In a recent JPL study, a fatigue life relationship of equivalent thermal cycles was determined over different temperature ranges as follows:

THERMAL TEST LEVELS / DURATIONS

$$C_2 = C_1 \left(\frac{T_1}{T_2}\right)^Y$$

where: C_1 is the number of thermal cycles over a T_1 range
C_2 is the number of thermal cycles over a T_2 range
and $Y = 2.6$ for eutectic solder.

As a frame of comparison for workmanship purposes, the JPL protoflight test of 1 cycle over a -20/75^{0}C range can be correlated to an acceptance test of 6 cycles over a 0/50^{0}C range. In this case:

$C_1 = 1$, $T_1 = 95^0$C,
C_2 = TBD, $T_2 = 50^0$C

and the equivalent cycles of the JPL test are:

$C_2 = 1(95^0\text{C}/50^0\text{C})^{2.6} = 5.3$ cycles.

Therefore, in terms of solder joint fatigue life, the JPL protoflight test equivalency to 5.3 cycles over a 50^{0}C range says that, for workmanship acceptance purposes, the JPL protoflight test is essentially the same as the example thermal cycle acceptance test, i.e., $5.3 \approx 6$ cycles.

On page 8, a comparison of solder joint fatigue life comparisons has been made. The recommended -20/+75^{0}C single cycle dwell test uses only 0.14% of the fatigue life of a solder joint qualified to NHB 5300.4 (3A-1). The point of this comparison is that the JPL protoflight test is less strenuous to solder joints than thermal cycle testing performed by most organizations.

Ground Test & Thermally Related Problem/Failure Statistics:

These practices were applied to the Mariner spacecraft series, the two Viking 75 spacecrafts, the two Voyager 77 spacecrafts and more recently the Galileo spacecraft. These spacecraft all completed (or exceeded) their intended mission successfully (the Galileo mission is still underway). In fact, the Voyager spacecraft have worked for over 13 years.

THERMAL TEST LEVELS / DURATIONS

The total number of assembly problems/failures during these missions is small, and the number of thermally induced problems even smaller. This is shown in the following table where the number of problem/failures identified during assembly level thermal testing are compared with suspected flight problems/failures for the Viking, Voyager, and Galileo programs:

	Number of Problem/ Failures Identified during Assembly Thermal Testing	Number of Known Thermally Induced Flight Problem Failures
VIKING (2 SPACECRAFT)	251	None Obvious
VOYAGER (2 SPACECRAFT)	123	1
GALILEO (1 SPACECRAFT)	50	None to Date

Impact of Non-practice:

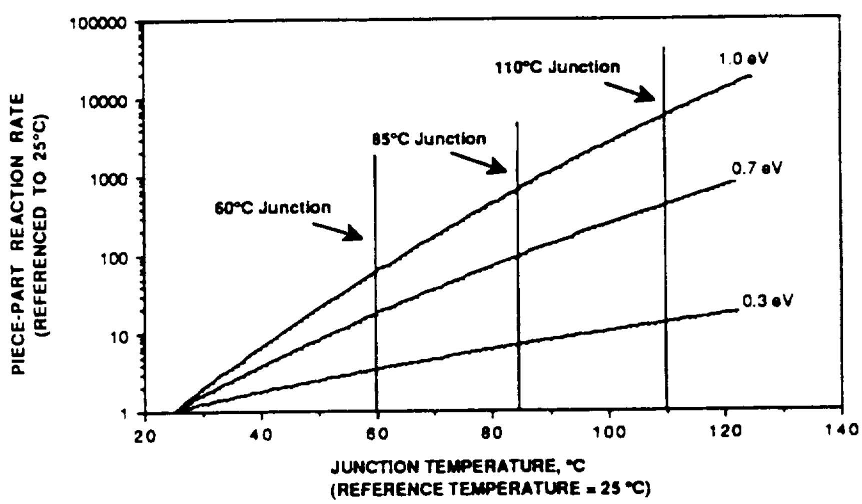
Demonstrated design adequacy and its implications to long term reliability are affected. For example, testing at 50°C instead of 75°C and for about 20 hours instead of 144 hours reduces test demonstrated reliability by a factor on the order of 50.

