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Operational Environments for Electrical Power Wiring on NASA Space Systems

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CONTENTS

Figures		iii
Tables		iii
Acronyms		iv
Acknowledgments		v
Section		**Page**
1.0	**INTRODUCTION**	1-1
2.0	**BACKGROUND**	2-1
	2.1 History	2-1
	2.2 Status	2-1
	2.3 Program Plan	2-2
3.0	**PURPOSE AND OVERVIEW**	3-1
4.0	**SCOPE**	4-1
5.0	**DEGRADATION AND FAILURE OF ELECTRICAL WIRING INSULATION**	5-1
	5.1 Insulation Degradation Types	5-1
	5.2 Insulation Failure Modes	5-2
6.0	**OPERATIONAL ENVIRONMENTS OF WIRING SYSTEMS**	6-1
	6.1 NASA Pressurized Module Operational Environments	6-1
	6.2 NASA LEO/GEO Operational Environments	6-3
	6.3 NASA Trans-atmospheric Vehicle Operational Environments	6-7
	6.4 NASA Lunar and Martian Surface Operational Environments	6-8
	6.5 Aircraft Operational Environments	6-10
7.0	**NASA TESTING REQUIREMENTS FOR SPACECRAFT WIRING**	7-1
	7.1 NASA Spacecraft Design/Operational Factors	7-1
	7.2 Comparison of NASA Requirements to Existing Testing Data	7-2
	7.3 Additional NASA Testing Requirements	7-2
8.0	**CONCLUSIONS**	8-1
9.0	**APPENDIX A: SUMMARY OF EXISTING TEST DATA**	9-1
	9.1 Candidate Wiring Constructions	9-1
	9.2 Department of Defense Wire Testing Programs	9-2
	9.3 NASA Wire Testing Programs	9-5
	9.4 Summary of Tests Performed vs. Wiring Constructions	9-5
10.0	**REFERENCES**	10-1

FIGURES

Figure		Page
1	Acoustic Environment of Payloads for Launch Systems	6-3
2	Number of Eclipses for Satellites in Circular Orbit	6-4
3	Eclipse Seasons in Geosynchronous Orbit	6-4
4	Spectral Distribution of Sunlight	6-5
5	Near Earth Particulate Radiation Department	6-6
6	Dependence of AO Exposure on Solar Activity	6-7

TABLES

Table		Page
1	Space Missions with Wiring System Failures	2-1
2	Operational Environments	6-2
3	Lunar Particulate Radiation	6-9
4	Additional NASA Testing Requirements	7-3
5	Tests Performed vs. Mission Environments Addresses Matrix	7-5
6	Description of Insulation Constructions	9-1
7	WL Aerospace Wiring Test Results - General Properties	9-2
8	WL Aerospace Wiring Test Results - Mechanical Properties	9-2
9	WL Aerospace Wiring Test Results - Electrical Properties	9-3
10	WL Aerospace Wiring Test Results - Environmental Properties	9-4
11	WL Aerospace Wiring Test Program Final Statistical Test Results	9-4
12	NRL Aerospace Wiring Test Results	9-5
13	NASA Wiring Test Results	9-5
14	NASA MAPTIS Test Data	9-6
15	Tests Performed vs. Wiring Construction Matrix	9-7

ACRONYMS

AC	Alternating Current
AFMC	Air Force Mission Command
AM0	Air Mass Zero
AO	Atomic Oxygen
AU	Astronomical Unit = 1 AU = 1.496 x 10^8 km
DC	Direct Current
DOD	Department of Defense
ELV	Expendable Launch Vehicle
EM	Electromagnetic
ESA	European Space Agency
ESH	Equivalent Sun Hours
ETFE	Ethylene Tetrafluoroethylene
FEP	Fluorinated Ethylene Propylene
GEO	Geosynchronous Earth Orbit
GSFC	NASA Goddard Space Flight Center
HSCR	High Strength Crush Resistant
JSC	NASA Johnson Space Center
KSC	NASA Kennedy Space Center
LaRC	NASA Langley Research Center
LDEF	Long Duration Exposure Facility
LEO	Low Earth Orbit
LeRC	NASA Lewis Research Center
MDA	McDonnell Aerospace Corporation
mil	25 μm
MSFC	NASA Marshall Space Flight Center
NASA	National Aeronautics and Space Administration
NEMA	National Electrical Manufacturers Association
NRL	Naval Research Laboratory
OL	Overlap
ORU	Orbital Replacement Unit
P-FP	Polyimide - Fluoropolymer
PFA	Perfluoroalkoxy
PFPI	Partially Fluorinated Polyimide
pH	Hydrogen ion concentration of a solution. Measure of acidity.
PTFE	Poly Tetrafluoroethylene
RH	Relative Humidity
SPL	Sound Pressure Level (dB)
SRB	Solid Rocket Booster
SSF	Space Station Freedom
STS	Space Transportation System
TFE	Tetrafluoroethylene
TKT	TFE (PTFE) - Polyimide - TFE (PTFE) construction
TPT	PTFE, Polyimide, PTFE Tape
TVS	Thermal Vacuum Stability
UV	Ultraviolet
VCM	Volatile Combustible Materials
VUV	Vacuum Ultraviolet
WL	Wright Laboratory
WSTF	White Sands Test Facility
XL	Cross-linked

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SECTION 1. INTRODUCTION

Electrical wiring systems are used extensively on spacecraft and satellites for power management and distribution, control and command, and data transmission. The reliability of the wiring systems when exposed to the harsh environments of space is very critical to the success of the mission and the safety of the crew.

Failures in aerospace vehicles have been reported both on the ground and in flight due to arcing and arc propagation in the wiring harnesses. Arc tracking is the propagation of an arc along wiring bundles, and is made possible by insulation degradation. Therefore, it is necessary to develop arc track resistant wiring insulation in order to minimize, if not eliminate, the risk of failure of critical systems caused by electrical shorts and arc propagation. The arc tracking failure mode represents a more severe risk to the aerospace vehicle than a simple electrical short, due to the difficulty of fault detection, and the possibility of "flashover" of the arc track to adjacent wires, leading to the possible loss of entire wiring harnesses.

A NASA Office of Safety and Mission Assurance (Code Q) program is currently underway to identify and characterize wiring systems in terms of their potential use in aerospace vehicles. Electrical wiring designed for power handling, management and distribution will be characterized in this program. Signal-level cables and wires, which are intended for data transmission and communication, will be excluded because these wire types handle lower power and voltage levels, and therefore are less susceptible to arc tracking. The goal of this program is to provide the information and guidance needed to develop and qualify reliable, safe, lightweight wiring systems, which are resistant to arc tracking and suitable for use in space power applications. New guidelines will be issued once safe operating limits for these systems have been established. This program is being performed by the Electrical Components and Systems Branch, Power Technology Division, at the NASA Lewis Research Center (LeRC), and is managed by NASA LeRC under the top level management of NASA Headquarters, Office of Safety and Mission Assurance, Technical Standards Division (Code QW).

Extensive data already exists from testing performed in recent years on the characteristics of wiring insulations, including the susceptibility to arc-tracking. Therefore, this program is intended to complete the database of testing information on previously analyzed wire types, and consider new insulation constructions and materials not previously evaluated. The wire types which are likely to perform best in each of the different NASA environments will then be identified. The program is divided into three technical tasks: identifying the NASA operational environments (Task #1), performing testing and analysis (Task #2), and analyzing the wiring systems technology (Task #3).

The purpose of this report is to identify the environments which NASA spacecraft will operate, and to determine the specific NASA testing data which needs to be gathered to verify the wiring insulations for NASA use. This data will be valuable to spacecraft designers in determining the best wiring constructions for the different NASA applications. This report contains the background information related to the existence of previous spacecraft wiring failures, and other programs which have addressed the arc tracking phenomenon (Section 2). The various types of insulation degradation which can lead to wiring failures are discussed (Section 5). Then the operational environments which are encountered by spacecraft during different NASA missions are introduced (Section 6), and the testing which needs to be performed to address these conditions are outlined (Section 7). Finally, the testing plans and a summary of existing test data are given in this report (Section 9: Appendix A). This report will be combined with the reports for Task #2 and Task #3, when complete, to form the final program report.

SECTION 2. BACKGROUND

2.1 HISTORY

In the aerospace arena, wiring system failures have proven to be very costly in terms of loss of very expensive equipment, imperilment of missions, and loss of lives. Often, a wiring system failure is not simply the result of inadequate insulation, but it is due to a combination of wiring system factors. These include mishandling of wiring insulation, system designs which expose wires to abnormal stresses, and exposure to fluids which degrade the insulation. Some of the NASA missions with wiring system failures are shown in Table 1.

Table 1. Space Missions with Wiring System Failures [1 - 10]

Mission	Cause	Result
Gemini 8	Electrical Wiring Short	Shortened Mission - Near Loss of Crew
Apollo 204	Damaged Insulation, Electrical Spark, 100% O_2	Fire, 3 Astronauts Lost
Apollo 13	Damaged Insulation/Short Circuit/Flawed Design	Oxygen Tank Explosion, Mission Incomplete
STS - 6	Abrasion of Insulation/Arc Tracking	Wire Insulation Pyrolysis, 6 Conductors Melted
STS - 28	Arc Tracking	Teleprinter Cable Insulation Pyrolysis
Magellan	Wrong Wiring Connection, Wiring Short	Wire Insulation Pyrolysis - Ground Processing
Spacelab	Damaged Insulation/Arc Tracking	Wiring Insulation Pyrolysis During Maintenance
Delta 178/GOES-G	Mechanical or Electrochemical Insulation Damage	Loss of Vehicle
STS-48	Insulation Breakdown - Fluid Exposure	SRB Fuel Isolation Valve Failure
ESA - Olympus	Electrical Wiring Short	Loss of Solar Array

Since the mid-sixties, polyimide (MIL-W-81381) has been the most common material used as wiring insulation in aerospace applications due to its high dielectric strength, low weight, non-flammability, good thermal properties, and high abrasion resistance. However, it has been reported that MIL-W-81381 may undergo some degradation under certain operational environments [4,11-14]. This degradation can lead to arc-tracking, which is the propagation of an arc along a wire. The Navy, which has had an extensive failure history with polyimide wire and has investigated the use of polyimide wire in Naval aircraft thoroughly at its Naval Research Laboratory, has banned the use of polyimide wire (MIL-W-81381) in its aircraft [11-13,15].

2.2 STATUS

NASA will engage in manned and unmanned space activities that will demand larger amounts of electrical power over longer lifetimes than those of current spacecraft, increasing the likelihood of electrical failure. Arc-tracking, which has often not been accounted for in the engineering design, can represent a serious and potentially catastrophic event for aerospace vehicles, and testing to assess the susceptibility to arc tracking is necessary. However, since arc initiation usually results from damage to the wiring insulation, potential for arc tracking can be greatly reduced by implementing specific design features and correct installation procedures. These include: (a) wire separation criteria, (b) physical barriers to arc propagation, (c) advanced circuit protection devices, (d) wiring protection such as wraps and conduit, and (e) proper handling and scheduled maintenance of the wiring systems. System redundancy is also an effective method of ensuring that arc tracking in critical system does not result in a catastrophic failure. In the absence of an electrical insulation which is perfect for every application, it is necessary to consider the overall wiring system when dealing with spacecraft electrical systems. New technologies of fault detection may improve the system safety and therefore need to be investigated.

Recent programs and testing efforts have analyzed the insulation properties for aerospace applications, including the arc-tracking phenomenon, for various insulation types. These programs included those of the Air Force Wright Laboratory (WL) and McDonnell Aerospace Company (MDA) which tested "hybrid" insulations for the aircraft environment, the Johnson Space Center (JSC) which have characterized insulations for the Space Station and Space Shuttle programs, the Marshall Space Flight Center (MSFC) which test insulations for various space environments, the Naval Research Laboratory which investigated the properties of insulation hydrolysis, and the DuPont Corporation which has examined the properties of various DuPont insulations. These programs and testing data focus primarily on experimental testing and evaluation for a range of aircraft operational conditions, or a specific NASA application such as Space Station Freedom or the Space Shuttle Program. This program will establish a comprehensive test matrix and considering the testing performed previously by these groups, will conduct the additional testing needed to evaluate the most promising insulation constructions for use in NASA applications.

2.3 PROGRAM PLAN

A workshop was held at the NASA Lewis Research Center in Cleveland, Ohio in July 1991 to address issues and concerns about electrical wiring for aerospace applications. Scientist and engineers representing several US federal agencies, national laboratories, academia, and private industry, exchanged results and experiences in dealing with a variety of wiring insulation materials. A NASA Office of Safety and Mission Quality (Code Q) program was then established to identify and characterize wiring systems in terms of their potential use in aerospace vehicles.

2.3.1 Objective

The goal of this program is to provide the information and guidance necessary to develop and qualify reliable, safe, lightweight wiring systems, using new wiring insulation and constructions which are resistant to arc tracking and suitable for use in space power applications.

2.3.2 Approach

The approach combines NASA LeRC in-house, other NASA centers, Department of Defense (DOD) laboratories, and contracted efforts to achieve the objective. The program is divided into the following tasks:

Task #1 - NASA Operational Environments: The objective of this task is to identify operational environments as relevant to electrical power wiring for a variety of NASA space missions and vehicles, to evaluate the applicability of the findings of previous aerospace wiring test programs to NASA missions, and to identify the additional testing necessary to identify the best candidate insulation constructions for NASA applications.

