CABLES AND CONNECTORS FOR LARGE SPACE SYSTEM TECHNOLOGY (LSST)

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Final Report

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SEATTLE, WASHINGTON

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National Aeronautics
and Space Administration

George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812

NASA

CABLES AND CONNECTORS FOR LARGE SPACE SYSTEM STRUCTURES (LSST)

Final Report

by

W. G. Dunbar

BOEING AEROSPACE COMPANY

April, 1980

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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FORWARD

This document was prepared by the Boeing Aerospace Company for the National Aeronautics and Space Administration, George C. Marshall Space Flight Center in compliance with contract NAS8-33432, "Grounding/Bonding and Data Power Distribution Connectors and Cables for Large Space System Technology (LSST)."

This report is one of two volumes documenting contract results. It consists primarily of the data generated during the task on cables and connectors for large space systems with metal, metal composites and graphite epoxy members. The studies were based on airplane and spacecraft cable and connector applications now in service or planned for the near future.

ABSTRACT

This conceptual design and analytical study program examined the influence of the environment and extravehicular activity/remote assembly operations on the cables and connectors for spacecraft with metallic and/or non-metallic structures. Cable and connector philosophy was outlined for the electrical systems and electronic compartments which contain high-voltage, high-power electrical and electronic equipment. The influence of plasma and particulates on the system was analyzed and the effects of static buildup on the spacecraft electrical system discussed. Conceptual cable and connector designs were assessed for capability to withstand high current and high voltage without danger of arcs and electromagnetic interference. Also shown were the extravehicular activities required of the space station and/or supply spacecraft crew members to join and inspect the electrical system, using manual or remote assembly construction.

Mr. Wayne Shockley was the NASA contracting officer's technical representative and Mr. Charles Souther was the assistant technical representative for cables and connectors. Boeing performance was under the management of Sidney W. Silverman; William G. Dunbar was the technical leader. Major participants in this program were:

Power Systems	Kenneth J. Ferschweiler
Electromagnetic Compatibility	Thomas H. Herring
Plasma Physics	Dr. Nazih N. Youssef

KEY WORDS

Bonding
Cables
Connectors
Distribution
Grounding
Particulates
Plasma
Power/electronic Systems
Spacecraft Charging
Structures

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1.0 SUMMARY

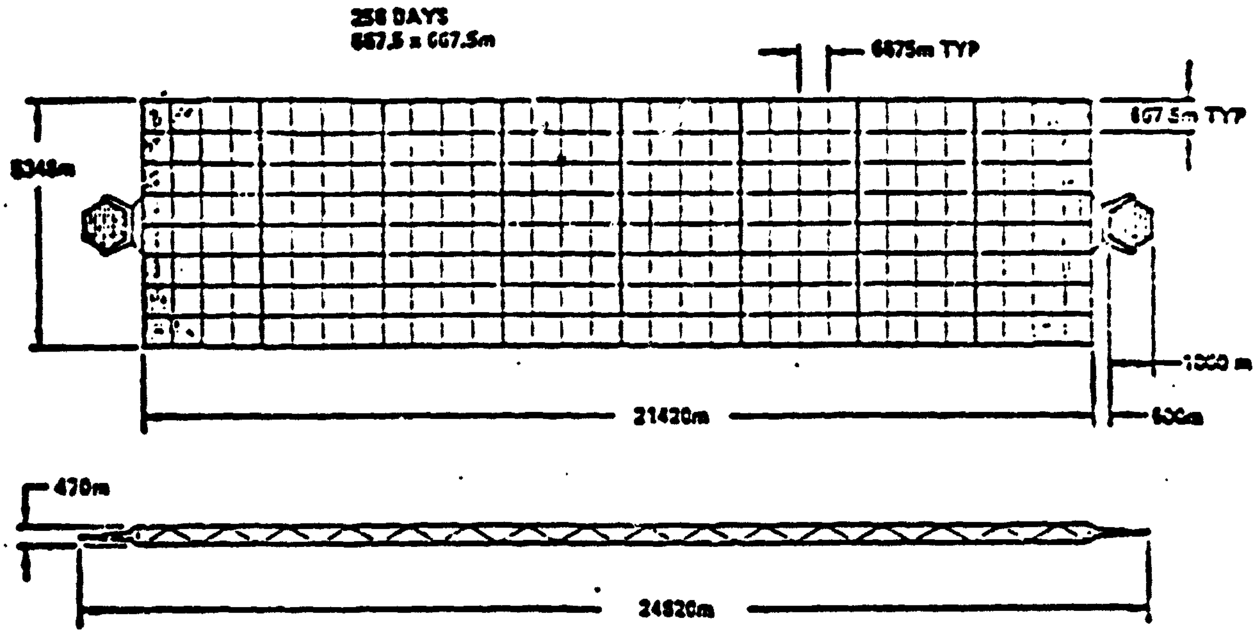
This analytic and conceptual design study program examined the spacecraft power distribution, and electrical loads and their influence on the cable and connector requirements for large space structures technology (LSST) spacecraft. Structural metarials' electrical characteristics and the probable fault currents to which the structure could be subjected, were analyzed and conceptual designs for LSST cables and connectors conceived. Available data from the United States Air Force and NASA space programs were used for some of the analyses and suggested conceptual designs. New materials, latching devices, and configurations, adaptable to automatic and remote construction in space were compared. The round configurations with close tolerance alignment pins, mounted on a mounting plate hold promise of eliminating much of the alignment problem. Each connector would require freedom of movement in two planes for self alignment.

2.0 INTRODUCTION

Studies have shown that many of the space missions proposed for the time period 1980 to 2000 will require large spacecraft structures to be assembled in orbit. Large antennas and power systems up to 2.5 MW size are predicted to supply the electrical/electronic subsystems, solar electric propulsion, and space processing for the near-term programs. Platforms of 100 meter length for stable foundations, utility stations, and supports for these multi-antenna and electronic powered mechanisms are also being considered.

A literature review was made of NASA, U.S. Air Force, and industry reports for large spacecraft structures. From this review it was concluded that spacecraft configurations with three power levels are planned between CY 1985 and 2005; i.e., (1) below 25kw, (2) 25kw to 2.5 megawatts, and (3) over 2.5 megawatts. Spacecraft with power levels to 25 kw may use 1980 state-of-the-art materials and cable and connector methodology. Spacecraft with power levels between 25 kw and 2.5 MW will require higher distribution voltages and currents, depending upon the spacecraft design. Very large spacecraft and space stations, planned for the 21st centruy, with power levels exceeding 2.5 MW will require high-voltage and high-current distribution lines, cables and connectors.

Three structural configurations for large structure space system with power levels over 25 kw, were selected for study of the electrical power system cables and connectors. These configurations are shown in figures 2.0-1, 2.0-2 and 2.0-3. The configuration of figure 2.0-1 has long lines to the loads which will result in large voltage drops or very heavy conductors if low voltages are used. Thus, high voltage systems and equipment will be considered for optimal design concepts. In figure 2.0-2, a cluster type configuration is shown. Lower voltage, higher current lines can be used for this construction, provided the current can be successfully handled by the rotary joint (low losses and voltage drops). The figure 2.0-3



TOTAL SOLAR CELL AREA:	101.2 km^2
TOTAL ARRAY AREA:	110.2 km^2
TOTAL SATELLITE AREA:	114.5 km^2
OUTPUT:	16.93 GW MINIMUM TO SLIPRINGS

FIGURE 2.0-1 SOLAR POWER SATELLITE CONFIGURATION (CIRCA CY 2000)