THERMAL TEST LEVELS / DURATIONS

ARRHENIUS REACTION RATE THEORY

$$\frac{\lambda}{\lambda_o} = e^{-\frac{E_a}{k}\left(\frac{1}{T} - \frac{1}{T_o}\right)}$$

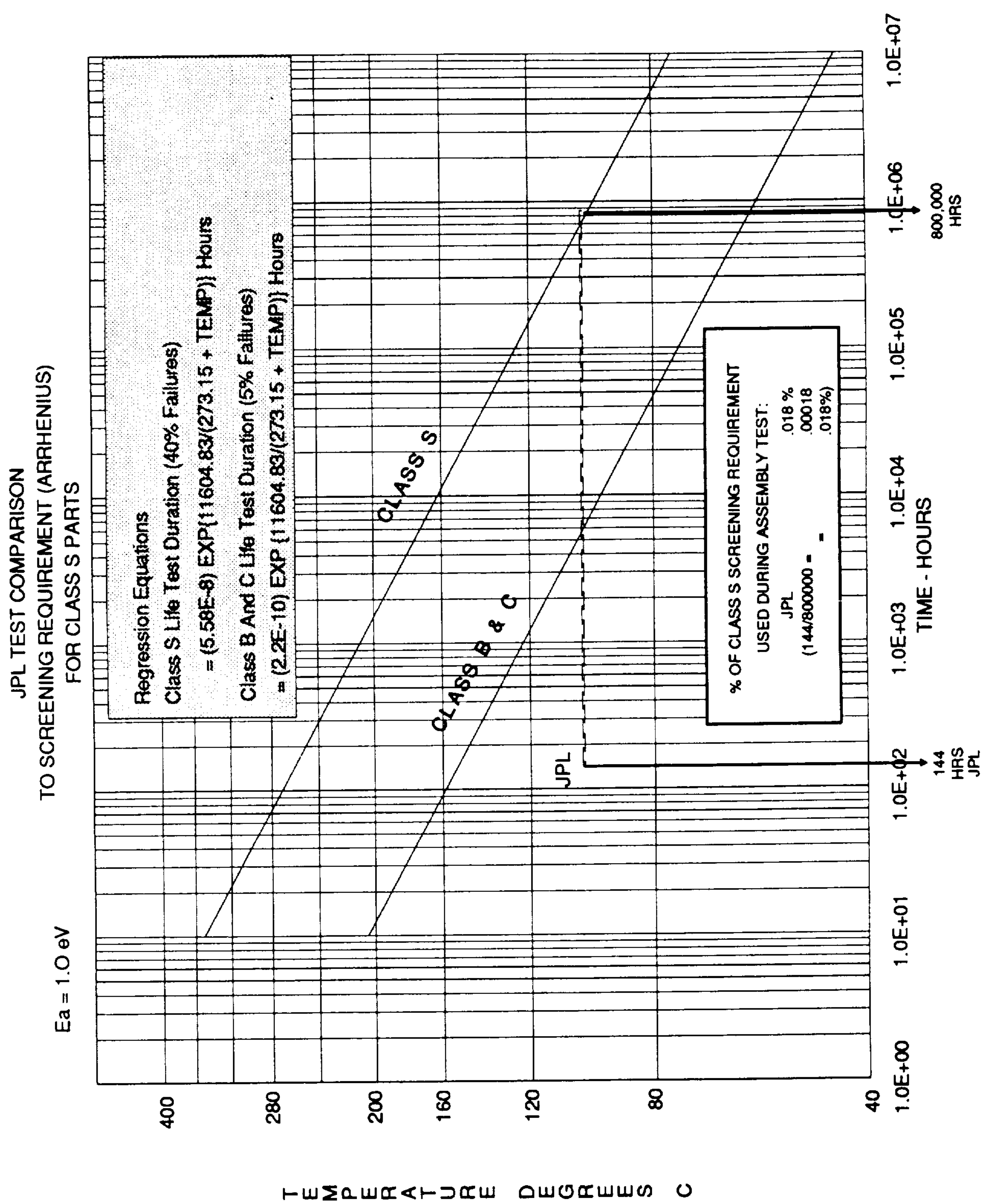
where:
λ - Reaction rate at temperature T (a measure of failures/time)
λ_o - Reaction rate at reference temperature T_o
E_a - Activation energy, eV
T - Temperature in degrees Kelvin (°K)
T_o - Reference temperature in degrees Kelvin (°K)
k - Boltzmann's constant (8.617×10^{-5} eV/°K)



ACTIVATION ENERGY (eV) →		0.3	0.7	1.0
TEST CONDITION 75 °C SHEARPLATE 110 °C JUNCTION	$\frac{\lambda_{JUNCT.}}{\lambda_{25°C}}$	13.3	420.2	5595.1
SHORT TERM FLIGHT TRANSIENT 50 °C SHEARPLATE 85 °C JUNCTION	$\frac{\lambda_{JUNCT.}}{\lambda_{25°C}}$	7.1	95.8	676.5
LONG TERM FLIGHT CONDITION 25 °C SHEARPLATE 60 °C JUNCTION	$\frac{\lambda_{JUNCT.}}{\lambda_{25°C}}$	3.4	17.5	59.6
TEST CONDITION OVER SHORT TERM FLIGHT TRANSIENT	$\frac{\lambda_{110°C}}{\lambda_{85°C}}$	1.9	4.4	8.3
TEST CONDITION OVER LONG TERM FLIGHT CONDITION	$\frac{\lambda_{110°C}}{\lambda_{60°C}}$	3.9	24.0	93.9