Task #2 - Wiring Construction Testing and Analysis: The objective of this task is to evaluate potential insulation systems and to determine their suitability for use in NASA aerospace environments.

Task #3 - Wiring Systems Technology: The objective of this task is to address safety and reliability issues of complete wiring systems. This task will identify related technologies which have an impact on prevention, detection, and isolation of wiring failure and system reconfiguration following failure.

Task #4 - Management Planning: The objective of this task is to plan, manage, and report on the progress of this program.

SECTION 3. PURPOSE AND OVERVIEW

The purpose of the NASA wiring program is to identify and characterize wiring systems which enhance the safety and reliability of aerospace vehicles. This information will provide the basis of new guidelines for the wiring of NASA power systems. To facilitate the acceptance of these guidelines throughout the aerospace community, the input of the various NASA centers and DOD laboratories, industry, and academia has been solicited in the initial phases of the program, and a high level of involvement will continue throughout the duration of the program.

The military has developed an extensive database of testing information for aircraft wiring systems, identifying potential materials and wiring constructions and their arc tracking behavior. However, the NASA environments, and therefore the insulation requirements, may be significantly different than for aircraft. These include wiring systems which must operate in Earth-orbiting satellites, inside pressurized modules, on the lunar and martian surfaces, and in trans-atmospheric applications, such as the Space Shuttle and other launch vehicles. The NASA program extends beyond the existing testing database for wiring to completely address the effects of the NASA unique mission environments. The operational environments for space missions, and the existing testing databases for wiring systems, are presented for use by spacecraft design engineers.

SECTION 4. SCOPE

The scope of this program, and therefore this report, is limited to the electrical wiring of "conventional" power systems. The electrical operating conditions of such systems are discussed in Section 6. Wiring for applications other than power handling, such as for data transmission and communication, while not required to meet the electrical requirements of power wiring, are still expected to be subject to the space environments defined in this report. Wiring systems with extreme electrical requirements, such as lunar surface transmission of power at very high voltages (kV's), are not considered to be within the scope of this program. Additionally, other potential space operational conditions, such as the high temperature and radiation environment in the close vicinity of a space nuclear reactor, are not included in this program.

In this report, the operational environments for aircraft and space missions are presented and discussed. The classifications of NASA missions which are considered in this report are as discussed below:

a. Pressurized Module Environments: These include the manned environment of the space shuttle, and the Space Station Freedom habitation and laboratory modules, which are characterized by an enriched oxygen environment. The operational effects of launch/decent of the pressurized modules are also included in this environment.

b. Low Earth Orbit/Geosynchronous Earth Orbit (LEO/GEO) Environments: These include the LEO orbiting Space Station Freedom, and the many satellites (i.e. communications, remote sensing) which are positioned throughout the LEO and GEO orbits. Again, the launch operational effects are included in this case.

c. Trans-atmospheric Vehicle Environments: These include the operational environments of the space shuttle (manned) and the expendable launch vehicles (unmanned) as they travel from the Earth's surface to space. The operational temperature ranges (hot and cold) of the NASA applications which are considered in this program will be given in Section 6. Extreme temperatures outside of these ranges, such as engine heat during launch, friction upon vehicle re-entry, or deep space, may require specialized insulations, and are beyond the scope of this program.

d. Lunar and Martian Environments: Included will be permanent outposts on the lunar and martian surfaces. Like the other cases, the operational effects of launch/descent will be included in this environment.

The testing which was performed to verify the insulations for the aircraft environment are in many cases sufficient for NASA missions as well. There are, however, environments which are unique to NASA spacecraft. The testing required for wiring which will operate in each of the NASA application environments, as compared to the testing already performed, will be outlined in Section 7.

SECTION 5. DEGRADATION AND FAILURE OF ELECTRICAL WIRING INSULATION

This section discusses the various forms of wiring insulation degradation and failure modes which can occur when the wiring is in operation on spacecraft. In Section 6, the environmental conditions which will be encountered by NASA spacecraft will be introduced, and in Section 7, the testing which is needed to evaluate wiring constructions for use on NASA spacecraft, when the environments, spacecraft design, and mission operations are considered, will be determined.

5.1 Insulation Degradation Types

5.1.1 **Abrasion**. Vibration and tight confinement can result in abrasion, causing wire damage during flight and while servicing. The vibration during launch can result in abrasion damage to both the launch vehicle and its spacecraft cargo [4,16]. While maintenance procedures can result in high levels of insulation degradation as has been experienced on the Space Shuttle [9].

5.1.2 **Atomic Oxygen (AO) Effects**. Exposure to AO erodes certain insulation materials to the point of causing a loss of mass, which results in a reduced insulation thickness and a change in the functional properties [17]. Additionally, the synergistic effects of AO and UV radiation exposure can be more extreme than either effect when considered separately [14].

5.1.3 **Charged Plasma Effects**. In the GEO environment, a low-density, high energy charged plasma can lead to differential charging of different spacecraft parts. At times of solar substorm activity, the plasma interactions can be elevated to levels exceeding the breakdown voltages, leading to and arc between spacecraft surfaces [18-20]. The Low Earth Orbits have a high-density, low energy plasma. This type of plasma does not ordinarily lead to differential charging, but to charging of the spacecraft surfaces with respect to the surrounding plasma. Arcs in LEO occur from conductor-insulator junctions (including holes in wiring insulation) when the conductor is highly negative when compared to the surrounding plasma [20]. It has also been shown that the combined effect of debris impacts and the charged plasma environment can result in the vaporization and ionization of material during the impact, and the initiation of an arc [21]. For wiring, the plasma environment would only be a factor for externally exposed wires

5.1.4 **Electrical Breakdown**. The electrical breakdown of the insulations can result from corona, which is the discharge of an electrical arc from the wire conductor to a point of lower potential, either another conductor or simply the surrounding space. According to the Paschen relationship, with increasing voltage the possibility of corona rises, with the inverse relationship holding for pressure [16].

5.1.5 **Electromagnetic Radiation**. Electromagnetic radiation includes ultraviolet (UV) light, x-rays, and gamma rays. Oxidation is the most severe damage due to UV radiation, but this will only occur where oxygen is present, such as on the Earth's surface. However, UV radiation is generally minor. Nonetheless, the UV radiation can cause insulation embrittlement as a result of chain scission or cross-linking, reduce insulation mass, and also cause color changes [16,22]. Additionally, the synergistic effect of UV and AO exposure can cause even more severe mass loss [14].

5.1.6 **Hydrolysis**. Hydrolysis is a degradation of the insulation material due to exposure to certain fluids or moisture; it results in loss of strength, causes embrittlement, and makes the insulation material susceptible to cracking. The hydrolytic reaction reduces the polymer chain length and renders it weak and brittle, with a tendency to crack radially at sharp bends [4,17]. The exposure of insulation to strong alkalis (high pH solutions) accelerates the rate of degradation from negligibly small to unacceptable [17,23].

5.1.7 **Insulation Cut Through**. Cuts or notches in the insulation can be caused by a number of mechanical stresses including vibration, maintenance procedures, and meteoroids and debris. These can damage the insulation integrity, rendering the wires susceptible to arc initiation. Meteoroids and debris impacting the surface of the wiring can create holes in the insulation, even particles less than 1 mm in diameter may be damaging because of the frequency and velocity of impact, while large pieces of debris could completely sever a wire [16].

5.1.8 **Outgassing(Thermal Vacuum Stability)**. Outgassing occurs at low pressures or in a vacuum, where molecules with relatively low weight fractions, unreacted additives, contaminants, adsorbed (on surfaces) and absorbed (in bulk) gases, or moisture evaporate. The loss of these additives and contaminants can change important properties of the insulation. For example, the loss of a plasticizer by evaporation in a vacuum environment will produce a more rigid or brittle material, with a corresponding decrease in elongation and increase in tensile and flexure strength. Chemical changes may occur when water and gases gradually diffuse out of the material, which can lead to the degradation of the wiring insulation [16].

5.1.9 **Particulate Radiation**. The particulate radiation environment is composed of cosmic rays, Van Allen belt radiation, aurora particles, and solar flare particles. It consists of electrons, protons, neutrons, alpha particles, and others. The damage is dependent upon the energy and type of particles. Radiation damage such as removal of a bonded electron leading to bond rupture, free radicals and discoloration can occur. The result can include the loss of mechanical strength, an increase in vapor pressure and viscosity, and a reduction in molecular weight [16].

5.1.10 **Temperature Effects**. Under normal gravity and atmospheric conditions, gasses can provide cooling to an overheated or burning wire due to convection processes. However, in space, hot gasses can remain stagnant in the area of the heat source. Because of the absence of most heat transport systems, space wiring systems may be subjected to extreme high and low temperatures. Short of direct thermal damage, the effects of aging may be accelerated under temperature extremes. Elevated temperatures can cause damage such as softening, melting, and chemical decomposition, while extremely cold temperatures can cause some insulations to become brittle [16,24].

When exposed to thermal cycling, the cable conductors and insulations, which have different coefficients of thermal expansion, will experience mechanical stress each thermal cycle. Repeated tensile and compressive forces will react against the connector pins, assuming that the connectors are restrained. Although small, the movement is cyclic and continuous for the life of the cable insulation. Some polymers have a "memory" in that they tend to crease, stretch, or fold in the same place once this action has occurred. Repeated creasing or stretching in the same place will eventually lead to insulation failure [25].

5.2 Insulation Failure Modes

5.2.1 **Arc-Tracking**. Arc tracking in electrical wire insulation has recently been identified as a failure mode that can cause extensive damage to aircraft wire harnesses and possible secondary ignition of other materials. Arc tracking occurs when an insulated wire sustains a propagating arc at a certain current or voltage. An electrical arc can be produced due to a short circuit, overload current, or localized stressing of the wiring systems. Wiring insulations can become susceptible to electrical arc propagation when mechanical, chemical, or thermal damage has occurred (dry arc tracking), or when a conductive fluid is present (wet arc tracking). Once initiated, the arc can propagate along the wire or to adjacent wires (flashover) causing a circuit malfunction. The rate and extent of arc tracking depends on the type of insulation material and construction, applied power and frequency, wire gauge, and environmental factors, such as temperature, pressure, and humidity. When relating voltage level to arc-tracking, tests have shown that in general the probability of arc-tracking becomes greater as the voltage level rises, with the exception of a possible dip at intermediate voltages (120 to 160 Volts) [16,26].

In some cases, the conductive path of the carbon arc track displays a high enough resistance such that the current is limited, and therefore may be difficult to detect using conventional circuit protection. Tests at the NASA Johnson Space Center (JSC) have shown that for the space shuttle power system, arc-tracking was limited to lengths of less than 1" up to 6" [17,25-28]. New fault detection technology may improve the detection of these faults.

5.2.2 <u>**Combustion**</u>. The combustion of materials which can result from an electrical arc is influenced by the percent oxygen and pressure level in the area where the wire is operating. MIL-W-81381 was chosen because of its favorable properties, including being non-flammable. To be considered for NASA applications, all of the insulation materials must also be non-flammable [17,23]. An enriched oxygen concentration can increase flame-spread rate, and increase extinguishment difficulty [24].

5.2.3 <u>**Electrical Short**</u>. An electrical short is the most common form of failure occurrence in electrical wiring. Arc tracking (see above) is a secondary type of failure which can occur as a result of a short circuit fault. Most short circuits result in extremely high currents, and are interrupted by the protection systems. However, even momentary short circuits can result in permanent insulation damage or the initiation of combustion. The extent of damage which occurs as a result of a short circuit fault is dependent upon the circuit protection, wiring insulation, and operational environment.

SECTION 6. OPERATIONAL ENVIRONMENTS OF WIRING SYSTEMS

The specific electrical, mechanical, and environmental conditions for the NASA missions and military aircraft are given in Table 2, and discussion and references are provided in this section. The NASA mission environments can then be compared to the aircraft environment (Section 7), for which testing has already been performed in the WL testing program (Section 9). This comparison, combined with the database of testing results which currently exists from other DOD and NASA programs (Section 9), will yield the additional testing which needs to be performed in this program. The specific operational conditions to be addressed, as defined in Section 4, consist of the NASA pressurized modules, low Earth orbit (LEO) spacecraft, geosynchronous Earth orbit (GEO) spacecraft, trans-atmospheric vehicles, lunar surface and Mars surface missions.

6.1 NASA Pressurized Module Operational Environments

6.1.1 Electrical. The space shuttle has a 28 VDC power system, supplied by 3 primary fuel cells which can supply a total of 7 kW of steady state power to the payload and habitat region of the vehicle [29-32]. The original Space Station Freedom design had a total power requirement of 75 kW from four 18.75 kW solar modules. Presently the design requirement has been scaled back to 56.25 kW due to the elimination of the fourth solar power module [33]. The distribution voltage inside the Space Station Freedom pressurized modules, between the internal DC to DC converter units and the loads is 120 VDC. At the loads the distribution voltage is 28 VDC [34].

6.1.2 Temperature. The interior temperature of the pressurized modules is to be regulated for habitation by astronauts, these temperatures can range between 18.3°C and 26.7°C [35]. Because of the regulated temperature, thermal cycling is not a significant factor.

6.1.3 Atmospheric. In the NASA pressurized environments, the atmosphere will be nominally air (21% oxygen, 101 kPa total pressure) enriched to 30% oxygen at 69 kPa total pressure for prebreathing prior to an extravehicular activity [35,36].