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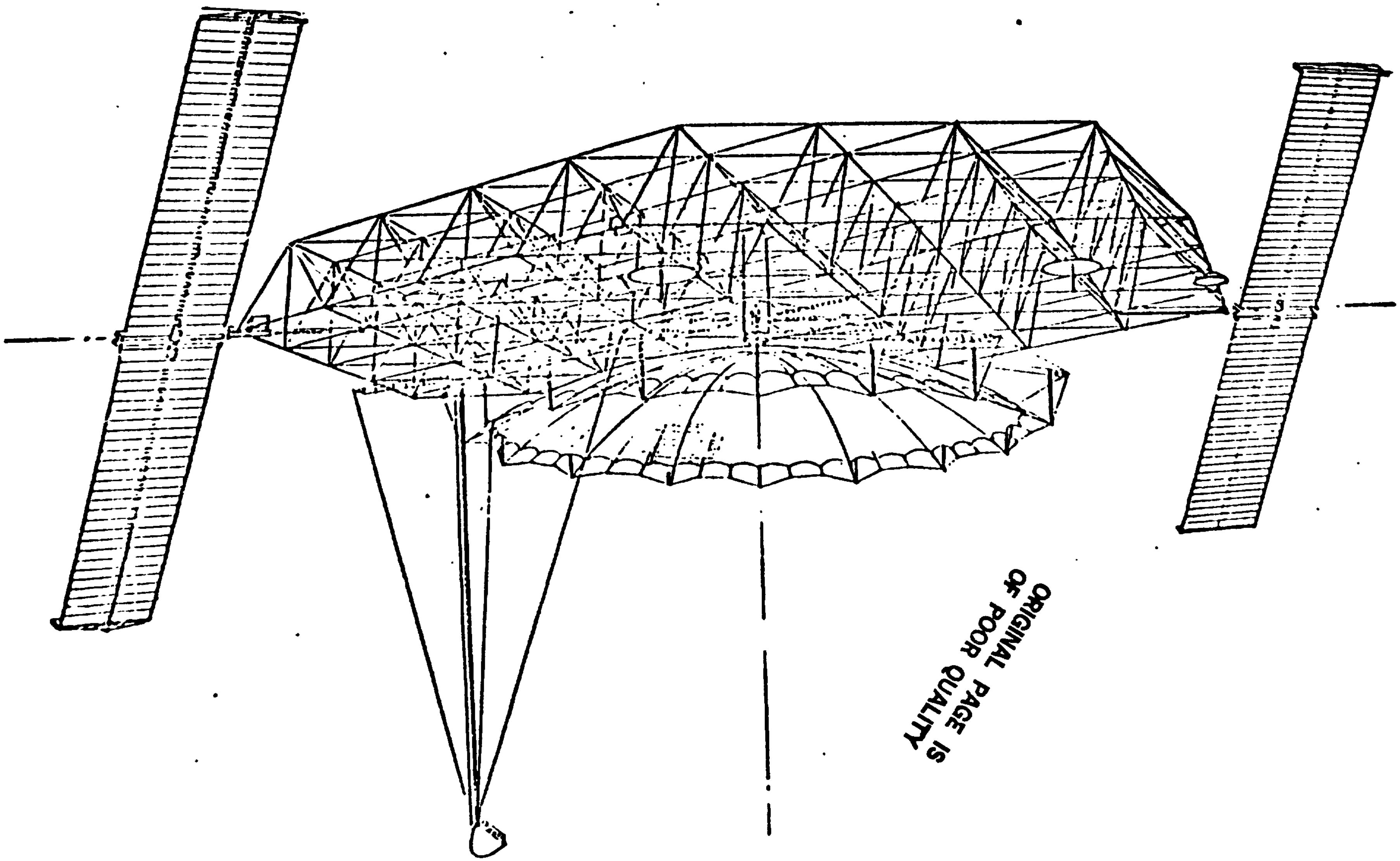
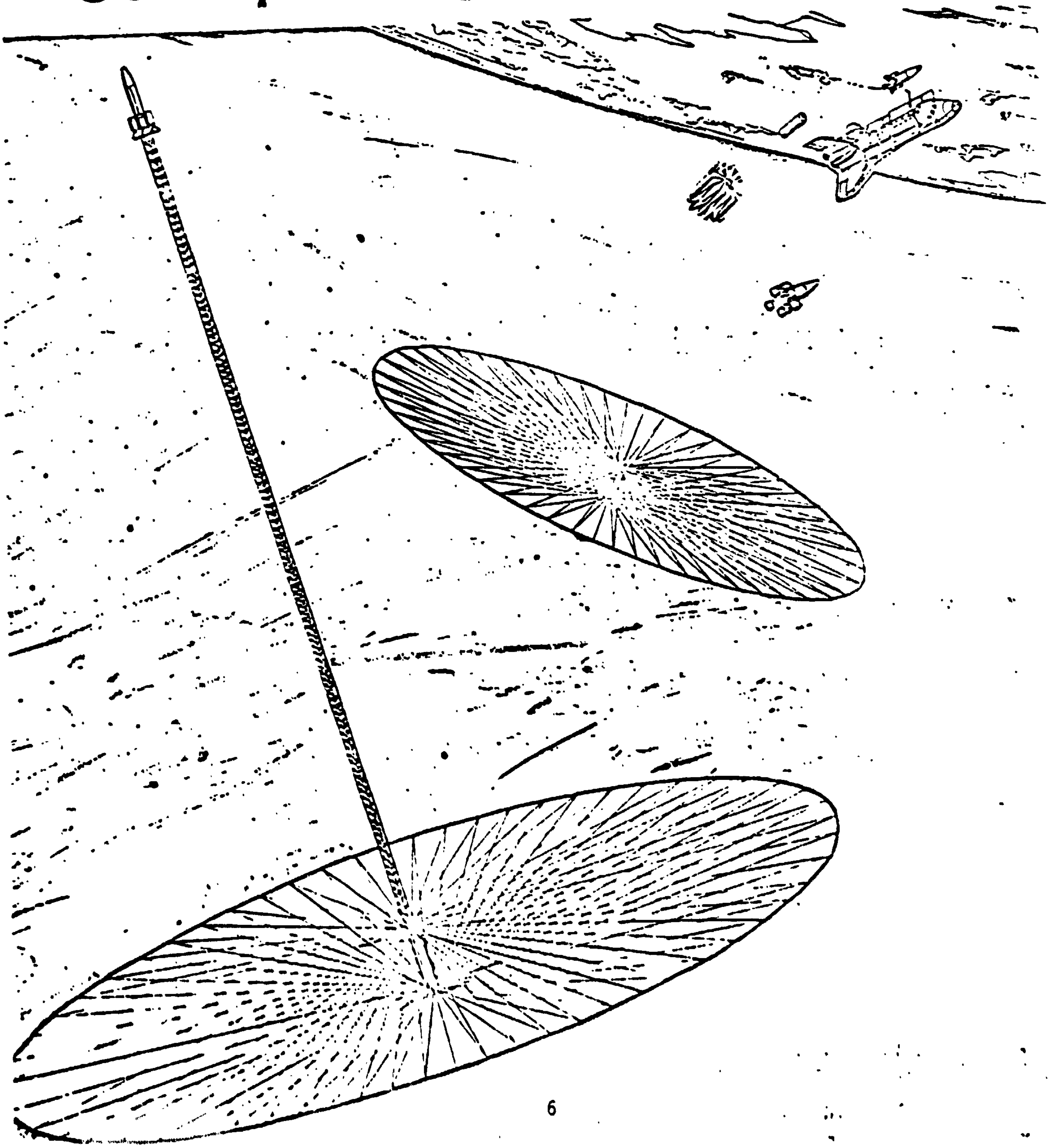


FIGURE 2.0-2 DISH ANTENNA SPACECRAFT CONFIGURATION

FIGURE 2.0-3

On-Orbit Assembly Concept Design



configuration has very long lines between the loads and the power source. This type of construction requires high-voltage transmission lines between the load and the generation center, similar to an electrical utility distribution network.

This report documents the conceptual analyses and requirements plan to be used to arrive at the recommended cable and connector concepts. A functional flow diagram is shown in figure 2.0-4 which depicts the methodology used for the study.