RATIO OF ARRHENIUS FAILURE RATES FOR VARIOUS ACTIVATION ENERGIES AND PRACTICE CONDITIONS

THERMAL TEST LEVELS / DURATIONS



THERMAL TEST LEVELS / DURATIONS

SOLDER JOINT FATIGUE LIFE COMPARISON

NHB 5300.4 (3A-1) PACKAGING QUALIFICATION TEST	JPL PROTOFLIGHT	
-55°C to 100°C 200 cycles	-20°C to 75°C 1 cycle 24 hrs cold 144 hrs hot	
QUALIFICATION BASELINE	EXPOSURE	LIFE EFFECT*
	1 cycle of 95°C	0.14% of NHB 5300.4(3A-1) solder joint

$* \left(\frac{95}{155}\right)^{2.6}\left(\frac{1}{200}\right) = .0014 = .14\%$

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PRACTICES

POWERED-ON VIBRATION

Practice:

Supply power to electronic assemblies during vibration, acoustics and pyroshock and monitor the electrical functions continuously while the excitation is applied.

Benefit:

Aids in the detection of intermittent or incipient failures in electronic circuitry not otherwise found. Benefit even for those electronics not powered during launch.

Programs Which Certified Usage:

Mariner series, Viking, Voyager, Magellan, Galileo.

Center to Contact for Information:

JPL

Implementation Method:

Apply service power to electronics assemblies. Monitor as many circuits as possible for intermittent behavior or change in voltage/current level. Record for later analysis the most critical electrical functions. Employ instrumentation such as a storage logic analyzer to monitor relay contacts, especially during pyroshock testing.

Technical Rationale:

NASA and industry practice of powering electronic assemblies during dynamics testing has proven to be effective in uncovering otherwise undetected "soft" failures. Studies by the Institute of Environmental Sciences, U.S. military, Tustin Technological Institute, Hobbes Engineering and others have all arrived at the same general conclusion - power-on vibration is a valuable tool for exposing latent defects in electronic hardware with the eventual resultant improvement in product quality.

Intermittencies in electronic circuity can often be detected <u>during</u> vibration but may not be observed under ambient functional testing. These intermittencies may not reappear until after launch and in some cases degenerate into hard failures.

POWERED-ON VIBRATION

Examples of this include:

- Component shorts due to internal conductive particles
- Loose or contaminated connectors
- Fractured component-to-board solder joints
- Electrical arcing
- Data number changes in digital equipment
- Relay transfer or chatter

Powering of electronic equipment during vibration allows for detection of failures or intermittent conditions when they occur. This can be extremely useful in diagnosing the problem and formulating corrective action. In vibration, it is advantageous to know in what environment, level, axis, and time the anomaly occurred. Also, this procedure allows a test to be discontinued at the time the anomaly occurs to avoid potential further damage.

Impact of Non-Practice:

Increased probability of flight equipment containing flaws or intermittencies that cause mission compromises or failures, for example: electrical arcing, open circuits, and relay chatter.

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SINUSOIDAL VIBRATION

Practice:

Subject assemblies and the full-up flight system to swept sinusoidal vibration.

Benefit:

Certain failures are not normally exposed by random vibration. Sinusoidal vibration permits greater displacement excitation of the test item in the lower frequencies.

Programs Which Certified Usage:

Mariner Series, Viking, Voyager, Galileo

Center to Contact for Information:

JPL

Implementation Method:

Apply sinusoidal vibration to the test item by sweeping over a frequency range beginning at ≈ 10 Hz (± one octave) up to ≈ 100 Hz (± one octave). Sweep the frequency range at a logarithmic rate (i.e. $\Delta f/f$ is constant). Sinusoidal vibration is performed with the same fixturing and concurrent with random vibration.