6.1.4 Vibration. The only significant vibration which the pressurized environments will experience are during launch, since very little vibration will occur while in orbit. The launch vibration environment, depending upon the launch configuration, can approach values of acceleration as high as 10g and frequencies approaching 1000 Hz [37].

For both the Space Shuttle and expendable launch vehicles, vibration induced from acoustic fields may be an important factor. After ignition, the intensity of the acoustical fields from the rocket engine exhaust increase until lift-off. As the launcher rises, the strength of the field reflected from the ground decreases. The acoustic field experiences a second increase as the vehicle approaches the speed of sound due to aerodynamic disturbances. A vibrational response results, which can be many times larger than the structurally transmitted vibration. For typical launch vehicles, the acoustic noise environment has a sound pressure level of 137 to 145 dB as shown in Figure 1 [16,37].

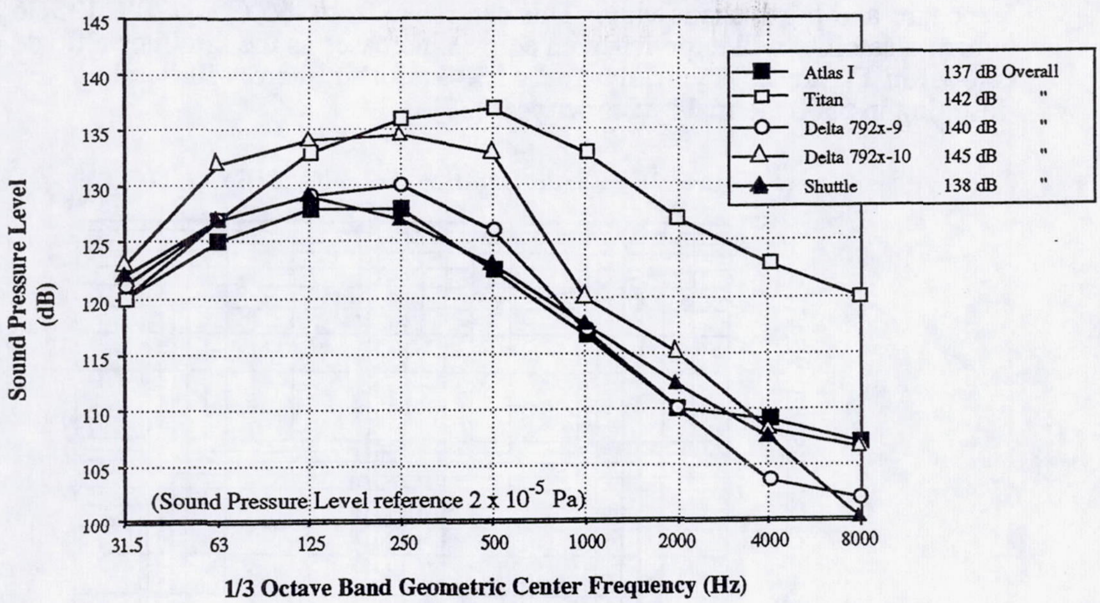
6.1.5 Meteoroid/Debris. Obviously, if space debris or a meteorite penetrates a pressurized module, the damage caused to wiring insulation will be a trivial part of the overall hazard.

6.1.6 Gas/Fluid. For NASA pressurized modules, the relative humidity (RH) is not to exceed 70% or fall below 25% [35]. Most NASA spacecraft will experience a worst case humidity environment when being assembled, stored, and transported before launch, as the environment is not always regulated, and can rise to the earth's humidity level of up to 100% [23]. While the launch vehicles are on the launch pad, the payload environment is regulated to 50% RH or less [3,15,29,38].

Table 2. Operational Environments

	Pressurized Modules	Low Earth Orbit	Geosynchronous Earth Orbit	Trans-atmospheric Vehicle	Lunar Surface	Martian Surface	Military Aircraft
Electrical							
Voltage	$28 \rightarrow 120$ V	$28 \rightarrow 160$ V		$28 \rightarrow 270$ V	$28 \rightarrow 160$ V		28 V
Frequency	DC				DC $\rightarrow$ 20 kHz		DC $\rightarrow$ 400 Hz
Mechanical							
Vibration	$1 \rightarrow 10$ g $137 \rightarrow 145$ dB SPL						25 μm amplitude 500 Hz
Particle Impacts	N/A	$11 \rightarrow 26$ impacts/m^2/yr (Altitude Dependent)	< LEO	LEO $\rightarrow$ GEO (Altitude Dependent)	$0.01 \rightarrow 0.5$ impacts/m^2/yr	Very Low Probability	N/A
Environmental							
Temperature	18.3°C $\rightarrow$ 26.7°C	-65°C $\rightarrow$ 120°C 6000 cycles/yr	-196°C $\rightarrow$ 128°C 90 cycles/yr	-200°C $\rightarrow$ 260°C Cycles Altitude Dependent	-171°C $\rightarrow$ 111°C 13 cycles/yr	-143°C $\rightarrow$ 27°C 356 cycles/yr	-65°C $\rightarrow$ 230°C
Atmosphere	Earth $\rightarrow$ 30% O_2	Earth $\rightarrow$ Very Low O_2				Earth $\rightarrow$ 0.13% O_2, 95.3% CO_2	Earth Atmosphere
Gas/Fluid Compatibility	$25 \rightarrow 75$% RH 100% RH Salt Fog Space Fluids	100% RH Salt Fog Space Fluids					$25 \rightarrow 75$% RH Aerospace Fluids
Pressure	$517 \rightarrow 760$ Torr	$10^{-5} \rightarrow 10^{-10}$ Torr	7.5×10^{-14} Torr	$760 \rightarrow 7.5 \times 10^{-14}$ Torr	$10^{-8} \rightarrow 10^{-12}$ Torr	$4.4 \rightarrow 11.4$ Torr	$49 \rightarrow 760$ Torr
Electromagnetic Radiation	N/A	$2220 \rightarrow 5800$ ESH/yr (Altitude Dependent)	8760 ESH/yr	8760 ESH/yr (Altitude Dependent)	8760 ESH/yr	1656 ESH/yr	Earth UV
Particulate Radiation	N/A	Protons, α Particles, and Electrons				N/A	N/A
Atomic Oxygen	N/A	10^{20} - 10^{22} atoms/cm^2/yr (Altitude Dependent)	N/A	LEO $\rightarrow$ GEO (Altitude Dependent)	N/A	N/A	N/A
Reduced Gravity	$10^{-3} \rightarrow 10^{-6}$ g	$10^{-3} \rightarrow 10^{-6}$ g		$1 \rightarrow 10^{-6}$ g	0.165 g	0.38 g	N/A
Charged Plasma	N/A	$0.3 \rightarrow 5 \times 10^4$ atoms/cm^3 $0.1 \rightarrow 0.2$ eV	$0.24 \rightarrow 1.12$ atoms/cm^3 $120 \rightarrow 295$ keV	LEO $\rightarrow$ GEO	N/A	$10^3 \rightarrow 10^5$ atoms/cm^3	N/A

Figure 1. Acoustic Environment of Payloads for Launch Systems [37]



All wiring for space applications should be compatible with typical space fluids such as hydrazine, hydraulic fluid, monomethylhydrazine, nitrogen tetroxide, and ammonia as well as others that may result in serious degradation of the wiring insulation.

6.1.7 Pressure. The pressurized modules are regulated at constant pressures of 69 to 101 kPa (10 - 14.7 psi, 517 - 760 Torr) [36].

6.1.8 Electromagnetic Radiation. The pressurized modules will shield the wiring from exposure to the ultraviolet radiation of the space environment.

6.1.9 Particulate Radiation. The pressurized modules, as a basic requirement, must be radiation resistant enough to enable human habitation. The resistance of electrical insulations to particulate radiation will be better than that of humans or electronics [39].

6.1.10 Atomic Oxygen. Not Applicable.

6.1.11 Gravity. In LEO, the residual gravity is not zero, but ranges from < 10^{-6}g to 10^{-3}g. While the fundamental aerodynamic minimum level is 10^{-6}g, the 10^{-4}g to 10^{-3}g range will result due to venting forces, station keeping thrusters, crew motion, and the gravity gradient [30]. The payloads can also experience accelerations of up to 8g during launch [37].

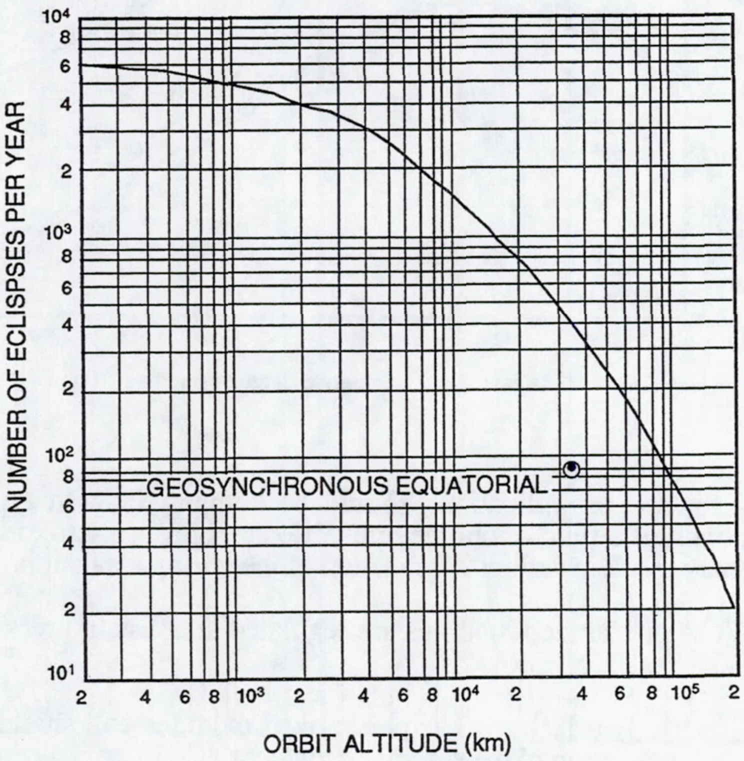
6.1.12 Charged Plasma. All of the wires for the Space Station Freedom pressurized modules are assumed to be inside of the pressurized environment, and therefore unexposed to the plasma.

6.2 NASA LEO/GEO Operational Environments

6.2.1 Electrical. In general, the vast majority of satellites built to date for both LEO and GEO have power distribution at a potential of 28 VDC [34]. The power levels of present LEO and GEO spacecraft range in power capability from a few Watts to a few kW. The highest flown to date was Skylab, which had an average power level of 8 kW [40,41].

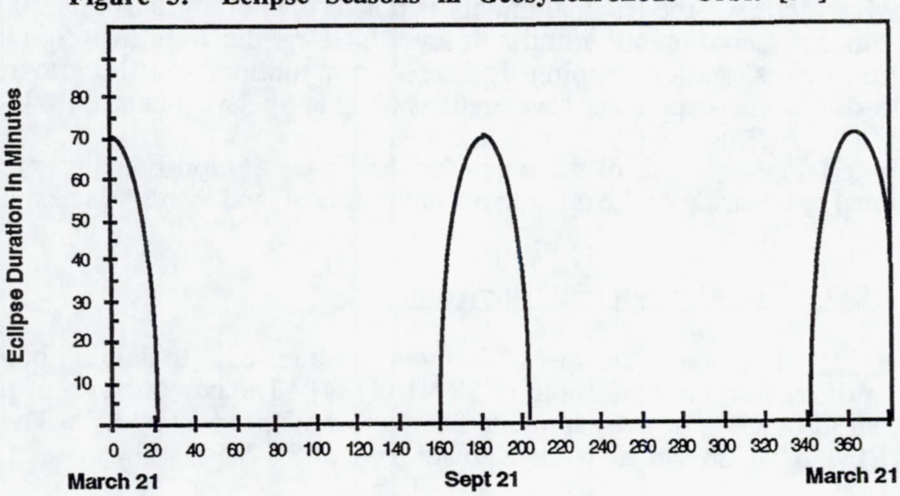
6.2.2 **Temperature**. The temperature environment in LEO is extreme due to the contrast between Sun exposure and Earth shadowing. This can range from -65°C to 120°C [25,36]. The frequency which a satellite will experience an eclipse increases as the satellites altitude decreases, this is shown in Figure 2. Typically, for a 550 km orbit, there will be about 15 eclipses per day, resulting in 5500 thermal cycles per year [37].

Figure 2. Earth Orbiting Satellite Eclipses [16]



In GEO, the eclipse seasons are as shown in Figure 3. There are seasons of about 45 days, twice per year, with a maximum shadow time of 1.2 hours per day. As a result, there will be 90 thermal cycles per year. The approximate temperature range of these cycles is expected to be from -196°C to 128°C [37,42].

Figure 3. Eclipse Seasons in Geosynchronous Orbit [37]



6.2.3 **Atmospheric**. The lack of significant oxygen or pressure makes a fire on orbit an impossibility unless it occurs in the presence of oxygen such as inside a fuel cell oxygen tank [8].

6.2.4 **Vibration**. See discussion for Pressurized Modules Section 6.1.4.