The study results include the following key items:

- New concept technology materials and equipment for cables and connectors.
- Environmental impact on voltage rating of bare and insulated cables and connectors on the spacecraft.
- Effects of voltage rating on the advanced cables and connectors.
- Influence of conductor protective coatings on breakdown and arcing.
- Conceptual conductor shapes and orientation as a function of voltage, heat transfer, line length, current, coatings and insulation, and environment.
- Connection and connector conceptual designs for lines and cables that are easily assembled on the spacecraft with minimum effort for the astronaut, or assembled automatically as the spacecraft is deployed and assembled.
- Maintenance and checkout procedures of the lines, cables, and connectors with and without the electrical power system energized. A schedule of events for maintenance personnel,

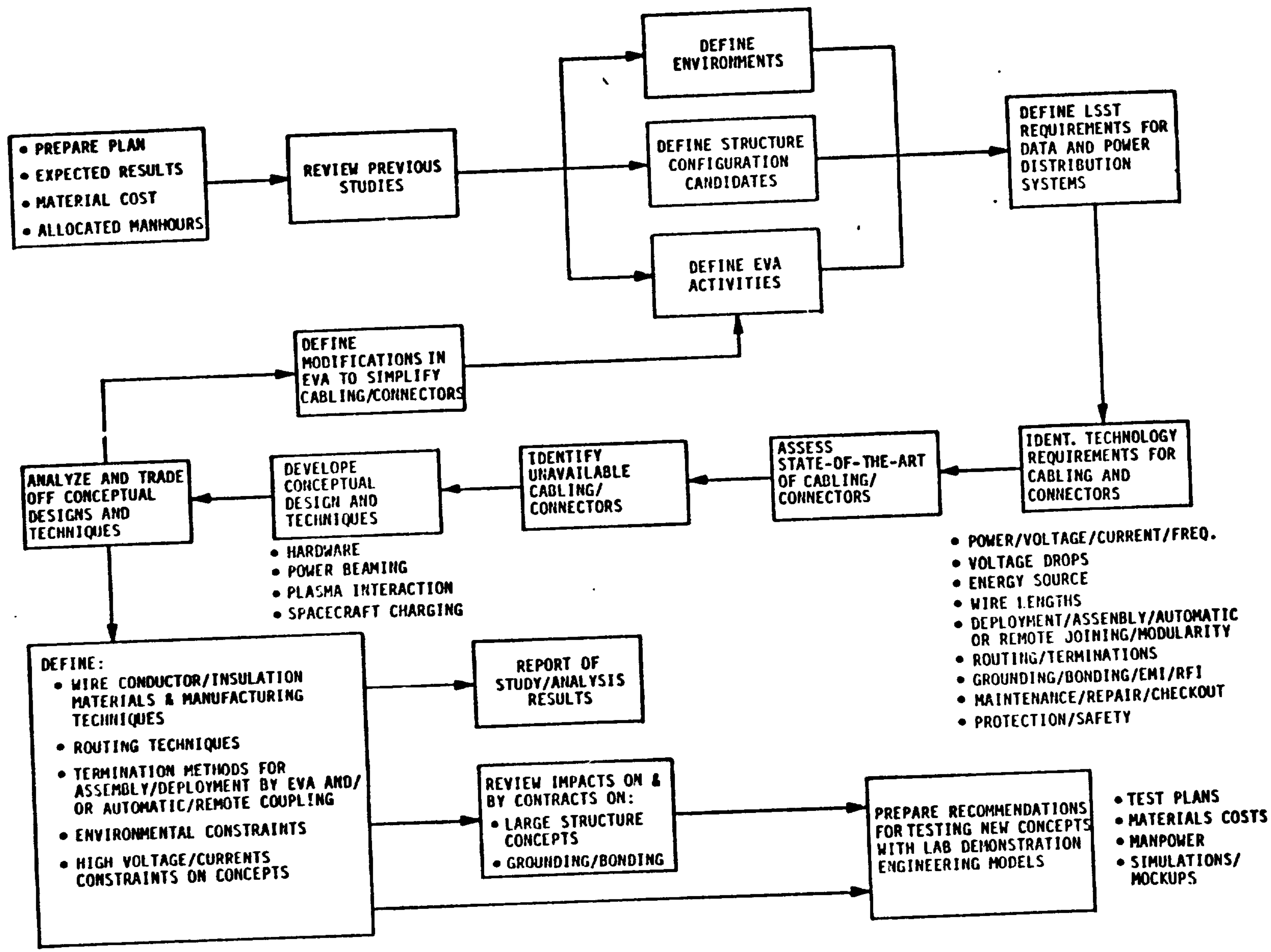


Figure 2.0-4: Functional Flow Diagram, Cable and Connectors

including sensors and necessary equipment requirements.

- Effects of outgassing products, spacecraft and space derived debris on the operation of long line cables and connections/connectors.

- Use of special tools and materials for the assembly of the cabling systems.

- Wire routing conepts for typical LSST vehicles.

3.0 BACKGROUND

A literative review of previous studies made by government, and industry was completed for spacecraft design configurations, power and vo.tage levels, and cable and connector requirements.

3.1 Survey

In addition, Boeing has published many documents for government agencies (including NASA), and has done independent research and development programs for airplanes and spacecraft. These documents also have been reviewed. One significant finding was the effects of high-current/high-voltage arcs on various structural materials. The graphite-epoxy materials withstand these arcs much better than many other types of epoxies, polyurethanes, and silicone-coated materials.

Many of the documents reviewed contained valuable information on the structural materials, (reference 1, 2, 3, 4, and 5) and the fabrication and assembly of those structural members on Earth and in space (references 6, 7, 8 and 9). Three documents were oriented toward the electrical reauirements and equipment (references 9, 10, and 11).

The data, in reference 11, is for the far-term, solar power satellite, a system having about 10-15 gigawatts of power output. That document and many Boeing dorcuments reviewed had much data on the power system schematic, but little data on the details involving cables and connectors or bonding and grounding. The data available were mostly oriented toward the cooling of the conductors, not the interconnects or fabrication in space by automatic or manual methods (reference 12). Information gleaned from reference 9 also involved power profile data of both large and small satellites. One document did have data on detailed concepts for cables, connectors, bonding and grounding (reference 9). This data was considered in the study. In that study (reference 9) it was assumed that an astronaut would be available to perform the latching of the connector latching device. This study included concepts for connections and attachments by automatic methods.

3.2 System Requirements

System requirements are dictated by the spacecraft mission, design, and operational life in space. Many spacecraft are under consideration by NASA for service in the CY 1980 through 2000 time period. Some are for near-term missions through 1989, while others are in the conceptual planning stages for the CY 1990 through 2010 time period. This contract will be oriented primarily toward the near-term missions, but with some applications studies for far-term missions.

Stanely R. Sadin shows a plot of space structures size and vehicle energy for the period from CY 1960 to CY 2000 (figure 3.2-1). Structural sizes are shown for 1976/1977 for Skylab, 1984 for Molecular Wake Shields, for Electronic Mail in 1990, and for SPS in 2000. Energy levels are shown as 10^4 KWH/yr for Earth Resources in 1982; and 10^5 KWH/yr for Space Manufacturing in 1985. Also listed in the planning tables given in the paper are other high power consuming systems such as the Large Power Module in 1986 and the SPS in 2000; the power level of SPS will be to 15×10^9 watts which is 876×10^6 KWH/yr to $1{,}314 \times 10^6$ KWH/yr. Once again the need for the LSST program is evident from the projected spacecraft requirements.

Lott W. Brantley shows a curve from CY 1980 through CY 2000 and power levels of 25KW Power Module in 1980 through the large power module, the SPS demonstrator, and a SPS of 5-10 GW in CY 2000. This also supports the requirement for the LSST program.