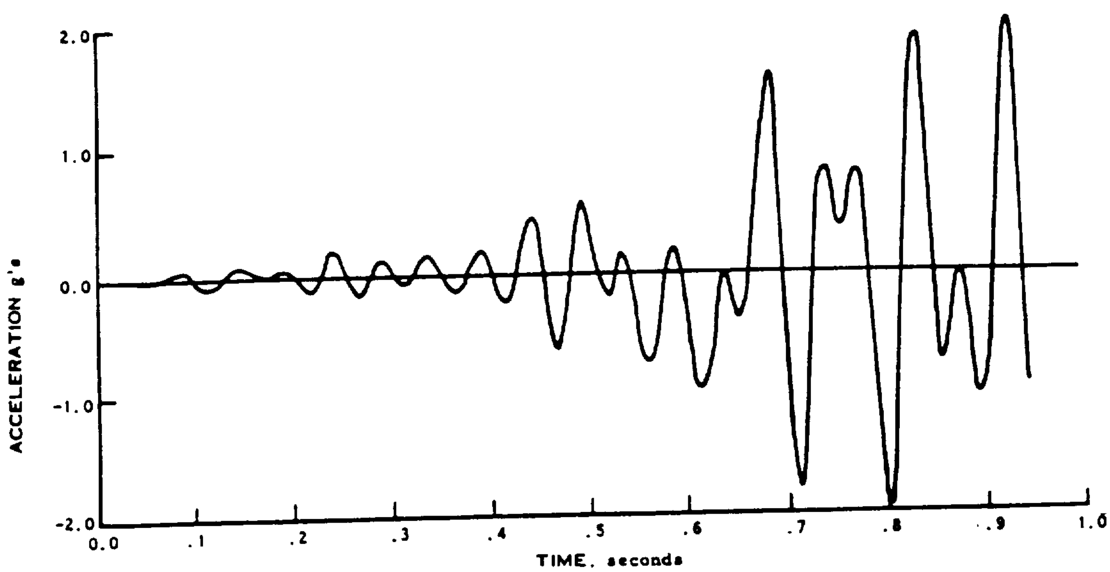
Technical Rationale:

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 currently widespread 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.

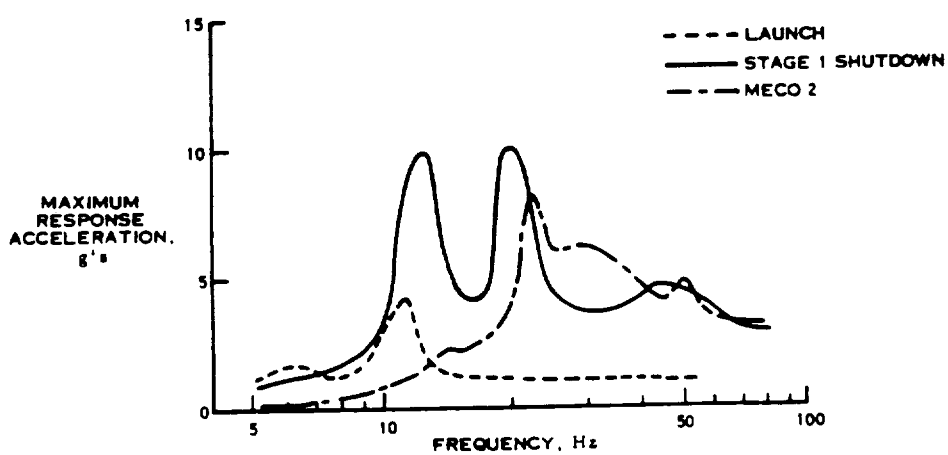
SINUSOIDAL VIBRATION

Sinusoidal vibration levels can be derived as in the following example:

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

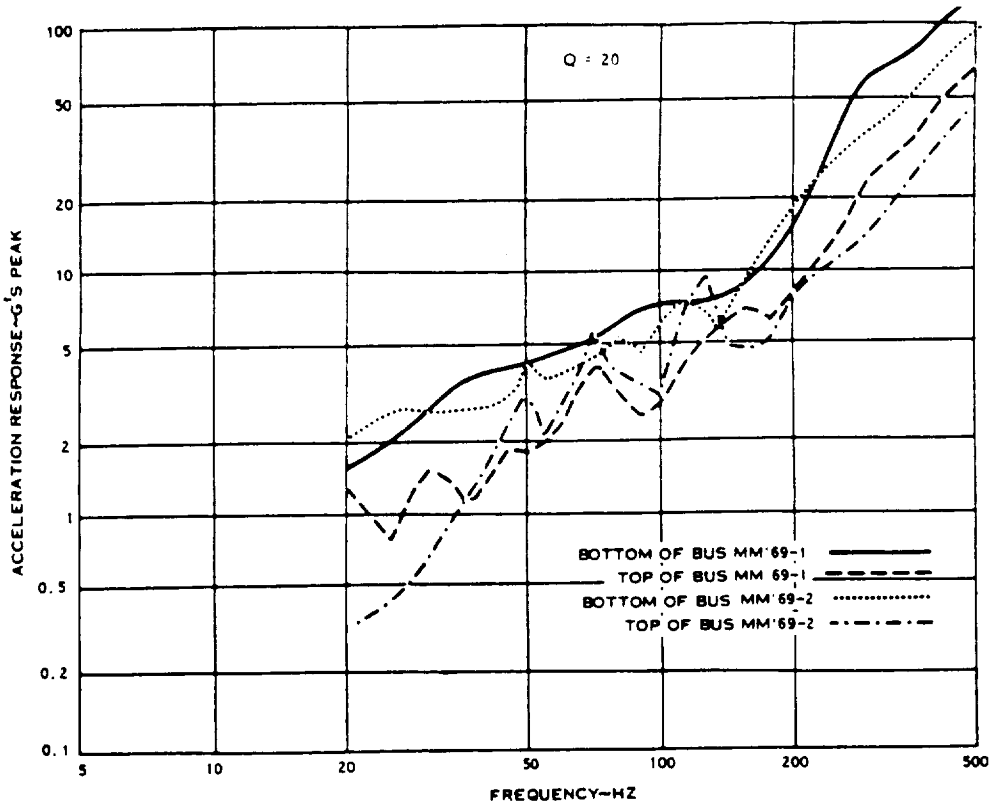


2. Compute the shock spectra for each of the waveforms in Step 1:

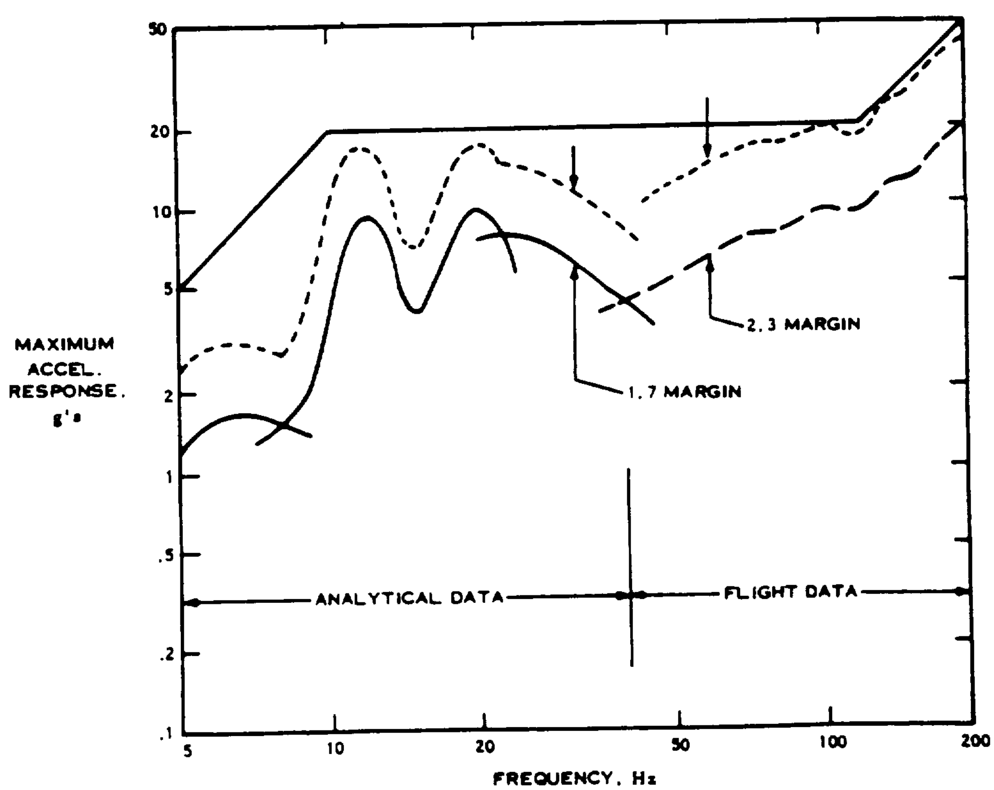


SINUSOIDAL VIBRATION

3. Take data from previous flight measurements:

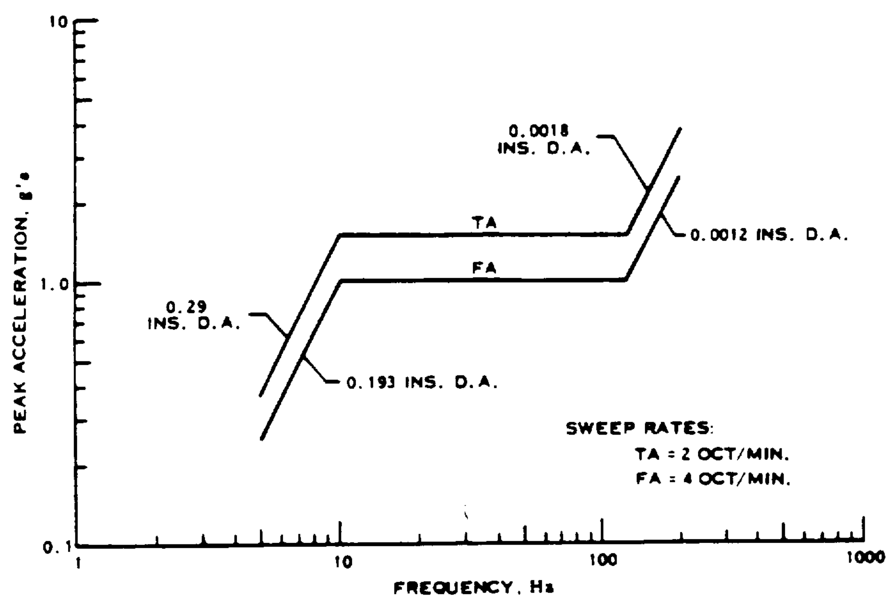


4. Combine results from steps 2, and 3 and envelope:



SINUSOIDAL VIBRATION

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



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.