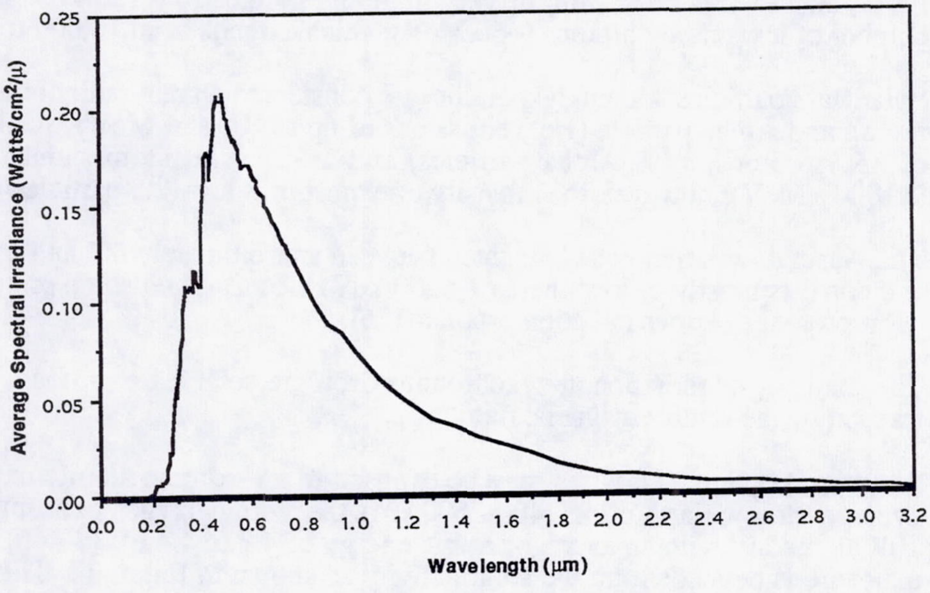
6.2.5 **Particle Impacts**. The LEO environment is more severe than the GEO environment, and can result in an impact flux of 11 to 26 impacts/m²/year for particles of significant size [43]. Smaller particles can be significant if there is a charged plasma environment because an arc can result, or if the surface has important optical properties [21].

6.2.6 **Gas/Fluid**. See discussion for Pressurized Modules Section 6.1.6.

6.2.7 **Pressure**. The LEO environment will have pressures ranging from 10^{-8} to 10^{-3} Pa (10^{-10} to 10^{-5} Torr) [19,44]. The GEO environment will have pressures which approach the interplanetary value of 10^{-11} Pa (7.5×10^{-14} Torr) [45].

6.2.8 **Electromagnetic Radiation**. The total energy received from the Sun per unit area at 1 Astronomical Unit (AU) is the solar constant, and it equals 1353 W/m² at air mass zero (AM0). The region of the solar spectrum which contains 99.5 percent of the total energy is the region from 0.12 μm to 10 μm, or the UV, visible, and IR bands as shown in Figure 4. The energy of the radiation is inversely proportional to the wavelength, therefore the ultraviolet light is higher in energy than the visible or infrared. The energy of the UV radiation can be high enough to cause excitation, if not ionization, in some materials. Over the whole UV wavelength range, up to 400 nm, the intensity average is 118 W/m² [16].

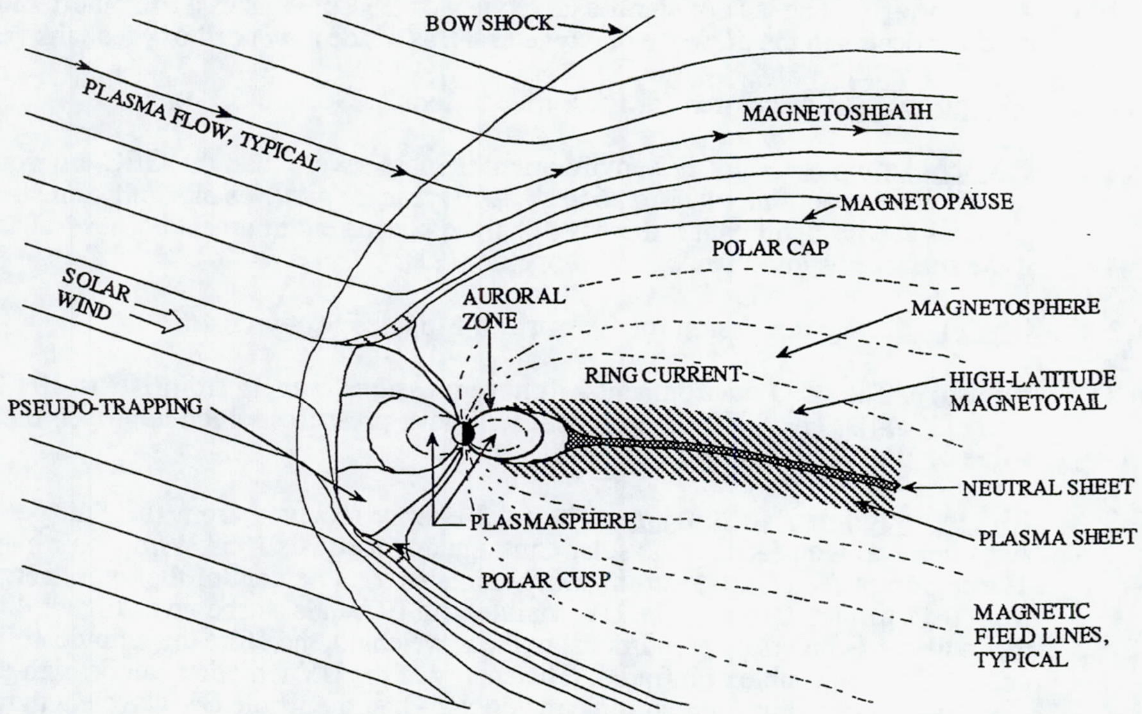
Figure 4. Spectral Distribution of Sunlight [16]



Other types of Electromagnetic radiation include x-rays and gamma rays having wavelengths of approximately 10^{-8} cm. Although these types of radiation are high in energy, they are not found in high enough levels to be a significant part of the natural radiation environment [16].

6.2.9 **Particulate Radiation**. Particulate solar radiation consists primarily of the protons and electrons of the solar wind. These are trapped by the Earth's magnetic field forming the Plasmasphere or the Van Allen radiation belts. The formation of the Earth radiation environment is shown in Figure 5.

Figure 5. Earth Particulate Radiation Environment [16]



The plasmasphere is divided into two zones, an area of mostly low energy electrons (20 keV to 1 MeV) and high energy protons (≥600 MeV) extending from about 480 km to 6400 km above the Earth, and an area consisting of very high energy electrons (20 keV to 5 MeV) with a small number of low energy protons (~60 MeV) which extends from 16,000 to 58,000 km [16,22].

Solar flare particles, although sporadic, are considered a great radiation hazard, as they result in proton and Alpha particle (He^+) emissions of up to 100's of MeV's. Galactic radiation consists of 85% protons, 14% Alpha particles, and 1% heavier atoms, and have very high energy (>1000's MeV); however, the intensity of exposure is low (2.5 particles/cm^2s) [16,22].

The Auroral radiation zone is located between approximately 60° and 65° geomagnetic latitude and consists mostly of low energy (< 200 keV) electrons and some protons. These particles do not represent a serious radiation problem [16].

The majority of radiation at synchronous orbit are solar flare protons, as opposed to particles trapped by the Earth's magnetic field [16].

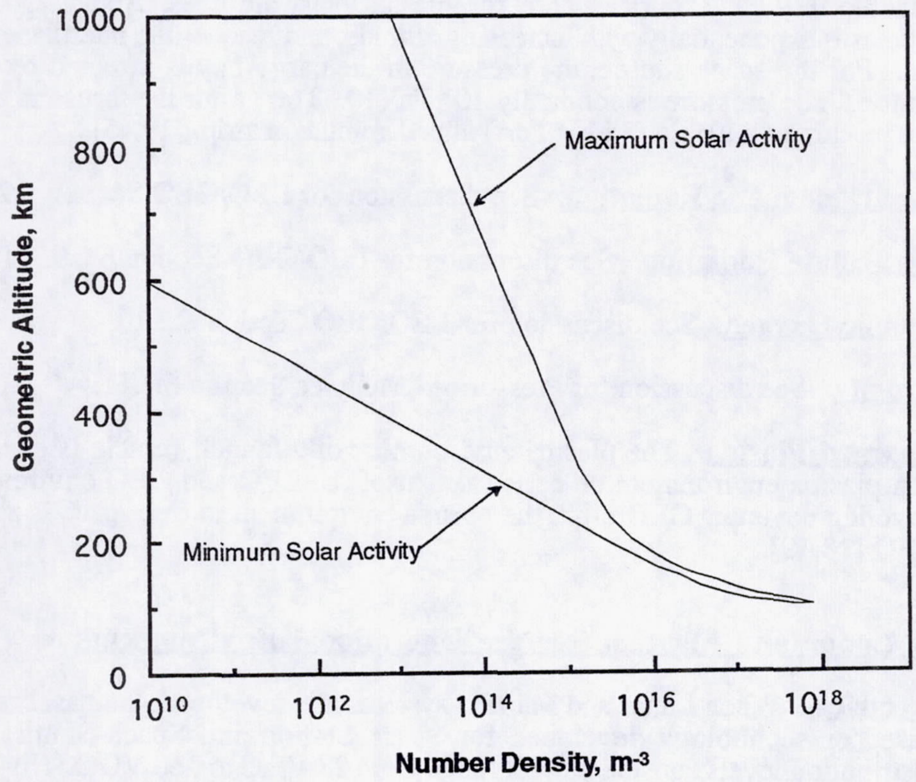
6.2.10 **Atomic Oxygen**. The wires may be exposed to an extreme atomic oxygen environment. On average for low Earth orbits (400 - 500 km), the atomic oxygen exposure can range from 10^{11} - 10^{12} atoms/m^3 with an average atomic energy of 4.3 to 4.4 eV [43,46]. However, the level of exposure is dependent on the solar activity as shown in Figure 6. This leads to an equivalent fluence for testing of wiring insulation of 10^{20} - 10^{22} atoms/cm^2/year [14,46].

6.2.11 **Gravity**. See discussion for Pressurized Modules Section 6.1.11.

6.2.12 **Charged Plasma**. The plasma environment for LEO has an ion density (positive or negative) of 3×10^4 to 9×10^5 cm^{-3} depending upon the degree of solar activity and orbit position. The thermal energies of the electrons and positive ions are in the range of 0.1 to 0.2 eV, corresponding to temperatures of 1200 K to 2400 K. In general, the plasma in LEO is a high density, low energy environment and can lead to arcing from exposed surfaces to the ambient plasma [18-20,36].

The GEO plasma environment is a low density, high energy plasma, and can result in significant charging which can lead to arcing between spacecraft surfaces. The worst case geosynchronous plasma environment, which should be used in predicting spacecraft potentials, are given by ion and electron densities from 0.24 to 1.12 cm^{-3} and thermal energies of 120 to 295 keV [18-20].

Figure 6. Atomic Oxygen Density Levels [47]



6.3 NASA Trans-atmospheric Vehicle Operational Environments

6.3.1 **Electrical**. The space shuttle has a 28 VDC power system, supplied by 3 primary fuel cells, while the power distribution systems for the expendable launch vehicles are from 28 to 270 VDC. The space shuttle's primary fuel cells can supply a total of 7 kW of steady state power to the payload and habitat region of the vehicle [29-32].

6.3.2 **Temperature**. For the space shuttle, the temperature range is -156°C to 200°C, while for expendable launch vehicles the range is -200°C to 260°C [3,29].

The launch vehicle will endure thermal cycling dependent upon its altitude and duration in orbit. The space shuttle in LEO will endure frequent thermal cycling with great extremes (-65°C to 120°C); however, it is only in orbit for a few days at a time. An expendable launch vehicle is only operational in orbit for a few minutes/hours, so thermal cycling is not a concern.

6.3.3 **Atmospheric**. See the discussion for the LEO/GEO environment (6.2.3), except for the pressurized regions of the space shuttle, which are included in the Pressurized Module discussion (6.1.3).

6.3.4 **Vibration**. See the discussion for Pressurized Modules Section 6.1.4.

6.3.5 **Particle Impacts**. See the discussion for LEO/GEO Section 6.2.5.

6.3.6 Gas/Fluid. See discussion for Pressurized Modules Section 6.1.6. For the Space Shuttle, the only areas with a controlled humidity environment are the payload bay and crew cabin. The humidity is not controlled during the roll out from the KSC Vehicle Assembly Building to the launch pad [23].

6.3.7 Pressure. The trans-atmospheric vehicle pressures will range from 101 kPa (760 Torr) at Sea Level to 10^{-11} Pa (7.5×10^{-14} Torr) for interplanetary altitudes. Above 2500 km, the gas pressure decreases exponentially with increasing altitude, and reaches the interplanetary value near 20,000 km. For the space shuttle, the pressure in the cargo bay is affected by the movement of the shuttle. The pressure is nominally 10^{-4} Pa (10^{-6} Torr) after the thrusters are fired for 1 second, and reaching 4×10^{-4} Pa (3×10^{-4} Torr) after 1 minute of firing[19,45].

6.3.8 Electromagnetic Radiation. See discussion for LEO/GEO Section 6.2.8.

6.3.9 Particulate Radiation. See discussion for LEO/GEO Section 6.2.9.

6.3.10 Atomic Oxygen. See discussion for LEO/GEO Section 6.2.10.

6.3.11 Gravity. See discussion for Pressurized Modules Section 6.1.11.

6.3.12 Charged Plasma. The plasma environment of a launch vehicle is expected to range from a non-plasma environment on earth, to that of the LEO and GEO environments. For a vehicle traveling beyond a GEO orbit, the plasma environment in free space is approximated by that at GEO [18,19].