NASA near and far-term missions with respect to electrical power requirements are shown in figure 3.2-1 and in Tables 3.2-1 and 3.2-2. It is to be noted that many missions have electrical power requirements of several kilowatts to a few magawatts for the near-term missions. This implies that either very large currents at low voltage

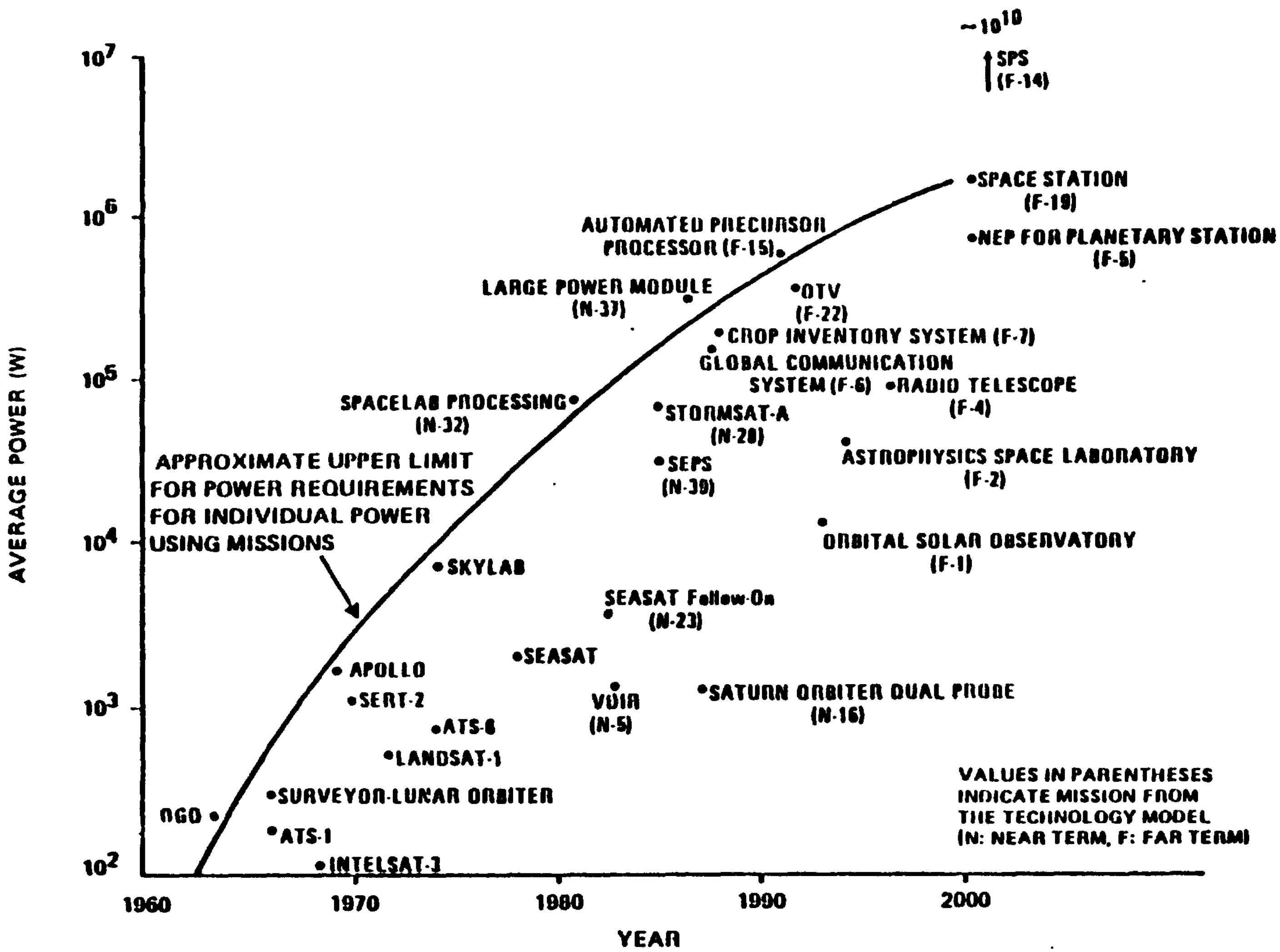


Figure 3.2-1: Mission Power Requirements

TABLE 3.2-1 INITIATIVE GROUP RANK ORDERING

INITIATIVE		IOC Date			POWER LEVEL
GROUP/ SUBGROUP	TITLE	OPTIMISTIC PROGRAM	STRETCHED PROGRAM	CONSERVATIVE PROGRAM	
2/1	PUBLIC SERVICE SYSTEMS USING LONG MICROWAVE STATIONKEPT ANTENNAS - I	1983	1983	1990	1.0 kW
3/1	POWER DISTRIBUTION SYSTEMS AND ACTIVE/PASSIVE RADAR - I	1982	1982	1989	1.0 kW
2/2	PUBLIC SERVICE SYSTEMS USING LONG MICROWAVE STATIONKEPT ANTENNAS - II	1987	1991	1994	1.3 kW
5/1	HIGH ALTITUDE NAVIGATION, LOCATION, AND RELAY SYSTEM - I	1983	1983	1990	1.7 kW
2/3	PUBLIC SERVICE SYSTEMS USING LONG MICROWAVE STATIONKEPT ANTENNAS - III	1992	1999	1999	2.0 kW
4 & 6/2	OPTICAL OBSERVATION, DESIGNATION, AND MEASUREMENT - I	1982	1982	1989	2.0 kW
9 & 11/1	SCIENTIFIC/RESEARCH EXPERIMENTS AND NATIONAL FACILITIES - I	1984	1984	1991	2.0 kW
5/2	HIGH ALTITUDE NAVIGATION, LOCATION, AND RELAY SYSTEM - II	1988	1992	1995	2.2 kW
5/3	HIGH ALTITUDE NAVIGATION, LOCATION, AND RELAY SYSTEM - III	1994	2001	2001	3.0 kW
1/1	SERVICE PLATFORMS USING MICROWAVE MULTIBEAM ANTENNAS - I	1983	1983	1990	4.0 kW
3/2	POWER DISTRIBUTION SYSTEMS AND ACTIVE/PASSIVE RADAR - I	1986	1993	1993	5.0 kW
4 & 6/2	OPTICAL OBSERVATION, DESIGNATION, AND MEASUREMENT - II	1986	1988	1993	5.0 kW
9 & 11/2	SCIENTIFIC/RESEARCH EXPERIMENTS AND NATIONAL FACILITIES - II	1988	1991	1995	5.0 kW
4 & 6/3	OPTICAL OBSERVATION, DESIGNATION, AND MEASUREMENT - III	1990	1994	1997	10.0 kW
7/1	SPACE PROCESSING AND MANUFACTURING - I	1983	1983	1990	10.0 kW
9 & 11/3	SCIENTIFIC/RESEARCH EXPERIMENTS AND NATIONAL FACILITIES - III	1993	2000	2000	10.0 kW
4 & 6/4	OPTICAL OBSERVATION, DESIGNATION, AND MEASUREMENT - IV	1995	2002	2002	20.0 kW
1/2	SERVICE PLATFORMS USING MICROWAVE MULTIBEAM ANTENNAS - II	1987	1990	1994	25.0 kW
8/1	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - I	1982	1982	1989	25.0 kW
3/3	POWER DISTRIBUTION SYSTEMS AND ACTIVE/PASSIVE RADAR - III	1990	1997	1997	50.0 kW
7/2	SPACE PROCESSING AND MANUFACTURING - II	1988	1992	1995	50.0 kW
7/3	SPACE PROCESSING AND MANUFACTURING - III	1993	2000	2000	100.0 kW
1/3	SERVICE PLATFORMS USING MICROWAVE MULTIBEAM ANTENNAS - III	1993	2000	2000	100.0 kW
8/2	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - II	1984	1986	1990	210.0 kW
3/4	POWER DISTRIBUTION SYSTEMS AND ACTIVE/PASSIVE RADAR - IV	1994	2001	2001	300.0 kW
8/3	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - III	1987	1990	1993	2.0 MW
8/4	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - IV	1992	1996	1999	15.0 MW
8/5	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - V	1996	2000	2003	1.0 GW
8/6	LARGE SCALE, HIGH ENERGY, FAR-TERM SYSTEMS - VI	2000	2004	2007	15.0 GW