Impact of Non-Practice:

Probability of failure is increased in flight due to low frequency transient environment. Some workmanship defects in large structures and full-up systems may go undetected.

III. RELIABILITY DESIGN GUIDELINES

A. INTRODUCTION

This section contains Reliability Design Guidelines for consideration by the aerospace community. The guidelines presented in this section contain information that, in the opinion of the sponsoring activity, represents a technically credible process that are applied to ongoing NASA programs/projects. Unlike a Reliability Design Practice, a guideline lacks specific operational experience or data to indicate that a topic area has contributed to mission success. However, a guideline does contains information that represents current "best thinking" on a particular topic.

B. RELIABILITY GUIDELINE FORMAT DEFINITIONS

The format for the reliability guidelines is shown in Figure 2.

GUIDELINE FORMAT DEFINITIONS

Guideline: *A brief statement of the guideline.*

Benefit: *A concise statement of the technical improvement realized from implementing the guideline.*

Center to Contact for More Information: *Source of additional information, usually the sponsoring NASA Center. See "CENTER CONTACTS", page ii.*

Implementation Method: *A brief technical discussion that is not intended to give the full details of the process, but rather to provide a design engineer with adequate information to understand how the guideline should be used.*

Technical Rationale: *A brief technical justification for use of the guideline.*

Impact of Nonpractice: *A brief statement of what can be expected if use of the guideline is avoided.*

Related Guidelines: *Identification of other topic areas in the manual that contain related information.*

References: *Publications that contain additional information about the guideline.*

SPONSOR
OF
GUIDELINE

Figure 2

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GUIDELINES AS OF APRIL, 1991

GD-AP-2301 Earth Orbit Environmental Heating

NASA
PREFERRED
RELIABILITY
PRACTICES

EARTH ORBIT ENVIRONMENTAL HEATING

Guideline:

Use the currently accepted values for the solar constant, albedo and earth radiation when calculating the heat balance of earth orbiters. This practice provides the heating rates for the black body case without consideration of spectral effects or collimation.

Benefit:

Consideration of the solar, albedo, and earth radiation thermal inputs, including seasonal variation with tolerances, is required to accurately predict the thermal environment of orbiting devices.

Center to Contact for More Information:

Goddard Space Flight Center (GSFC)

Implementation Method:

SOLAR CONSTANT

The nominal solar constant value is 1367.5 W/m^2. The variation of the earth-sun distance causes a ± 3.5% seasonal variation from nominal. The accuracy of the solar constant is taken as ± 0.5%. The following are the values for various seasons in the northern hemisphere.

NOMINAL	1367.5 W/m^2
WINTER	1422.0 W/m^2 (NOM + 4.0%)
SUMMER	1318.0 W/m^2 (NOM - 4.0%)

*ALBEDO FACTOR**

The nominal albedo factor is 0.30. The variation around the nominal should be ± 0.05. No variation during the sunlit portion of a given orbit should be assumed unless extremely light weight items

GODDARD
SPACE FLIGHT
CENTER

*Note: Since earth temperature and albedo vary with latitude, as the orbit approaches either extreme of a polar or equatorial orbit, further study of the literature should be made. (see AIAA - 87-1596)

EARTH ORBIT ENVIRONMENTAL HEATING

are being considered. Programs that compute albedo energy should use 0.35 (hot case), 0.30 (nominal case), and 0.25 (cold case), respectively.

*EARTH EMITTED ENERGY**

The nominal earth temperature for earth emitted IR energy is 255°K. This temperature produces a heating rate of 241 W/m^2. A reasonable variation can be obtained by maintaining consistency using the following relationship between Solar, Albedo, and Earth Emitted Energy:

Earth Emitted Energy = [(1-Albedo Factor) x Solar Constant] / 4.0

Table 1 shows the variations in Earth Emitted Energy that result from using the above recommended Solar and Albedo ranges.

Software programs that compute Earth Emitted Energy should use the appropriate hot, nominal, or cold case Solar and Albedo values; and the corresponding black body Earth temperature to achieve an energy balance.