6.4 NASA Lunar and Martian Surface Operational Environments

6.4.1 Electrical. When Lunar and Martian systems are developed, the baseline system will likely be based on technology developed for Space Station and Space Shuttle programs. Electrical distribution levels considered will range from 28 VDC to 160 VDC. High frequency AC power distribution may also be considered, because of the expanding power requirements and utility type system requirements [48]. Total power system requirements for permanently manned outposts have been proposed ranging up to 150 kW for Mars and 725 kW for the Moon. However, recent estimates for the Moon and Mars power requirements are 50 kW and 20 kW respectively. Additionally, systems with even lower power levels have been proposed as a way to reduce the mission costs [33,48-51].

6.4.2 Temperature. The surface temperature of the Moon, due to the lack of an atmosphere, varies greatly during the day and night cycles. During the 364 hour lunar day, the temperature can rise to 111°C, while during the equally long lunar night, the temperature can drop to -171°C. The lunar day/night cycle of 28 days results in 13 cycles in 1 Earth year [45].

The existence of the thin Martian atmosphere, and the decreased solar intensity (590 W/m^2) due to the increased distance from the sun (1.5 AU), result in a smaller temperature range than on the lunar surface. During the daytime, the temperature is a moderate 27°C, while at night, the temperature can drop to -143°C. Similar to the lunar surface case, the Mars thermal cycles are dependent upon the day/night cycles (12.3 hr day/12.3 hr night). Therefore, 356 cycles per Earth year will occur [45,52].

6.4.3 Atmospheric. See discussion for LEO/GEO Section 6.2.3.

6.4.4 Vibration. See discussion for Pressurized Modules Section 6.1.4.

6.4.5 Particle Impacts. The Moon is bombarded with meteoroids and micrometeoroids. These micrometeoroids are the result of cometary debris, interstellar grains, and lunar ejecta. With no protection from an atmosphere, impact velocities range from 24 to 72 km/s. The actual hazard is small for meteoroids with pit sizes approximately equal to 500 μm in diameter, which have an impact rate between 0.01 to 0.5 impacts/m^2/yr. Additionally, meteoroids of approximately 100 grams will have an impact rate of 1.2 x 10^{-4} impacts/km^2/yr, with larger meteoroids impacting at even lower rates [45].

On Mars, due to the atmosphere, the probability of meteoroid impact is very low [52].

6.4.6 __Gas/Fluid__. See discussion for Pressurized Modules Section 6.1.6. Additionally, on Mars the average amount of atmospheric water vapor is 0.03% by volume. The water vapor concentration is closely linked to the temperature distribution; during most of the year and at most latitudes the atmosphere holds all of the water possible (100% relative humidity) [45].

Build up of dust particles on lunar surface wiring is likely to occur due to surface activities. Direct degradation of the insulation in contact with the lunar soil may not be an issue, but the secondary effects of dust accumulation must be considered. Studies have estimated that a layer of dust more than about 11 μm would not only effectively block thermal radiation, but it would also limit all heat transfer to the insulation, due to the poor conductivity of the lunar soil [53].

6.4.7 __Pressure__. Due to the fact that there is no continuous "lower atmosphere", the "upper atmosphere" extends to the surface of the Moon and it is nearly a collisionless gas. The density is approximately 2 x 10^5 molecules/cm^3 at night corresponding to a pressure of 10^{-10} Pa (10^{-12} Torr) [45,54].

The surface atmospheric pressure on Mars varies with the seasons, ranging from 600 to 1500 Pa (4.4 to 11.3 Torr) [45].

6.4.8 __Electromagnetic Radiation__. When the lunar surface is exposed to the Sun, the solar flux is 1371 W/m^2, due to the lack of significant atmosphere. The ultraviolet radiation exposure on the lunar surface will be equal to 1 UV sun [45].

The solar radiation on the surface of Mars ranges from 590 W/m^2, without forward scattering by small particles in the Mars atmosphere, to 649 W/m^2, if the scattering is taken into account [43,45].

6.4.9 __Particulate Radiation__. The particulate radiation exposures on the lunar surface are shown in Table 3. The solar wind consists mostly of H and He nuclei, and is the dominant source of the lunar particulate radiation. Surface features exposed to the solar wind, with a velocity of approximately 400 km/s, are gradually smoothed by a process known as sputtering. However, this process poses little hazard to equipment on the surface, for example the erosion lifetime of a rock with a diameter of only 10^{-6} m is estimated to be 10^5 years [45]. Solar cosmic rays are ejected from the sun during solar flares, and are dominated by the nuclei of light atoms (H, He) while heavier nuclei (Ca, Fe) also occur. These high energy particles can penetrate the lunar surface materials to a depth of 1 cm [45]. Galactic cosmic rays are charged particles from outside the solar system, and are the most energetic particles to reach the lunar surface. They have lunar penetration depths that exceed 1 m, but the flux rate is significantly lower than that of solar cosmic rays [45].

Table 3. Lunar Particulate Radiation [45]

Source	MeV/nucleon	Proton Flux ($cm^{-2}sec^{-1}$)	Penetration Depth (cm)
Solar Wind	10^{-3}	10^8	10^{-6}
Solar Cosmic Rays	1 to 10^2	10^2	10^{-3} to 1
Galactic Cosmic Rays	10^2 to 10^4	1	1 to 10^3

On Mars, the atmosphere below 120 km is dominated by CO_2. Just below 130 km, the ion concentrations of O_2^+ and CO_2^+ are at their peak of 10^5 cm^{-3} and a temperature of 150 K. At altitudes of 225 km, O^+ ions are at a peak of 10^3 cm^{-3}, with a temperature of 210 K. Above this altitude, departure from thermal equilibrium with the neutral gas occur, and the ion temperature increases rapidly to over 1000 K [45].

6.4.10 Atomic Oxygen. Not Applicable.

6.4.11 Gravity. The gravity on the surface of the Moon, because it is less massive than the Earth, is 0.165 g or 162.3 cm/s^2. On Mars the gravity is 0.38 g or 372.52 cm/s^2 [45].

6.4.12 Charged Plasma. At the Moon, the solar wind plasma consists mostly of charged H and He nuclei. Additionally, solar cosmic rays ejected from the sun during solar flares, and Galactic cosmic rays, from the outside the solar system are present. These charged particles may cause potentials to develop between the spacecraft and ground, resulting in electrostatic discharges [45].

The interaction of Mars with the solar wind is a cross between a magnetospheric interaction and an atmospheric interaction [45].

6.5 Aircraft Operational Environments

6.5.1 Electrical. The electrical power systems for military aircraft are primarily 28V with frequencies from DC to 400 Hz [53].

6.5.2 Temperature. The properties of aircraft wiring have been analyzed in testing with temperatures in the range of -65°C to 230°C [53].

6.5.3 Atmospheric. Military aircraft operate in the earth's atmosphere, where dry air has an oxygen content of 20.95 % by volume on average [55].

6.5.4 Vibration. Acoustical vibration can result in aircraft because of jet wake and combustion turbulence. This vibration can range up to 500 Hz and a maximum amplitude of approximately 25 μm [16].

6.5.5 Particle Impacts. Not Applicable.

6.5.6 Gas/Fluid. A wide range of aerospace fluids need to be considered, including lubricating oil, hydraulic fluid, dielectric coolant fluid, isopropyl alcohol, gasoline, and others [53].

6.5.7 Pressure. The aircraft pressures range from the sea level atmospheric pressure of 101 kPa (760 Torr) to 6.5 kPa (49 Torr), which simulates an environment of 18,000 m (60,000 feet) [53].

6.5.8 Electromagnetic Radiation. See discussion for Trans-atmospheric Vehicle Section 6.3.8.

6.5.9 Particulate Radiation. Not Applicable.

6.5.10 Atomic Oxygen. Not Applicable.

6.5.11 Gravity. Not Applicable.

6.5.12 Charged Plasma. Not Applicable.

SECTION 7. NASA TESTING REQUIREMENTS FOR SPACECRAFT WIRING

The operational environments, combined with specific operational and design factors for the various spacecraft, which can lead to additional degradation or provide built-in protection from the space environments, determine the NASA testing requirements. Then by considering the testing which already has been performed by NASA, the DOD and other agencies, the additional testing necessary to address the NASA operational environments are given in this section.

7.1 <u>NASA Spacecraft Design/Operational Factors</u>

Along with the operational environments presented in Section 6, spacecraft are designed to reduce or eliminate the exposure to certain environmental conditions. In most cases, this is because, depending on mission length, none of the insulations used to date could survive full exposure to the space environment without protection of some kind. Additionally, the mission of some NASA spacecraft lead to degradation due to operations which are not addressed by strictly considering the environments, an example of this is the space shuttle maintenance requirements. In this section, the spacecraft design and operational factors which reduce the influence of space environments or give rise to additional degradation possibilities are discussed.

7.1.1 <u>**Pressurized Modules**</u>. One area where the design of the pressurized modules may eliminate the need for testing is in exposure to space fluids. The pressurized modules, as shown by current designs for both the space shuttle and space station, will have a much more controlled environment, fluid systems are designed to ensure that their fluids do not contact electrical power wiring, and exposure inside the modules to launch vehicle fluids are unlikely [23]. Additionally, the activity of the crew members, as well as any maintenance procedures, can lead to unacceptable mechanical stresses on wiring insulations, accelerating the rate of degradation [23,33,40,49,50].

7.1.2 <u>**LEO/GEO**</u>. When considering the meteoroid and debris, atomic oxygen, and radiation environment, system design must take into account the probability and criticality of a failure, and then shield the insulation, or provide redundancy, to bring the hazard down to an acceptable level. Many current and future satellites are expected to require operations beyond 10 years, all polymeric materials will degrade rapidly in such environments, and they must be protected if significant lifetime is required [23]. For LEO/GEO satellites, because they are generally autonomous and unserviced, improper handling of insulation is only a concern before launch during assembly, test, and launch procedures [23,33,40,49,50]. According to existing NASA recommendations, electrical cables and wiring must be enclosed in a "faraday cage" for protection from the space plasma [18].

7.1.3 <u>**Trans-atmospheric Vehicles**</u>. The operational environment for the trans-atmospheric vehicles are in general the same as the LEO/GEO spacecraft, and ELV's have a relatively short mission life. However, the space shuttle orbiters can sustain additional mechanical damage because of their reusabilty and resultant continuous maintenance procedures. These lead to high levels of personnel traffic in areas of limited working space, which in the past have resulted in insulation damage due to extreme mechanical stresses [23,33,40,49,50]. This has resulted in redesigns to wiring harnesses, rerouting, and additional physical protection for the wiring systems [9]. Any systems with similar maintenance requirements should consider the problems associated with the space shuttle wiring systems.

7.1.4 <u>**Lunar and Martian Surface Missions**</u>. Again, as in the LEO/GEO case, assuming a significant lifetime is required, the system must be designed to alleviate the damage as a result of meteoroids (Moon only), and radiation. The activity of the crew members, as well as any maintenance procedures can lead to unacceptable mechanical stresses on wiring insulations, accelerating the rate of degradation [23,33,40,49,50].

7.2 Comparison of NASA Requirements to Existing Testing Data

Many of the conditions outlined in Section 6 have been addressed previously in DOD and NASA testing programs. In most cases, repetition of testing is not required or desired. This section will discuss the areas where additional testing is not needed due to the existing database of testing information.

7.2.1 Existing Testing for Aircraft Environment The testing which has been performed for the military aircraft in the DOD programs, as detailed in Section 9, also addresses many of the NASA requirements for the testing of wiring insulations. These programs are sufficient to address the NASA requirements in the following areas:

General Properties: This includes analysis of the wire thickness, workmanship, diameter, and weight when compared to the wire specifications.

Electrical Properties: Including the requirements for corona inception and extinction, impulse testing, insulation resistance, and dielectric strength under aircraft environmental conditions (i.e. non-vacuum).

Mechanical Properties: Many mechanical tests have been performed to verify the insulations resistance to abrasion, cut through, repeated flexing, notching, cold flow, wire to wire abrasion, and crushing. The wires stiffness was also taken into account.

Environmental Properties: The resistance to hydrolysis as a result of exposure to humidity (KSC salt fog environment), water, or alkaline cleaners meet the requirements of a NASA system, since in general, the most severe humidity environment will be while on earth either during storage or transport [13,23,53].

Certain space environments are also addressed by standard military wire ratings. For example, wiring is rated for 100,000 hours at the maximum operating temperature, based on the rate that insulations age when exposed to high temperatures. In general, space wiring systems will not be exposed to temperatures higher than the wire rating [23]. However, additional testing may be required to address the effects of thermal cycling.

7.2.2 Existing Testing for NASA Environments. Other ongoing programs, such as the Space Shuttle and Space Station programs, have performed testing which addresses many of the unique NASA operational environments. These include enriched oxygen (30%) and vacuum environmental testing. The testing relevant to this program is summarized in Section 9. This testing will be leveraged whenever possible, and repeated tests will not be performed. In many cases, since the insulations being considered are hybrid constructions, testing may have been previously performed on wire constructions using the same materials present in these constructions. Therefore, the general behavior may be known, however, certain tests may be necessary to determine the comparative capabilities of each specific wire type.

7.3 Additional NASA Testing Requirements.

By considering the NASA environments, the spacecraft design and operational factors, and the existing database of testing, the additional testing required for NASA are determined. The tests to be performed, the NASA missions which they address, the test conditions/parameters, test method, and rationale for testing are given in Table 4. Table 5 outlines the matrix of testing which has been identified, indicating where the NASA operational environments have been satisfied by previous testing, and where additional testing is necessary.