TABLE 3.2-2 INITIATIVE SUBGROUP POWER DEMAND VS IOC DATE

OPTIMISTIC PROGRAM IOC											
1932-1984		1985-1987		1988-1991		1992-1994		1995-1997		1998-2000	
CONSERVATIVE PROGRAM IOC											
1990-1992		1993-1995		1996-1998		1999-2001		2002-2004		2005 - 2007	
Subgroup	Power	Subgroup	Power	Subgroup	Power	Subgroup	Power	Subgroup	Power	Subgroup	Power
2/1	1.0 kW	2/2	1.3 kW	5/2	2.2 kW	2/3	2.0 kW	4 & 6/4	20 kW	8/6	15 GW
3/1	1.0 kW	3/2	5.0 kW	9 & 11/2	5.0 kW	5/3	3.0 kW	8/5	1 GW		
5/1	1.7 kW	4 & 6/2	5.0 kW	4 & 6/3	10.0 kW	9 & 11/3	10.0 kW				
4 & 6/1	2.0 kW	1/2	25.0 kW	3/3	50.0 kW	1/3	100.0 kW				
9 & 11/1	2.0 kW			7/2	50.0 kW	8/2	210.0 kW				
1/1	4.0 kW			8/3	2.0 kW	3/4	300.0 kW				
7/1	10.0 kW					8/4	15.0 MW				
8/1	25.0 kW										

must be generated and distributed or the voltage must be increased to keep the current levels down. To meet the multimegawatt goals, both high current (over 1000 amperes) and high voltage must be considered. A plot of the current and voltage relationship to power and advanced technology cables, connectors, and grounding and bonding is shown in figure 3.2-2. Three voltage regimes are included in figure 3.2-2 Voltages to 200 volts, between 200 volts and 1,000 volts, and over 1,000 volts. There are many specifications and standards for the lower voltages between 0 and 200 volts (S), few for the transition voltage (T) regime (200 to 2,000 volts), and fewer for the high-voltage (HV) over 2,000 volts. Therefore, the space missions should identify the voltage level. This has been added to the missions shown in Table 3.2-3. Ground return via cables, connectors, on composite structures will all be influenced by the voltage, current, and power level of the spacecraft. Vehicle size will affect the cable size, thus the voltage drop and voltage variance across such items as solar panels. Secondary effects will include transients, traveling waves, electrostatic charging of the plates beneath the high-voltage cables, and the debris collected on the wires.

The technologies supporting the LSST program must be developed to provide those techniques and equipment compatible with the electronic data and electrical power distribution systems which will be a part of the large spacecraft. Consequently, identification of the requirements for the electronic data and power distribution systems and techniques, materials and components including cables and connectors are important to this program.