REFERENCES FOR QUICK CHECKS OR SIMPLE CALCULATIONS

Hand calculations should be made to verify that computer outputs of heating values for flat surfaces of known orientation and minimal reflected inputs from other surfaces are reasonable. Hand calculations also may be necessary when time does not permit a computer study. A check of incident Albedo energy to a flat plate at various altitudes and orientations can be made by using TN-D 1842 "Earth Reflected Solar Radiation Incident Upon an Arbitrary Oriented Spinning Flat Plate," by F. Cunningham. Figures 1 through 9 show the orbit-averaged incident Earth and Albedo energies to an Earth-oriented flat plate at various altitudes and orbit/sun angles. Eclipse factors for elliptical orbits are provided in "Calculation of the Eclipse Factor for Elliptical Satellite Orbits", by F. Cunningham. A hand calculation of incident Earth Emitted Energy to a flat plate at various attitudes and altitudes also is possible. Figure 10 shows the instantaneous geometric shape factor for a planar surface as a function of altitude and attitude (h/R is the ratio of the orbit altitude to the Earth radius). The earth radius is 6,365 km. The incident Earth Emitted Energy is found by multiplying the shape factor times the black

*Note: Since earth temperature and albedo vary with latitude, as the orbit approaches either extreme of a polar or equatorial orbit, further study of the literature should be made. (see AIAA - 87-1596)

EARTH ORBIT ENVIRONMENTAL HEATING

body emissive power at the earth temperature. For an altitude of 1,000 km and a flat plate whose normal is 90 degrees to the nadir (λ = 90); h/R = 0.157, which gives a shape factor of 0.19. The Earth Emitted Energy incident on the plate is 0.19 x 241 W/m^2 or 46 W/m^2.

TABLE 1. VARIATIONS IN EARTH EMITTED ENERGY FOR RECOMMENDED SOLAR AND ALBEDO RANGES*

SOLAR CONSTANT (W/m^2)	ALBEDO FACTOR	EARTH EMITTED ENERGY (W/m^2)	EQUIV. EARTH TEMP (°K)
NOMINAL 1368	0.25 0.30 0.35	256 239 222	258 254 250
WINTER SOLSTICE 1422	0.25 0.30 0.35	267 249 231	262 258 253
SUMMER SOLSTICE 1318	0.25 0.30 0.35	247 231 214	256 251 246

* For use in Orbit Average Analyses

NOTE: Since earth temperature and albedo vary with latitude, as the orbit approaches either extreme of a polar or equatorial orbit, further study of the literature should be made (see AIAA - 87-1596).

EARTH ORBIT ENVIRONMENTAL HEATING

EQUIVALENT SINK TECHNIQUE

The equivalent sink technique can be used by replacing all surrounding surface radiant interchanges and the absorbed Solar and Earth energies to node i with a single radiation coupling to a single node at temperature T sink.

To derive the equation for this sink temperature, first consider an energy balance at node i where all the inputs are treated as gross inputs and node i has a view to space of 1.0

(1) $$Q_{s+A} + Q_{IR} + Q_I + \sum_{n=1}^{k} \mathcal{F}A_{i-n}\sigma T_n^4 = \epsilon_i A_i \sigma T_i^4$$

From planetary flux program (TRASYS or SSPTA)

From thermal program (SINDA) results obtained from Geometric Math Model (GMM) radiation exchange program

Where:

Q_{s+A} = *absorbed solar and albedo energy*
Q_{IR} = *absorbed earth IR energy*
Q_I = *internal power dissipation*
σ = *Stefan-Boltzmann constant*
$\mathcal{F}A_{i-n}$ = *Radiant interchange factor*
A_i = *Area of node i*
ϵ_i = *emissivity of node i*

Next consider the equivalent sink energy balance situation: $Q_I \rightarrow i \circ\!\!-\!\!\!\!\underset{}{\overset{\epsilon_i A_i}{\text{/\\/\\/\\/\\/}}}\!\!-\!\!\circ\ T_s$

(2) $$Q_I = \epsilon_i A_i(\sigma T_i^4 - \sigma T_s^4) = \epsilon_i A_i \sigma T_i^4 - \epsilon_i A_i \sigma T_s^4$$

Solving (1) for Q_I and setting equal to the right side of (2) gives:

(3)
$$\epsilon_i A_i \sigma T_s^4 = Q_{s+A} + Q_{IR} + \sum_{n=1}^{k} \mathcal{F}A_{i-n}\sigma T_n^4$$
$$\sigma T_s^4 = \frac{Q_{s+A} + Q_{IR} + \sum_{n=1}^{k} \mathcal{F}A_{i-n}\,\sigma T_n^4}{\epsilon_i A_i}$$

The equivalent sink for node i may be determined from the detailed thermal math model by determining the adiabatic temperature of node i when node i is disconnected from internal heat paths and heat dissipations. For a transient situation, node i must be an arithmetic node or a low mass node.