Table 4. Additional NASA Testing Requirements

Test Property	Environment	NASA Missions Addressed	Test Method	Test Conditions	Rationale for Testing/Discussion
Arc Tracking	Air	PM, LG, TA, L, M	NHB 8060.1C #18 MIL-STD-2223	28 - 270 VDC Ambient Conditions 10^{-2} g 5 x 10^{-5} Torr 30% O_2	Testing at NASA voltage levels. Earth surface operation, assembly, and test. NASA microgravity environment. NASA vacuum environment. NASA high oxygen environment.
	Vacuum	LG, TA, L, M			
	Enriched Oxygen	PM			
	Reduced Gravity	PM, LG, TA, L, M			
Insulation Resistance	Multistress	PM, LG, TA, L, M	ASTM D-3032	400 Hz, 200°C, 500 Hrs	Analysis of new insulation constructions.
Dielectric Strength	Multistress	PM, LG, TA, L, M	ASTM D-149	400 Hz & DC @ 23°C, 100°C, & 200°C	Analysis of new insulation constructions.
Corona Inception/Extinction	Aircraft	PM	SAE AS-4373 Method 502	400 Hz & DC, 5 x 10^{-5}, 49, & 758 Torr	Complete analysis of new wire construction for aircraft applications. Analyze all samples in vacuum.
	Vacuum	LG, TA, L, M			
Wire Fusing Time	Aircraft	PM, LG, TA, L, M	SAE AS-4373 Method 511	2.5 times rated current	Analysis of new insulation constructions.
Time/Current to Smoke	Aircraft	PM, LG, TA, L, M	SAE AS-4373 Method 507	10 A + 5 A/30 s intervals	Analysis of new insulation constructions.
Abrasion Resistance	Aircraft	PM, LG, TA, L, M	SAE AS-4373 Method 701	1 - 3 lbs, 60 Hz @ 25 & 150°C	Analysis of new insulation constructions.
Dynamic Cut Through	Aircraft	PM, LG, TA, L, M	SAE AS-4373 Method 703	23°C, 70°C, 150°C, 200°C	Analysis of new insulation constructions.
Flex Life	Aircraft	PM, LG, TA, L, M	SAE AS-4373 Method 704	180° bend @ 30 cycles/min.	Analysis of new insulation constructions.
Thermal Cycling	Temperature Range, Extreme Temperatures	LG, TA, L, M	To Be Determined	To Be Determined	Number of thermal cycles, and extreme cold warrants tests.
Flammability	Enriched Oxygen	PM	NHB 8060.1C #1 & #4	30% O_2, (200°C #4 only)	Comparison of hybrid samples and new construction.
Offgassing/Odoring	Crewed Cabin	PM	NHB 8060.1C #6 & #7	25.9% O_2, 50°C	Comparison of hybrid samples and new construction.
Fluid Compatibility	Space Fluids	LG, TA, L, M	NHB 8060.1C #15	N_2O_4, N_2H_4, $N_2H_3CH_3$	Comparison of hybrid samples and new construction.
Gas Compatibility	Martian Atmosphere	M	To Be Determined	95.3% CO_2	Mission specific, far term application, low priority.
Outgassing (VCM)	Vacuum	LG, TA, L, M	ASTM E-595	5 x 10^{-5} Torr, 125°C, 24 Hrs	Comparison of hybrid samples and new construction.
Electromagnetic Radiation	Limited Atmosphere	LG, TA, L, M	MSFC Method	10,000 ESH, 110 - 200 nm, 120°C	Analysis of earth orbit environment.
Particulate Radiation	Limited Atmosphere	LG, TA, L	To Be Determined	To Be Determined	Mission specific, low priority.
Atomic Oxygen	Space Environment	LG, TA	MSFC Method	10^{23} atoms/cm_2, 0.01 eV	Analysis of earth orbit environment.
Atomic Oxygen/EM Radiation	Synergistic Environment	PM, LG, TA, L, M	MSFC Method	Above VUV and AO levels	Analysis of earth orbit environment.
Corona Discharge	Charged Plasma	LG, TA, M	To Be Determined	To Be Determined	Analysis of earth orbit environment.
Debris Impact	Meteoroids/Plasma	LG, TA, L	To Be Determined	To Be Determined	Analysis of earth orbit environment.
Electrostatic Dust	Lunar Regolith	L	To Be Determined	To Be Determined	Mission specific, far term application, low priority.

Key: PM = Pressurized Modules, LG = LEO/GEO, TA= Trans-atmospheric Vehicles, L = Lunar Surface, M = Martian Surface

Table 5. Tests Performed vs. Mission Environments Addressed Matrix.

Test \ Mission/Environment	Aircraft	Pressurized Modules	LEO/GEO	Trans-atmospheric	Lunar Surface	Martian Surface
Examine Product	DOD	DOD	DOD	DOD	DOD	DOD
Workmanship	DOD	DOD	DOD	DOD	DOD	DOD
Wire Wall Thickness	DOD	DOD	DOD	DOD	DOD	DOD
Conductor Diameter	DOD	DOD	DOD	DOD	DOD	DOD
Finished Diameter	DOD	DOD	DOD	DOD	DOD	DOD
Finished Weight	DOD	DOD	DOD	DOD	DOD	DOD
Wire Surface Markability	DOD	DOD	DOD	DOD	DOD	DOD
Impulse Dielectric	DOD	DOD	DOD	DOD	DOD	DOD
Insulation Resistance	DOD	DOD	DOD	DOD	DOD	DOD
Spark Test	DOD	DOD	DOD	DOD	DOD	DOD
Dry Dielectric Test	DOD	DOD	DOD	DOD	DOD	DOD
Voltage Withstand	DOD	DOD	DOD	DOD	DOD	DOD
Dielectric Constant	DOD	DOD	DOD	DOD	DOD	DOD
CIV/CEV (AC & DC)	DOD	Some DOD	Some DOD	Some DOD	Some DOD	Some DOD
Surface Resistance	DOD	DOD	DOD	DOD	DOD	DOD
Time/Current to Smoke	DOD	Some DOD	Some DOD	Some DOD	Some DOD	Some DOD
Wire Fusing Time	DOD	DOD	DOD	DOD	DOD	DOD
Dry Arc Resistance	DOD	Not required	Not required	Not required	Not required	Not required
BSI Dry Arc Resistance	DOD	Not required	Not required	Not required	Not required	Not required
Arc Tracking - SSF	Not required	NASA	NASA	NASA	NASA	NASA
Arc Tracking - NHB Method	Not required	Additional	Additional	Additional	Additional	Additional
Arc Tracking - MIL-W-2223	Additional	Additional	Additional	Additional	Additional	Additional
Dielectric Strength	DOD	Additional	Additional	Additional	Additional	Additional
Abrasion	DOD	Some DOD	Some DOD	Some DOD	Some DOD	Some DOD
Dynamic Cut Through	DOD	Some DOD	Some DOD	Some DOD	Some DOD	Some DOD
Flex Life	DOD	Some DOD	Some DOD	Some DOD	Some DOD	Some DOD
Notch Propagation	DOD	DOD	DOD	DOD	DOD	DOD
Stiffness and Springback	DOD	DOD	DOD	DOD	DOD	DOD
Crush Resistance	DOD	DOD	DOD	DOD	DOD	DOD
Insulation Impact Resistance	DOD	DOD	DOD	DOD	DOD	DOD
Tensile Strength	DOD	DOD	DOD	DOD	DOD	DOD
Wire to Wire Rub	DOD	DOD	DOD	DOD	DOD	DOD
Aging Stability	DOD	DOD	DOD	DOD	DOD	DOD
Thermal Index	DOD	DOD	DOD	DOD	DOD	DOD
Thermal Shock	DOD	DOD	DOD	DOD	DOD	DOD
Thermal Aging	DOD	DOD	DOD	DOD	DOD	DOD
Cold Bend	DOD	DOD	DOD	DOD	DOD	DOD
Thermal Cycling	Not required	Not required	Additional	Additional	Additional	Additional
Flammability - Aircraft	DOD	DOD	DOD	DOD	DOD	DOD
Toxicity - Burning	DOD	DOD	DOD	DOD	DOD	DOD
Smoke Quantity	DOD	DOD	DOD	DOD	DOD	DOD

Test \ Mission/Environment	Aircraft	Pressurized Modules	LEO/GEO	Trans-atmospheric	Lunar Surface	Martian Surface
Flame Propagation	Not required	Some NASA	Not required	Not required	Not required	Not required
Flash Point of Liquids	Not required	NASA	Not required	Not required	Not required	Not required
Wire Flammability	Not required	Some NASA	Not required	Not required	Not required	Not required
Odor Assessment	Not required	Some NASA	Not required	Not required	Not required	Not required
Offgassing	Not required	Some NASA	Not required	Not required	Not required	Not required
Arc Tracking - 30% O_2	Not required	Additional	Not required	Not required	Not required	Not required
Wet Arc Tracking (ASTM)	DOD	DOD	DOD	DOD	DOD	DOD
Hydrolysis (ASTM)	DOD	DOD	DOD	DOD	DOD	DOD
Wet Arc Tracking (SAE)	DOD	DOD	DOD	DOD	DOD	DOD
Fluid Immersion	DOD	DOD	DOD	DOD	DOD	DOD
Forced Hydrolysis (SAE)	DOD	DOD	DOD	DOD	DOD	DOD
Humidity Resistance	DOD	DOD	DOD	DOD	DOD	DOD
Wicking	DOD	DOD	DOD	DOD	DOD	DOD
Impact of LOX and GOX	Not required	Not required	Not required	NASA	Not required	Not required
Fluid Compatability	Not required	Not required	Additional	Additional	Additional	Additional
Gas Compatability	Not required	Not required	Not required	Not required	Not required	Additional
Weight Loss	DOD	Not required	Not required	Not required	Not required	Not required
CIV/CEV - Vacuum	Not required	Not required	Additional	Additional	Additional	Additional
Outgassing (VCM)	Not required	Not required	Some NASA	Some NASA	Some NASA	Some NASA
Arc Tracking - Vacuum	Not required	Not required	Additional	Additional	Additional	Additional
Weathering Resistance	DOD	Not required	Not required	Not required	Not required	Not required
VUV Exposure	Not required	Not required	Additional	Additional	Additional	Additional
VUV/AO Exposure	Not required	Not required	Additional	Additional	Not required	Additional
Radiation Exposure	Not required	Not required	Additional	Additional	Additional	Additional
AO Exposure	Not required	Not required	Additional	Additional	Not required	Additional
Arc Tracking - µg	Not required	Additional	Additional	Additional	Additional	Additional
Flame Spread Rate - µg	Not required	Additional	Additional	Additional	Additional	Additional
Corona Discharge - Plasma	Not required	Not required	Additional	Additional	Not required	Additional
Debris Impacts	Not required	Not required	Additional	Additional	Not required	Not required
Electrostatic Dust	Not required	Not required	Not required	Not required	Additional	Not required

Key:

- Tests performed by DOD programs [11 - 13]
- Some DOD testing, more necessary [11 - 13]
- Tests performed by NASA programs [26, 56, 57]
- Some NASA testing, more necessary [26, 56, 57]
- Tests not required for this program
- Additional tests to be performed

Section 8. CONCLUSIONS

The results of Task #1 of the NASA Wiring for Space Applications Program were shown in this report. This task presents the operational environments for the electrical power wiring of NASA space vehicles, and outlines the additional testing required, beyond the testing performed previously by the DOD and NASA, for use of the insulation constructions in NASA spacecraft. The required tests will be performed as a part of Task #2 of this program. Also, as shown in this report, there are operational considerations such as maintenance procedures which can contribute to degradation of wiring systems. These wiring system issues will be addressed in Task #3 of the program. The results of the testing and analysis phase (Task #2) and the power wiring system analysis phase (Task #3) will be reported as they become available.

Through this program, in conjunction with the efforts of other US governmental laboratories, industry, and academia, a better understanding of arc tracking in wiring insulations will be achieved. In addition, the top performing "hybrid" insulation constructions, such as those being addressed in this program are expected to go through military qualification in the near future. It is anticipated that the efforts of the NASA wiring program will be closely coordinated with the military efforts, such that the NASA and military standards will be in agreement. The resulting database of information will help in the development of lightweight, safe, and reliable wiring systems applying new wiring constructions which are resistant to arc tracking, and suitable for use in aerospace applications.

SECTION 9. APPENDIX A: SUMMARY OF EXISTING TEST DATA

This program is leveraging the large quantities of valuable testing data gathered previously by DOD and NASA programs. The related program reports and databases which were identified and used to develop a database of existing testing information are given in references 11 - 13, 26, 53, and 56 - 59. This section will give a description of the wiring insulation types which are used in these testing programs, and a summary of the testing results from these programs to date.

9.1 Candidate Wiring Constructions

Table 6 describes the wire types which have been included in the existing testing for aerospace and space applications.