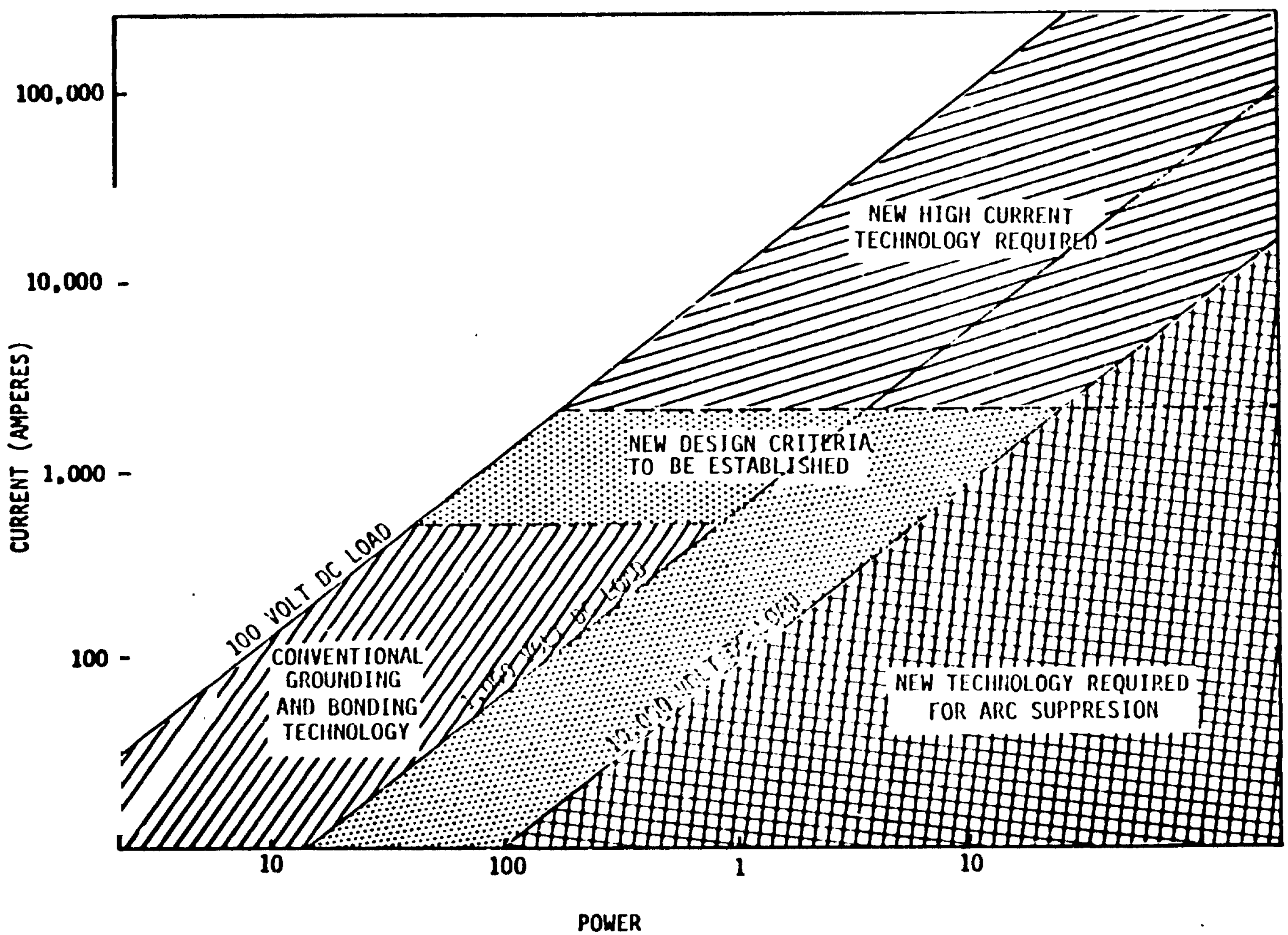


Figure 3.2-2: Bonding and Grounding Applications for Power Systems

TABLE 3.2-3 SPACE INDUSTRY OPPORTUNITIES

SPACECRAFT		POWER KW	TIME FRAME YEAR	ORBIT	VOLTAGE LEVEL 1
PERSONAL COMMUNICATIONS WRIST RADIO	(CC-9)	21	1990	SYNCH	S
URBAN/POLICE WRIST RADIO	(CC-2)	75	1985	SYNCH	S
3-D HOLOGRAPHIC TELECONFERENCING	(CC-11)	220	1990	GEO	
DIPLOMATIC/U. N. HOTLINES	(CC-10)	1	1985	SYNCH	S
NATIONAL INFORMATION SERVICES	(CC-8)	15	1990	SYNCH	S
ELECTRONIC MAIL TRANSMISSION	(CC-4)	15	1990	SYNCH	S
DISASTER COMMUNICATIONS SET	(CC-3)	75	1985	SYNCH	T
ADVANCED T. V. BROADCAST	(CC-6)	150	1990	GEO	T
ENERGY MONITOR	(CS-9)	23	1985	GEO	S
GLOBAL SEARCH & RESCUE LOCATOR	(CC-1)	1	1985	MED ALT	S
NUCLEAR FUEL LOCATOR	(CO-7)	0.3	1985	SYNCH	S
VEHICULAR SPEED LIMIT CONTROL	(CS-10)	1	1990	SYNCH	S
SYNCHRONOUS METEORLOGICAL SATELLITE	(CO-12)	1	1985	SYNCH	S
ATMOSPHERIC TEMP. PROFILE SOUNDER	(CO-11)	5	1990	600 N MI	S
WATER LEVEL & FAULT MOVEMENT INDICATOR	(CO-3)	0.25	1985	GEO	S
OCEAN RESOURCES & DYNAMIC SYSTEM	(CO-4)	25	1985	300 N MI	S
FIRE DETECTION	(CO-2)	2	1985	SYNCH	S
HIGH RESOLUTION EARTH MAPPING RADAR	(CO-13)	2,500	1990	400 K M	HV
ADVANCED RESOURCES/POLLUTION OBSERVATORY	(CO-1)	12	1985	500 N MI	S
U. N. TRUCE OBSERVATION SATELLITE	(CO-6)	3	1985	225 N MI	S
BORDER SURVEILLANCE	(CO-8)	20	1990	SYNCH	S
MULTINATIONAL AIR TRAFFIC CONTROL RADAR	(CO-5)	1	1985	300 N MI	S
TRANSPORTATION SERVICES SATELLITES	(CC-5)	0.6	1985	8,000 N MI	S
COASTAL ANTI-COLLISION PASSIVE RADAR	(CO-9)	3,000	1995	SYNCH	HV

(continued)

TABLE 3.2-3 SPACE INDUSTRY OPPORTUNITIES (CONTINUED)

SPACECRAFT		POWER KW	TIMEFRAME YEAR	ORBIT	VOLTAGE LEVEL 1
NEAR-TERM NAVIGATION CONCEPT	(CS-16)	1	1980	SYNCH	S
PERSONAL NAVIGATION WRIST SET	(CS-7)	2	1990	SYNCH	S
VEHICLE/PACKAGE LOCATOR	(CC-12)	23	1990	GEO	S
SPACE DEBRIS SWEEPER	(CS-11)	-	1985	LEO TO GEO	S
VOLTING/POLLING WRIST SET	(CC-7)	90	1990	SYNCH	T
ENERGY GENERATION-SOLAR TO MICROWAVE	(CS-1)	107	1995	SYNCH	HV
NIGHT ILLUMINATOR	(CS-6)	1.2	1990	SYNCH	S
NUCLEAR WASTE DISPOSAL	(CS-4)	-	1995	ESCAPE	S
ENERGY GENERATION-NUCLEAR/MICROWAVE	(CS-3)	10^7	2000	SYNCH	HV
MULTINATIONAL ENERGY DISTRIBUTION	(CS-8)	20	2000	225 NMI	S
POWER RELAY SATELLITE	(CS-15)	-	1995	SYNCH	HV
AIRCRAFT LASER BEAM POWERING	(CS-5)	-	2000	300 NMI	HV
ENERGY GENERATION HIGH EFF. SOLAR CELLS CONCENTRATOR	(CS-2)	10^7	1995	SYNCH	HV
TELEPHONE LONG LINE	(X-1)	100	1995	SYNCH	T
BURGLER ALARM	(CS-14)	1	1985	SYNCH	S
MILITARY COMMUNICATIONS WRIST RADIO	(MC-10)	100	1987	SYNCH	T
COMPUTER LONG LINE	(X-2)	400	1990	SYNCH	T
MILITARY AIRCRAFT COMMUNICATIONS	(X-3)	75	1990	SYNCH	T
MOBILE COMMUNICATIONS - TRUCK	(X-4)	750	1990	SYNCH	T
GLOBAL POSITIONING SYSTEM	(MS)	1	1980		S

1 S - 0 TO 200 VOLTS
T - 200 TO 2000 VOLTS
HV - OVER 2000 VOLTS

4.0 CABLES AND CONNECTORS

In this section are the cable and conductor analyses for high power distribution lines. Round or rectangular conductors are applicable for spacecraft systems having outputs of 75 kilowatts at 200 volts. Solar array outputs of 100 kilowatts to 2.5 megawatts should consider the use of flat conductors at higher current in order to improve heat dissipation. For low Earth orbits, space charging and/or plasma shorting become serious problems at very high voltage. Therefore, the distribution voltage magnitude must be limited if rectangular or sheet conductors are used rather than round or coaxial configuration.

4.1 Low Power Distribution

In studies by Martin Marietta (reference 15) and Rockwell International (reference 16), it was concluded that round, insulated, twisted pairs (shielded, as required) should be used for command, control, and low power wiring, that is, for power levels less than 1 KW or for very short lengths of wire runs. High frequency and communication coaxial cabling should be used or considered with interfaces to high voltage subsystems via fiber optics.

Spacecraft subsystems located distant (over 100 meters) from the central communications and/or load center should be linked to the central station by a communication link or fiber optics associated with a central computer. The 5 to 15 watts of power for the computer and radio link at the remote control center will be more efficient from an overall weight standpoint than the multi-pair conductors required to supply the signals to the remote center. It would also eliminate the probability of signal disruption and noise interference generated by the high voltage, high power portion of the solar array. It is recommended that the solar array have a low voltage power tap near each station.

4.2 Power Distribution Optimization

Cables for large spacecraft may be required to operate at high voltage and high current, dependent upon the physical layout of the solar array, the load power requirements, and the spacecraft orbit. The design of the power distribution system will require that the line losses be kept to an acceptable level without jeopardizing the thermal characteristics or subjecting the lines to arc-over. This assumes that high voltage switching devices, power conversion, power transfer, transformers and other equipment are available for protection and control of the spacecraft power generation and distribution system.

Near term, 1980 - 1990, spacecraft planning includes many opportunites for power levels to 25 kilowatts and a few for power levels from 50 kilowatts to 2.5 magawatts. Conventional low voltage levels of 28 volts dc to 200 volts dc are adequate for the systems operating at power levels to 25 kilowatts. However, lower voltages will result in high current operation with considerable line power losses at the higher power levels. Therefore, it is necessary to optimize, the distribution system for each application to minimize weight and power losses.

Many studies have been conducted on environmental interactions, such as the space plasma environment. Most of these studies indicate that the power losses will be small for distribution line and solar array voltages to 700 volts dc. There may be voltage limitations for the larger systems. This will need consideration during the power distribution system preliminary design.

One design feature that must be considered is the solar array power degradation with time. For long duration missions the initial power generated is greater than the power at the end of mission life due to degradation. This initial extra power capability may be used to offset some of the plasma power losses before the spacecraft is transferred to geosynchronous orbit.

The analysis that follows will not include environmental power losses as a voltage limiting design criterion since there are methods for preventing the problem.

4.2.1 Distribution System Mass

R. C. Finke, I. T. Myers, F. F. Terdan, and N. J. Stevens (reference 17) have shown that the mass of a spacecraft electric power distribution system is dependent upon three parameters:

(1) the power distribution lines

(2) the extra power system generation weight required to supply losses

(3) the heat rejection system weight required to handle the losses

Thus, the total weight penalty for the distribution lines is:

$$W_T = W_L + \Delta W_P + \Delta W_H$$

W_T = Total distribution system weight

W_L = Distribution line weight

ΔW_P = Extra power generation required to supply losses

ΔW_H = Heat rejection weight required

The distribution line weight can be calculated by

$$W_L = 2DAl$$

D = distribution line material density, g/cm^3

A = distribution line Area (cross section), cm^2

l = distribution line length, cm

The factor 2 is included in the formula because two lines are required, a positive and a negative or return line. Since most of the larger spacecraft will be designed using graphite epoxy structure a return line is necessary (return through structure may not be feasible, nor desirable for certain designs).