EARTH ORBIT ENVIRONMENTAL HEATING

Technical Rationale:

Thermal analysis of an earth orbiting spacecraft requires the accounting of incident thermal energy from all external sources. The most significant external sources of energy incident on the spacecraft are the sun, the thermal radiation of the earth, and the solar energy reflected from the earth (albedo). The modification of the energy incident on the spacecraft due to the earth-sun distance variation, and the accuracy of the measurements of the solar constant, are of sufficient magnitude to be important parameters in performing a thermal analysis.

Impact of Nonpractice:

Not considering the variations in the environmental thermal effects as described in this guideline will result in an incomplete thermal analysis. The temperature variation of the spacecraft could be grossly underestimated, thereby reducing its reliability.

APPENDIX A

Candidate practices and guidelines currently being considered for inclusion in future editions of this document:

Analytical Procedures

- Mechanical component probabilistic design
- Mechanical component redundancy
- Mechanical component failure prediction using Weibull
- Mechanical component nonoperating failure modes
- Mechanical component thermal analysis
- Weibayes criteria for life extension
- Probabilistic methods for inspection of turbine blades
- Reliability growth methodology applications to NASA hardware
- Risk rating of problem/failure reports
- Worst case analysis
- Parts stress analysis
- Piece part thermal analysis of Electronic assemblies
- Internal ESD analysis
- Magnetic field characterization
- Magnetic dipole placements
- Redundancy switching analysis
- Structural stress analysis
- Power transient analysis

Engineering Design

- Shaft design for power systems
- For helicopters/aircraft:
  a. Gears
  b. Bearings
  c. Gear boxes
- Optimization of turbine blades
- Roll rings for high voltage transfer
- Orbital fluid systems
- NiH batteries
- Data Recorders, preferred circuits
- Power supplies, preferred circuits
- Analog preferred circuits
- Digital preferred circuits
- Seals and gaskets
- Springs
- Solenoids
- Valve assemblies

- Bearings
- Gears and splines
- Actuators
- Pumps
- Filters
- Brakes/clutches
- Compressors
- Electric motors
- Batteries
- Lubrication/friction/wear
- Fasteners
- Heat pipes
- Stress corrosion cracking - structure/pressure vessels
- Assessment and control of electrical charge
- Methodology for extending Shuttle life of life limited items
- High permeability materials
- Electrical isolation
- Design practices to control ESD
- Radiated and Conducted Emissions design requirements
- Magnetic design practices
- Radiated and conducted susceptibility design requirements
- Plasma noise coupling in EMI design

Environmental Considerations

- Environmental effects on photovoltaic arrays
- Single event effects on EEE parts due to radiation
- Meteoroid/debris strikes

Test Elements

- Strain range partitioning
- Thermal cycling of photovoltaic arrays
- Testing of Photovoltaic cells
  a. Air mass 0 (AM0)
  b. Charged particle environment
  c. Flash
  d. Are avoidance
- Ultrasonic testing of high voltage PCBs
- Accelerated life testing
- Fault detection /isolation
- Acceptance testing
- Reliability verification of surface mount technology circuit assemblies
- Reliability testing and demonstration of NASA hardware
- Thermal vacuum vs thermal atmospheric testing of electronic assemblies

- ESD tests
- Assembly and system level vibrations testing
- Acoustic tests
- Pyroshock tests
- Thermal - voltage margin testing
- Performance of bearings in high turbo machinery for propulsion systems
- Design verification:

  a. Radiated emissions
  b. Conducted emissions
  c. Radiated susceptibility
  d. Conducted susceptibility
  e. Random vibration

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave blank)*

2. REPORT DATE
September 1991

3. REPORT TYPE AND DATES COVERED
Technical Memorandum

4. TITLE AND SUBTITLE
NASA Reliability Preferred Practices for Design and Test

5. FUNDING NUMBERS

6. AUTHOR(S)
NASA Reliability and Maintainability Steering Committee

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
NASA Office of Safety and Mission Quality
Reliability, Maintainability and Quality Assurance Division

8. PERFORMING ORGANIZATION REPORT NUMBER

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Washington, DC 20546

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
NASA TM-4322

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified - Unlimited

Subject Category 38

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

This document summarizes reliability experience from both NASA and the aerospace industry. It is intended to serve as a guidebook for promoting technical communication and a systematic approach to the reliability discipline in support of current and future civil space programs. The document contains both reliability practices and design guidelines. Practices include recommended analysis procedures, redundancy considerations, parts selection, environmental requirements considerations, and test requirements and procedures that have been successfully applied on previous space flight programs. Guidelines represent "best thinking" on topics where operational experience or information regarding contribution to mission success is lacking.

14. SUBJECT TERMS
reliability, quality assurance, design and test, systems engineering, space systems design

15. NUMBER OF PAGES
88

16. PRICE CODE
A05

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
Unclassified	Unclassified	Unclassified	