Table 6. Description of Insulation Constructions [53]

	Insulation Construction	Sample Description	Comments
1	MIL-W-81381/7	6 mil wall polyimide insulation, silver coated copper.	Control wire, restricted use by military and NASA due to arc tracking.[12,15].
2	MIL-W-81381/11	8.6 mil wall polyimide insulation, silver coated copper.	Control wire, restricted use by military and NASA due to arc tracking.[12,15].
3	MIL-W-22759/12	PTFE or TFE insulated wire, nickel coated copper.	High temperature insulation currently in use on spacecraft.
4	MIL-W-22759/16	ETFE insulated wire, tin coated copper, medium weight.	Poor flammability performance in enriched oxygen environment [57].
5	MIL-W-22759/18	ETFE insulated wire, tin coated copper, light weight.	Poor flammability performance in enriched oxygen environment [57].
6	MIL-W-22759/32	XL-ETFE insulated wire, tin coated copper.	Poor flammability performance in enriched oxygen environment [57].
7	MIL-W-22759/34	XL-ETFE insulated wire, tin coated copper, polyamide braid cover.	Poor flammability performance in enriched oxygen environment [57].
8	MIL-W-22759/43	XL-ETFE insulated wire, silver coated copper, normal weight.	Control wire for aircraft testing performed by Air Force [53].
9	MIL-W-16878	TFE insulated wire, silver coated copper, Type EE.	Not in NASA approved parts list (MIL-STD-975).
10	SSQ-21652	Silicone insulated wire, nickel coated copper.	Insulation tested and approved for Space Station Program [26].
11	SSQ-21656	TFE insulated wire, nickel coated copper (also NGTW-TFE-xx).	Insulation tested and approved for Space Station Program [26].
12	MP571-0086	Polyimide insulated wire, nickel coated copper.	Polyimide control wire.
13	Filotex - TKT	PTFE Extrusion/616 Kapton (50% Min OL)/ PTFE Dispersion	One of top "hybrid" insulations from the Air Force program [53].
14	Tensolite - TKT	200AJ919 (50% Min OL)/PTFE Tape (50% Min OL)	One of top "hybrid" insulations from the Air Force program [53].
15	Thermatics - TKT	Modified PTFE Tape (50% OL)/TPT Tape (50% OL)/ Mod PTFE Tape (50% OL)/PTFE Dispersion	One of top "hybrid" insulations from the Air Force program [53].
16	Gore HS-725	PTFE (50% OL)/HSCR PTFE (50% OL)	Considered promising in the Air Force program but dropped due to single source prohibition, characterization to be continued in NASA program [53].
17	Nema #3 - TKT	616 Kapton (45-50% OL)/Extruded XL ETFE	One of top "hybrid" insulations from the Air Force program , but not available for additional testing [53].
18	Barcel - TKT	2919 Kapton (50% OL)/Unsintered PTFE Tape, Buttwrap	Eliminated from further testing after the Air Force program [53].
19	Nema #2 - TKT	PTFE Tape/616 Kapton (50% OL)/PTFE Tape	Eliminated from further testing after the Air Force program [53].
20	DuPont (P-FP)	New P-FP Tape (50% OL)/New P-FP Tape (50% OL)/FP	Eliminated from further testing after the Air Force program [53].
21	Brand Rex - TKT	XL ETFE Tape (50% OL)/Kapton (50% OL) /XL ETFE Tape (50% OL)	Eliminated from further testing after the Air Force program [53].
22	Champlain - TKT	2919 Kapton (50% OL)/Extruded XL ETFE	Eliminated from further testing after the Air Force program [53].
23	TRW - PFPI	TRW - PFPI	New insulation material to be tested further in the NASA program.

Abbreviations:
2919 Kapton = 0.5 mil Fluorocarbon (PTFE), 1 mil Polyimide, 0.5 mil Fluorocarbon (PTFE)
616 Kapton = 0.1 mil Fluorocarbon (FEP), 1 mil Polyimide, 0.1 mil Fluorocarbon (FEP)
200AJ919 = 0.5 mil Fluorocarbon (PTFE), 1 mil Polyimide, 0.5 mil Fluorocarbon (PTFE)

9.2 Department of Defense Wire Testing Programs

This section gives a summary of the data which has been gathered for wire insulation testing by DOD programs which are also applicable to the objectives of the NASA Electrical Power Wiring Program. These include the programs of the Air Force Wright Laboratory and the Naval Research Laboratory.

The Air Force WL program evaluated new insulation constructions for aerospace wiring applications. The results from this work are summarized in Tables 7 - 10, where the sample numbers are as defined in

Table 7. WL Aerospace Wiring Test Results - General Properties [53].

Test Title	Test Method	Test Conditions/ Parameters	Samples	Conclusion/Comments
Finished Weight	SAE AS-4372	1000 hrs, 200°C thermal aging	1, 2, 8, 13-15, 17	Filotex, Thermatics, Tensolite and MIL-W-22759 performed best.
Finished Diameter	SAE AS-4372	Unconditioned wire	1, 2, 8, 13-15, 17	Filotex and Thermatics performed best.
Conductor Diameter	ASTM D-3032	Unconditioned wire	1,2,8,13-22	All samples comparable.
Wire Wall Thickness	SAE AS-4373 Method 401	Unconditioned wire	1,2,8,13-22	All samples comparable.
Workmanship	SAE AS-4373 Method 901	Unconditioned wire	1, 2, 8, 13-15, 17	All samples comparable.
Examine Product	SAE AS-4373 Method 902	Unconditioned wire	1, 2, 8, 13-15, 17	Filotex, MIL-W-22759 and MIL-W-81381 weighed least, Tensolite was heaviest.
Wire Surface Markability	SAE AS-4373 Method 713	Unconditioned wire	1,2,8,13-22	M22759 best, M81381 worst.

Table 8. WL Aerospace Wiring Test Results - Mechanical Properties [53].

Test Title	Test Method	Test Conditions/ Parameters	Samples	Conclusion/Comments
Abrasion	SAE AS-4373 Method 701	1 - 3 lbs, 60 Hz @ 25 & 150°C Unaged and at 1000 hrs/200°C	1,2,8,13-22	Filotex, Tensolite, Thermatics good, Gore sample did not perform well.
Dynamic Cut Through	SAE AS-4373 Method 703	23°C, 70°C, 150°C, 200°C Unaged and at 1000 hrs/200°C	1,2,8,13-22	Kapton performed best, Tensolite and Thermatics were next best performers.
Flex Life	SAE AS-4373 Method 704	180° bend @ 30 cycles/min Unaged and at 1000 hrs/200°C	1,2,8,13-22	Kapton & Gore best samples, all other samples similar in performance.
Notch Propagation	SAE AS-4373 Method 707	50 & 67% notch depth Unaged and at 1000 hrs/200°C	1,2,8,13-22	Filotex, Tensolite, Thermatics, and MIL-W-81381 performed equally well.
Stiffness and Springback	SAE AS-4373 Method 708	Unconditioned samples	1,2,8,13-22	MIL-W-81381 is stiffest, all other samples comparable in their performance.
Crush Resistance	ASTM D-3032 Section 20	Unconditioned samples	1, 2, 8, 13-15, 17	MIL-W-81381 is best, all other samples comparable in their performance.
Insulation Impact Resistance	ASTM D-256 Method A	0.0625 diameter impact edge	1, 2, 8, 13-15, 17	MIL-W-81381 is best, all other samples comparable in their performance.
Tensile Strength	SAE AS-4373 Method 706	Unconditioned samples	1, 2, 8, 13-15, 17	MIL-W-81381, Thermatics and Tensolite are best.
Wire to Wire Rub	MDA Test Method	48 cycles/min, 870 hrs	1, 2, 8, 13-15, 17	All samples comparable.

Table 6, and the final statistical wire rankings are given in Table 11. The top three "hybrid" candidate constructions when compared to MIL-W-81381 had increased flexibility, had good performance in wet and dry arc tracking tests, and had increased temperature capabilities, but they had slightly lower mechanical properties [53]. When compared to the MIL-W-22759 (XL-ETFE), they had superior mechanical properties in temperatures greater than 70°C and were superior in the flammability and smoke generation tests, but had less flexibilty [53].

Testing performed by the Naval Research Laboratory on polyimide degradation due to exposure to humidity environments and also for susceptibility to arc tracking are reported in references 11 - 13 and are outlined in Table 12. Again, the samples are as described in Table 6.

Table 9. WL Aerospace Wiring Test Results - Electrical Properties [53].

Test Title	Test Method	Test Conditions/ Parameters	Samples	Conclusion/Comments
Impulse Dielectric	SAE AS-4373 Method 503	8 kV, unconditioned wire	1, 2, 8, 13-15, 17	Filotex, Tensolite, Gore, MIL-W-81381 and MIL-W-22759 are all comparable.
Insulation Resistance	SAE AS-4373 Method 504	Thermal aging (1000 hrs/200°C)	1,2,8,13-22	All samples comparable.
Spark Test	SAE AS-4373 Method 503	1.5 kV, unconditioned wire	1, 2, 8, 13-15, 17	All samples comparable.
Dry Dielectric Test	SAE AS-4373 Method 503	2.5 kV, unconditioned wire	1, 2, 8, 13-15, 17	All samples comparable.
Voltage Withstand	SAE AS-4373 Method 510	Thermal aging (1000 hrs/200°C) 2.5 kV, 60 Hz @ 500 V/s	1,2,8,13-22	All samples comparable.
Dielectric Constant	SAE AS-4373 Method 501	Unconditioned wire	1, 2, 8, 13-15, 17	Tensolite, MIL-W-22759, and Filotex best samples.
CIV/CEV (AC & DC)	SAE AS-4373 Method 502	400 Hz & DC, 49 & 758 Torr	1, 2, 8, 13-15, 17	Tensolite best, Thermatics worst.
Surface Resistance	SAE AS-4373 Method 506	96 hrs/95% RH/25°C	1, 2, 8, 13-15, 17	All samples comparable.
Time/Current to Smoke	SAE AS-4373 Method 507	10 A + 5 A/30 s	1, 2, 8, 13-15, 17	All samples comparable.
Wire Fusing Time	SAE AS-4373 Method 511	2.5 times free air rated current (AWG #22 - I_{test} = 45 A)	1, 2, 8, 13-15, 17	Tensolite and MIL-W-81381 performed best.
Dry Arc Resistance	SAE AS-4373 Method 301	28 VDC	1,2,8,13-22	Bundled wires, Kapton showed propagation, most other samples didn't, most failed voltage test.
BSI Dry Arc Resistance	BSI 90/76828 BSI 90/80606	28 VDC, 115 V/400 Hz	1, 2, 8, 13-15, 17	Bundled wires, Thermatics, Tensolite, and Filotex best wire sample types.

Table 10. WL Aerospace Wiring Test Results - Environmental Properties [53].

Test Title	Test Method	Test Conditions/ Parameters	Samples	Conclusion/Comments
Aging Stability	SAE AS-4373 Method 510	230°C for 96 hrs	1, 2, 8, 13-15, 17	All samples comparable.
Thermal Index	SAE AS-4373 Method 804	220 - 280°C aging to bend failure	1, 2, 8, 13-15, 17	Filotex, Tensolite, Thermatics and MIL-W-81381 are comparable.
Thermal Shock	SAE AS-4373 Method 805	-55°C to 200°C, 4/30 min cycles	1, 2, 8, 13-15, 17	Filotex, Tensolite, MIL-W-22759 and MIL-W-81381 are comparable.
Thermal Aging	SAE AS-4373 Method 807	1000 hrs at 200°C	1, 2, 8, 13-15, 17	Thermatics, Filotex, Tensolite and MIL-W-22759 were similar.
Cold Bend	SAE AS-4373 Method 702	-65°C for 4 hrs, 2 rpm wraping	1, 2, 8, 13-15, 17	Filotex and Tensolite were best samples.
Flammability	SAE AS-4373 Method 801	Ambient air	1,2,8,13-22	MIL-W-81381, Tensolite and Filotex are comparable.
Toxicity - Burning	Naval Engineering Standard 713	1 to 2 gram burn mass	1,2,8,13-22	Filotex, Thermatics, and MIL-W-22759 are comparable.
Smoke Quantity	SAE AS-4373 Method 803	Radiant heat and flame exposure	1, 2, 8, 13-15, 17	All other samples performed similarly better than MIL-W-22759.
Wet Arc Tracking	SAE AS-4373 Method 509	Unconditioned wire	1, 2, 8, 13-15, 17	Filotex, Tensolite and Thermatics all comparable.
Fluid Immersion	SAE AS-4373 Method 603	Aerospace fluids	1,2,8,13-22	All samples are comparable.
Forced Hydrolysis	SAE AS-4373 Method 602	5% salt water, 70°C, 720 hrs; Unaged and at 1000 hrs/200°C	1, 2, 8, 13-15, 17	Tensolite and MIL-W-22759 were the best performers.
Humidity Resistance	SAE AS-4373 Method 603	70°C, 95% RH, 360 hrs	1, 2, 8, 13-15, 17	MIL-W-22759 was best, Filotex and Tensolite next best.
Wicking	SAE AS-4373 Method 607	Dye solution	1, 2, 8, 13-15, 17	All samples comparable.
Weight Loss	SAE AS-4373 Method 604	36 Torr, 200°C, 384 hrs	1, 2, 8, 13-15, 17	Tensolite performed best.
Weathering Resistance	SAE AS-4373 Method 606	120/8 hr UV cycles, 40 - 70°C	1, 2, 8, 13-15, 17	All insulations comparable.

Table 11. WL Aerospace Wiring Test Program Final Statistical Test Results [53].