The heat rejection weight is given by

$$\Delta W_H = \frac{2I^2 \rho l \alpha_H}{A}$$

Where

I = distribution line current, Amperes

ρ = distribution line resistivity, Ω-cm

α_H = added conduction specific weight of the heat rejection system, g/watt

This added weight is that part of the conductor required to stabilize the conductor temperature at a given upper temperature limit. The third parameter, extra power generated, is given by

$$\Delta Wg = \frac{2I^2 \rho l \alpha_g}{A}$$

α_g = specific weight of the power generation system, g/watt

The total weight penalty is

$$W_T = W_L + \Delta W_H + \Delta Wg$$

$$= 2l\left[DA + \frac{I^2\rho}{A}\ (\alpha_H + \alpha_g)\right] .$$

The optimum current density, J, can be obtained by setting

$$\frac{d(W_T)}{dA} = 0.$$

This will give the minimum distribution line penalty.

$$\frac{D(W_T)}{dA} = 2l\left[d\ \frac{-I^2\rho}{A^2}\ (\alpha_H + \alpha_g)\right] \cdot 0$$

$$\frac{I}{A} = \left(\frac{D}{\rho(\alpha_H + \alpha_g)}\right)^{\frac{1}{2}}$$

Thus the optimum area, A_o, is

$$A_o = I\left[\frac{\rho(\alpha_H + \alpha_g)}{D}\right]^{\frac{1}{2}}$$

From this the weight of the optimized distribution line can be found in terms of the power transmitted by substituting into the initial formula $W_T = 2DAl$ and substituting power, $P = IV$. Then

$$W_T = \frac{2DlP}{V}\sqrt{\frac{\rho}{D}\left(\alpha_H + \alpha_g\right)}$$

Or

$$W_T = \frac{2Pl}{V}\sqrt{\rho D(\alpha_H + \alpha_g)}$$

Where

P = power in watts
V = line voltage in volts

The optimum distribution line weight for a two-line, direct current system, in terms of power, P, output and line voltage, V, is

$$W_T = \frac{2Pl}{V} \sqrt{\rho D(\alpha_H + \alpha_g)}$$

The specific masses of the heat rejection system and power system, $(\alpha_H + \alpha_g)$, depend upon the shape and orientation of the conductors, the operating temperature of the conductors, the conductor material, and the construction of the solar array and cells. For this simplified anaysis it will be assumed that the conductors will be constructed of wide, thin, aluminum sheets, with the sheet oriented toward free space at the edge of the solar array. Aluminum sheet material was selected for its weight, strength, cost, and it's ease of fabrication. Other materials that may be used as conductors are shown in Table 4.2.1-1.

TABLE 4.2.1-1 Conductor Materials

Material	Resistivity Micro Ohm-cm (20^oC)	Density g/cm^3 (20^oC)	ρd $(\Omega\text{-}g)/cm^2$ $(x10^{-6})$
Aluminum	2.828	2.7	7.64
Beryllium	5.9	1.82	10.74
Copper (annealed)	1.724	8.89	15.33
Graphite	1589. (avg)	1.58	2507.
Lithium	8.55	0.54	4.62
Magnesium	4.6	1.74	8.0
Silver	1.629	10.5	17.1
Sodium	4.3	0.97	4.17

Several materials have better minimum weight than aluminum. Lithium and sodium have much better resistivity-density characteristics. But, these materials, along with beryllium and magnesium, are difficult to fabricate and store. Copper, silver, and silver alloys are heavier than aluminum but are much easier to join and will be considered for the connector materials. Graphite characteristics indicate that it should not be considered for the ground return; its resistivity is much higher than most metals. A more extensive study should be made of the silver alloys and copper alloys. Some of them may have good characteristics approaching that of aluminum. Aluminum materials do tend to crack and set with time. Continual flexing (caused by electrical transients) could cause joints to weaken with time.

4.2.2 Thermal Evaluation

Electrical conductors may be designed of either round or rectangular configuration. Round conductors may be used for low current distribution lines, and rectangular conductors may be used for distribution lines carrying more than 150 amperes per square centimeter. Round conductors will be used for lower current control, communication, and distribution lines. An analysis for the thermal evaluation of conductors follows. In this analysis it is assumed that rectangular solar arrays will be used. Each array will have less than six parallel distribution lines. For a greater number of lines the temperature rise of the centrally located conductors will be greater than the outer conductors due to the restricted heat radiation to space.

Aluminum sheet conductors may be selected for the higher current applications since they have better heat radiation to space than round conductors. Mounting provisions can be provided for large sheets on the back side of the primary structure. In order to determine the sheet conductor performance as a function of a solar array configuration, the curve shown in figure was developed. This curve is of the form

$$F(T) = \frac{I}{W \quad t}$$

where

I = Conductor current in amperes

W = Conductor width in centimeters

t = Conductor thickness in centimeters

$F(T)$ = Current density function of conductor operating temperature (^{o}C)

The curve was developed using a Boeing-derived thermal radiative interchange factor computer program. This program includes a ray trace capability to generate views of the conductors to free space and to the spacecraft to compute the conductor thermal performance, for the conductor location.

For each conductor unit length the mass per unit (M/L) can be computed using the curve of Figure 4.2.2-1 for any operating termperature as follows:

Where $M/L = Wt\,\sigma$

M/L = mass per unit length (gm/cm)

σ = metal density, g/cm^3

By Substitution $M/L = \frac{I\sqrt{t\,\sigma}}{F(T)}$

For conductor resistance

Where $R = \frac{\rho L}{A}$

A = Conductor area cm^2

ρ = Conductor resistivity, Ω-cm

R = Conductor resistance (ohms)

The conductor I^2R loss per unit length

$$\frac{I^2R}{L} = \frac{I^2}{L}\frac{\rho L}{A} = \frac{I^2\rho}{A} = \frac{I^2\rho}{Wt}$$