Wire Rank	Screen Test Unweighted	Screen Test Aircraft Weighted	Full Performance Unweighted	Full Performance Aircraft Weighted
1	Filotex	Filotex	Tensolite	Filotex
2	Thermatics	Thermatics	Filotex	Tensolite
3	Gore*	NEMA #3	MIL-W-81381	MIL-W-81381
4	NEMA #3	Gore*	Thermatics	Thermatics
5	MIL-W-81381	MIL-W-81381	NEMA #3	NEMA #3
6	Tensolite	Tensolite	MIL-W-22759	MIL-W-22759
7	Barcel	Champlain	---	---
8	Champlain	Barcel	---	---
9	MIL-W-22759	NEMA #2	---	---
10	NEMA #2	MIL-W-22759	---	---
11	DuPont	Brand Rex	---	---
12	Brand Rex	DuPont	---	---

* The Gore construction was eliminated after screening tests due to Air Force sole source restrictions.

Table 12. NRL Aerospace Wiring Test Results [11-13].

Test Title	Test Method	Test Conditions/ Parameters	Samples	Conclusion/Comments
Wet Arc Tracking	ASTM D-2303	1% salt water solution	2,4,7	The polyimide insulation exhibited arc tracking and flashover [12].
Hydrolysis	ASTM D-3032	60°C to 90°C, 70% - 100% RH	2	Humidity accelerates polyimide degradation [11,13].

9.3 NASA Wire Testing Programs

Testing has been performed in other NASA programs which are applicable to this program. Two specific programs, one which addresses arc tracking for the Space Station Freedom program and another which considers the flammability of ETFE and XL-ETFE constructions in enriched oxygen environments, are discussed in references 26 and 57 and summarized in Table 13, with the samples as given in Table 6.

Table 13. NASA Wiring Test Results [26,57].

Test Title	Test Method	Test Conditions/ Parameters	Samples	Conclusion/Comments
Arc Tracking	JSC Test Method TPS# 8PL9120004	28 - 120 VDC, 117 VAC	4,7,9 - 12, 15,16	The SSQ-21656, SSQ-21652 and MIL-W-22759/34 did not arc track [26].
Electrical Wire Flammability	NHB 8060.1C Test #4	> 25% O_2 concentration	4 - 7	Use of these insulations are not a safe choice for these enriched oxygen envirronments [57].

The NASA MSFC Materials and Processes Laboratory Technical Information System - MAPTIS [56] contains the data from testing performed according to the NASA test specifications for "Flammability, Odor, Offgassing, and Compatibility Requirements and Test Procedures for Materials in Environments that Support Combustion," NHB 8060.1C. This database contains information on many tests, those which are directly related to this program are summarized in Table 14. Again the samples are as described in Table 6, with the exception of sample A, which is an unspecified MIL-W-81381 construction (i.e. no /xx definition), and B is an unspecified MIL-W-22759 construction.

9.4 Summary of Tests Performed vs. Wiring Constructions

Table 15 presents for the wiring constructions described in Table 6, a summary of the testing status for the numerous tests identified as important for NASA wiring systems. The table outlines the tests which already have been performed, those to be performed in this program. those not necessary, and those which are a low priority for tests which are not scheduled to be tested at this time, but may have research value.

Table 14. NASA MAPTIS Test Data [56].

Test Title	Test Method	Test Conditions/ Parameters	Samples	Conclusion/Comments
Upward Flame Propagation	NHB8060.1C Test #1	20.9% O_2, 14.7 psi 25.9% O_2, 14.3 psi 30.3% O_2, 9.3 psi 30.3% O_2, 20.4 psi 100% O_2, 14.7 psi	5,6 2 2 A,B A,B	Various sample types have passed the 30% O_2 environment. Others have not passed or not tested.
Flash Point of Liquids	NHB8060.1C Test #3	24% O_2, 14.5 psi 25.9% O_2, 14.3 psi 30% O_2, 10 psi	A A,B 5	This test for information only, no future tests planned in this area.
Electrical Wire Flammability	NHB8060.1C Test #4A	20.9% O_2, 14.7 psi 25.9% O_2, 14.3 psi 30% O_2, 10.2 psi	5 4 11	Various sample types have passed the 30% O_2 environment. Others have not passed or not tested.
	NHB8060.1C Test #4B	20.9% O_2, 14.7 psi 25.9% O_2, 14.3 psi 30% O_2, 10 psi 100% O_2, 30 psi	5,6 2,3 A,B,12 A,B	
Odor Assessment	NHB8060.1C Test #6	26% O_2, 12.3 psi, 120°F, 72 hrs	A	Polyimide sample passed test.
Offgassed Products	NHB8060.1C Test #7	20.9% O_2, 14.5 psi, 120°F, 72 hrs	2,3,6	Polyimide, PTFE, and XL-ETFE samples passed test.
Mechanical Impact for Materials in LOX and GOX	NHB8060.1C Test #13	100% GOX, 50 psi, 480°F 100% GOX, 14.7 psi, 75°F 100% LOX, 14.7 psi, -297°F	2,11,16 A,B A,B	Various samples have passed test.
Outgassing in Vacuum	ASTM E-595-90	5×10^{-5} torr, 125°C, 24 hrs	2,11	Polyimide and silicone rubber samples have passed the test.

Table 15. Tests Performed vs. Wiring Constructions Matrix.

Construction \ Test	Examine Product	Workmanship	Wire Wall Thickness	Conductor Diameter	Finished Diameter	Finished Weight	Wire Surface Markability	Impulse Dielectric	Insulation Resistance	Spark Test	Dry Dielectric Test	Voltage Withstand	Dielectric Constant	CIV/CEV (AC & DC)	Surface Resistance	Time/Current to Smoke	Wire Fusing Time	Dry Arc Resistance	BSI Dry Arc Resistance	Arc Tracking - SSF	Arc Tracking - NHB Method	Arc Tracking - MIL-W-2223	Dielectric Strength	Abrasion	Dynamic Cut Through	Flex Life	Notch Propagation	Stiffness and Springback	Crush Resistance	Insulation Impact Resistance	Tensile Strength	Wire to Wire Rub	Aging Stability	Thermal Index	Thermal Shock	Thermal Aging	Cold Bend	Thermal Cycling	Flammability - Aircraft	Toxicity - Burning	Smoke Quantity
MIL-W-81381/7	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬛	⬜	⬜	⬜	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬜	🟩	🟩	🟩
MIL-W-81381/11	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬜	🟩	🟩	🟩
MIL-W-22759/12	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬜	⬜	⬜	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬜	⬛	⬛	⬛
MIL-W-22759/16	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟥	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛
MIL-W-22759/18	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛
MIL-W-22759/32	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛
MIL-W-22759/34	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟥	⬜	⬜	⬜	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬜	⬛	⬛	⬛
MIL-W-22759/43	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬛	🟩	🟩	🟩
MIL-W-16878	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟥	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛
SSQ-21652	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟥	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛
SSQ-21656	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟥	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛
MP571-0086	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟥	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛
Filotex - TKT	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬛	⬜	⬜	⬜	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬜	🟩	🟩	🟩
Tensolite - TKT	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬛	⬜	⬜	⬜	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬜	🟩	🟩	🟩
Thermatics - TKT	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟥	⬜	⬜	⬜	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬜	🟩	🟩	🟩
Gore HS-725	🟩	🟩	🟩	🟩	🟩	🟩	⬛	🟩	🟩	🟩	🟩	🟩	⬜	⬛	⬛	⬜	⬜	🟩	⬛	🟥	⬛	⬛	⬜	▦	▦	▦	🟩	🟩	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟩	🟩	⬛
Nema #3 - TKT	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	🟩	⬛	🟩	🟩	🟩
Barcel - TKT	🟩	🟩	🟩	🟩	🟩	🟩	⬛	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	⬛	🟩	⬛	⬛	⬛	⬛	⬛	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟩	🟩	⬛
Nema #2 - TKT	🟩	🟩	🟩	🟩	🟩	🟩	⬛	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	⬛	🟩	⬛	⬛	⬛	⬛	⬛	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟩	🟩	⬛
DuPont (P-FP)	🟩	🟩	🟩	🟩	🟩	🟩	⬛	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	⬛	🟩	⬛	⬛	⬛	⬛	⬛	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟩	🟩	⬛
Brand Rex - TKT	🟩	🟩	🟩	🟩	🟩	🟩	⬛	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	⬛	🟩	⬛	⬛	⬛	⬛	⬛	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟩	🟩	⬛
Champlain - TKT	🟩	🟩	🟩	🟩	🟩	🟩	⬛	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	⬛	🟩	⬛	⬛	⬛	⬛	⬛	🟩	🟩	🟩	🟩	🟩	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	🟩	🟩	⬛
TRW - PFPI	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬜	⬛	⬜	⬛	⬜	⬛	⬜	⬛	⬛	⬛	⬛	⬛	⬛	⬜	⬜	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬛	⬜	⬛	⬛	⬛

9-7

Table 15. Tests Performed vs. Wiring Constructions Matrix (Cont'd).

Construction \ Test	Flame Propagation	Flash Point of Liquids	Wire Flammability	Odor Assessment	Offgassing	Arc Tracking - 30% O_2	Wet Arc Tracking (ASTM)	Hydrolysis (ASTM)	Wet Arc Tracking (SAE)	Fluid Immersion	Forced Hydrolysis (SAE)	Humidity Resistance	Wicking	Impact of LOX and GOX	Fluid Compatability	Gas Compatability	Weight Loss	CIV/CEV - Vacuum	Outgassing (VCM)	Arc Tracking - Vacuum	Weathering Resistance	VUV Exposure	VUV/AO Exposure	Radiation Exposure	AO Exposure	Arc Tracking - µg	Flame Spread Rate - µg	Corona Discharge - Plasma	Debris Impacts	Electrostatic Dust
MIL-W-81381/7	O	B	R	O	r	O	B	B	G	G	G	G	G	B	O	O	G	O	O	O	G	O	O	O	O	O	O	O	O	O
MIL-W-81381/11	R	B	R	B	O	B	G	G	G	G	G	G	G	R	B	B	G	B	R	B	G	O	B	B	O	B	B	B	B	B
MIL-W-22759/12	B	B	R	B	R	O	B	B	B	B	B	B	B	B	B	B	B	B	B	O	B	B	B	B	O	O	O	B	B	B
MIL-W-22759/16	B	B	R	B	B	B	G	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B
MIL-W-22759/18	R	R	R	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	R	B	B	B	B	B	B	B	B	B	B	B
MIL-W-22759/32	R	B	R	B	R	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B
MIL-W-22759/34	B	B	R	B	B	O	G	B	B	B	B	B	B	B	B	O	B	O	B	O	B	B	B	O	O	O	O	O	O	O
MIL-W-22759/43	B	B	B	B	B	B	B	B	G	G	G	G	G	B	B	B	G	B	B	B	G	B	B	B	B	B	B	B	B	B
MIL-W-16878	R	R	R	R	R	B	B	B	B	B	B	B	B	R	B	B	B	B	R	B	B	B	B	B	B	B	B	B	B	B
SSQ-21652	B	B	B	B	B	B	B	B	B	B	B	B	B	R	B	B	B	B	R	B	B	B	B	B	B	B	B	B	B	B
SSQ-21656	B	B	R	B	B	B	B	B	B	B	B	B	B	R	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B
MP571-0086	B	B	R	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B
Filotex - TKT	O	B	O	O	O	O	B	B	G	G	G	G	G	B	O	O	G	O	O	O	G	O	O	O	O	O	O	O	O	O
Tensolite - TKT	O	B	O	O	O	O	B	B	G	G	G	G	G	B	O	O	G	O	O	O	G	O	O	O	O	O	O	O	O	O
Thermatics - TKT	O	B	O	O	O	O	B	B	G	G	G	G	G	B	O	O	G	O	O	O	G	O	O	O	O	O	O	O	O	O
Gore HS-725	O	B	O	O	O	B	B	B	B	G	B	B	B	R	O	B	B	B	O	B	B	O	O	B	B	B	B	B	B	B
Nema #3 - TKT	B	B	B	B	B	B	B	B	G	G	G	G	G	B	B	B	G	B	B	B	G	B	B	B	B	B	B	B	B	B
Barcel - TKT	B	B	B	B	B	B	B	B	B	G	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B
Nema #2 - TKT	B	B	B	B	B	B	B	B	B	G	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B
DuPont (P-FP)	B	B	B	B	B	B	B	B	B	G	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B
Brand Rex - TKT	B	B	B	B	B	B	B	B	B	G	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B
Champlain - TKT	B	B	B	B	B	B	B	B	B	G	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B	B
TRW - PFPI	O	B	O	O	O	O	B	B	B	B	B	B	B	B	O	O	B	O	O	O	B	O	O	O	O	O	O	O	O	O

Key:

- G: Tests performed by DOD programs [11 - 13]
- Some DOD testing, more necessary [11 - 13]
- R: Tests performed by NASA programs [26, 56, 57]
- r: Some NASA testing, more necessary [26, 56, 57]
- B: Tests not required for this program
- O: Additional tests to be performed

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9-8

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13. ABSTRACT ***(Maximum 200 words)***

Electrical wiring systems are used extensively on NASA space systems for power management and distribution, control and command, and data transmission. The reliability of these systems when exposed to the harsh environments of space is very critical to mission success and crew safety. Failures have been reported both on the ground and in flight due to arc tracking in the wiring harnesses, made possible by insulation degradation. This report was written as part of a NASA Office of Safety and Mission Assurance (Code Q) program to identify and characterize wiring systems in terms of their potential use in aerospace vehicles. The goal of the program is to provide the information and guidance needed to develop and qualify reliable, safe, lightweight wiring systems, which are resistant to arc tracking and suitable for use in space power applications. This report identifies the environments in which NASA spacecraft will operate, and determines the specific NASA testing requirements. A summary of related test programs is also given in this report. This data will be valuable to spacecraft designers in determining the best wiring constructions for the various NASA applications.

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