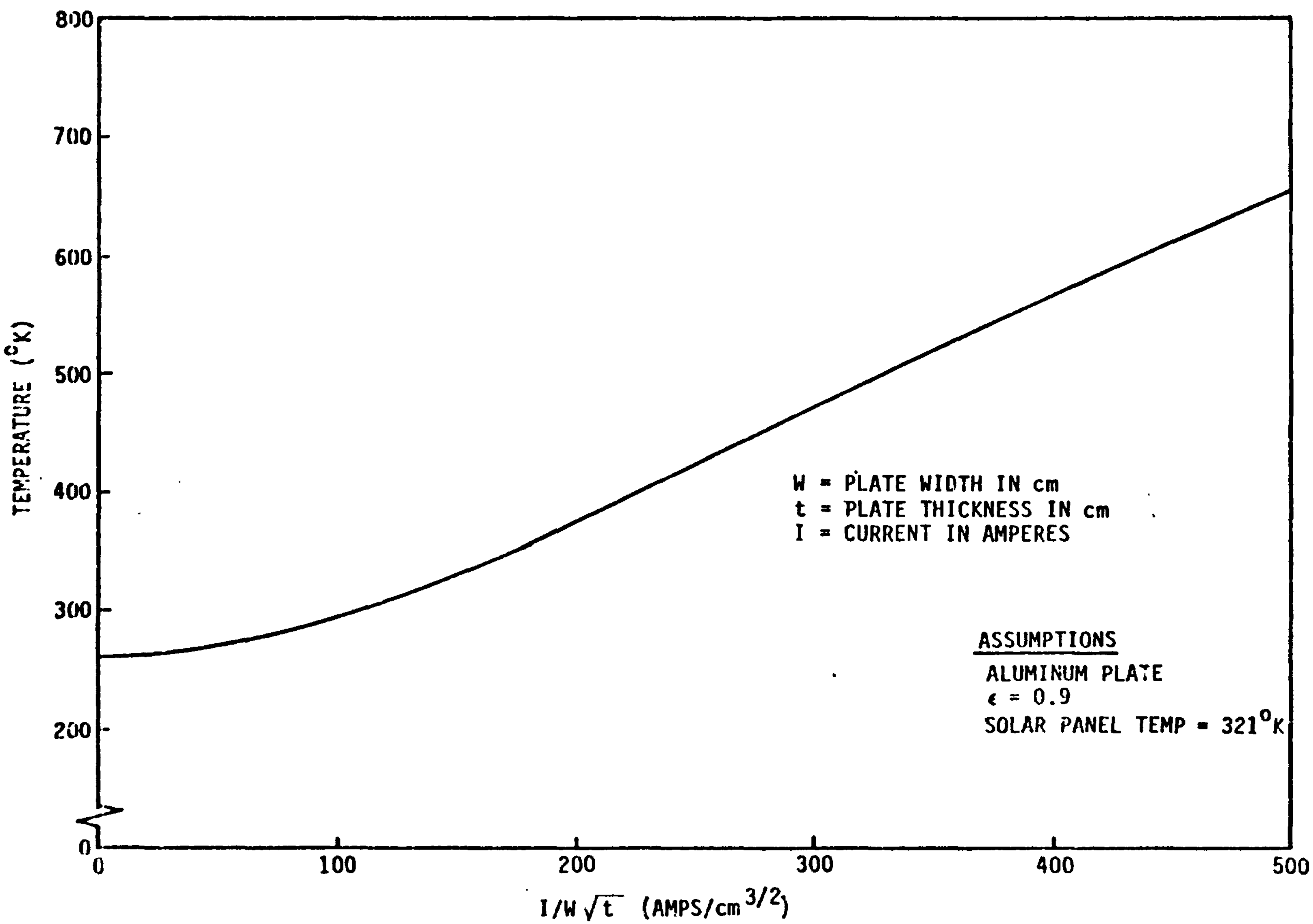


Figure 4.2.2-1: Current Density - Temperature Characteristic of Flat and Circular Conductors

but again

$$\frac{I}{W\sqrt{t}} = F(T)$$

then the per unit loss

$$\frac{I^2R}{L} = \frac{I^2\rho}{\frac{I\sqrt{t}}{F(T)}} = \frac{I\rho F(T)}{\sqrt{t}}$$

A large satellite has a specific mass of the order of 7 grams per watt for a 10-20 year life in synchronous orbit. The total mass which is attributable to the conductor system is the mass of the conductors plus the mass of the array required to compensate for the I^2R loss of the conductor system. That is:

Total Mass = Conductor Mass + 7 x conductor I^2R loss

For flat conductors the heat refection is by radiation to space, or absolute zero. On a per unit conduction length basis:

$$\frac{M(\text{total})}{L} = \frac{I\sqrt{t}(\sigma)}{F(T)} + \frac{7\,I\rho F(T)}{\sqrt{t}}$$

or

$$\frac{M(\text{total})}{L} = I\sqrt{t}\left[\frac{\sigma}{F(T)} + \frac{7\,\rho F(T)}{t}\right]$$

For the conductor segment represented by the curve in figure 4.2.2-2, a conductor operating temperature can be selected which will minimize satellite mass.

The specific mass is calculated by substituting the following information into the equation.

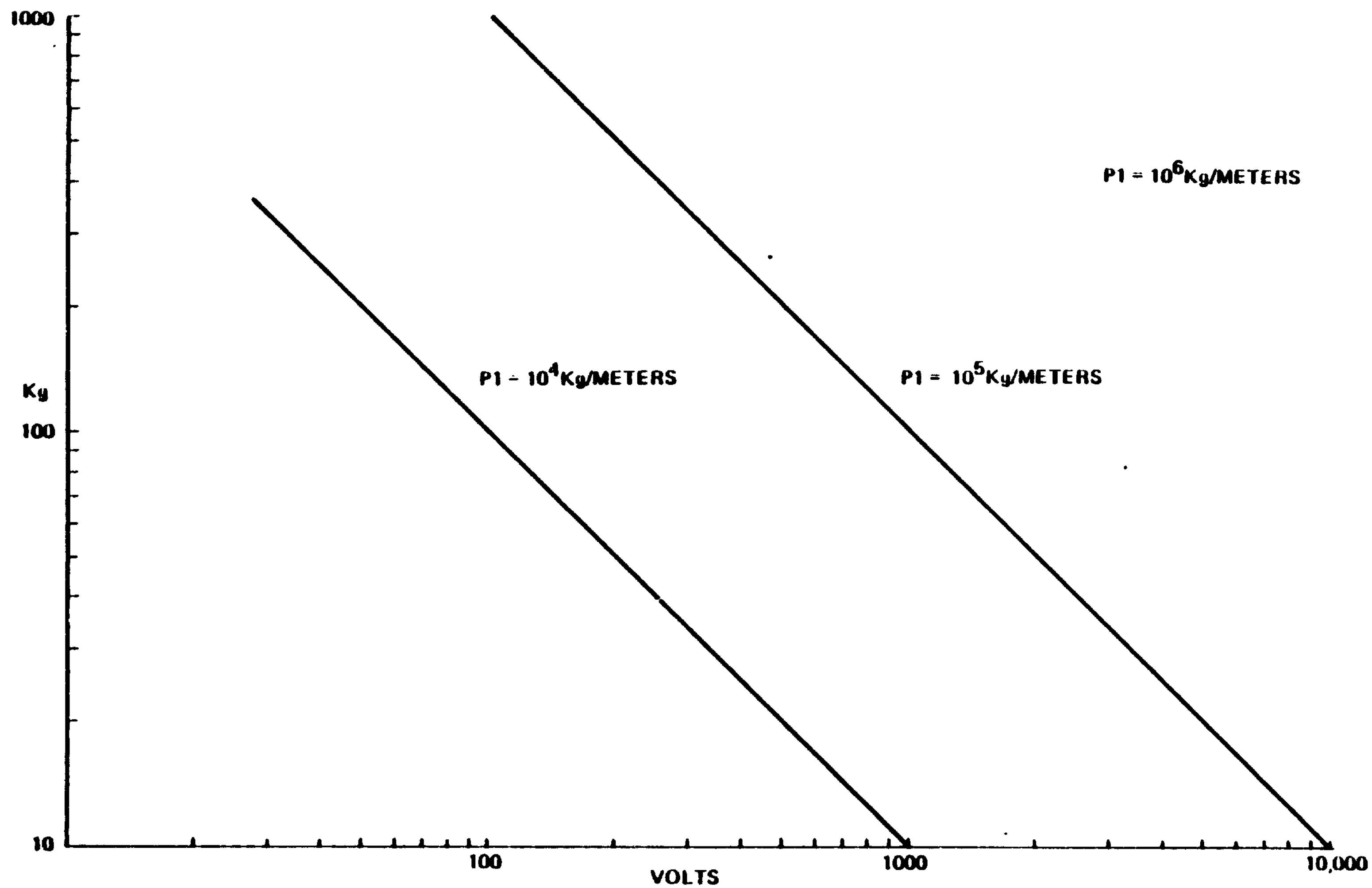


Figure 4.2.2-2: Distribution Line Mass Versus Voltage

$$\sigma = 2.7 - \left[(24.1 \times 10^{-6})^3 / {}^{\circ}C \right] \quad g/cm^3$$
above 27°C

$$\rho = 2.824 + (0.004/{}^{\circ}C) \text{ micro ohm-cm}$$
(at 20°C)

A reasonable operating temperature is 100°C. That is, 50°C above the nominal temperature of the solar array. Higher temperature will increase the incremental temperature of the solar array section under which the conductor passes. At 100°C

$$\rho = 3.144 \text{ ohm-cm}$$

The linear expansion of the conductor is less than 0.3% for the temperature range 0°C to 100°C. This will cause a slight bowing of the conductors between standoffs with the standoffs stationed on graphite epoxy structure.

The optimum distribution line weight for solar arrays operated at 100°C carrying a load of 150 amperes is

$$W_T \text{ (optimum)} = \frac{2Pl}{V} \sqrt{\rho D(\alpha_g + \alpha_H)}$$

Where

ρ = 3.144 micro ohm-cm

D = 2.7 g/cm^3

ρD = 8.49 × 10^{-6} g/cm^2

$$\alpha_g + \alpha_H = 30 \text{ g/W (assumed)}$$

Then

$$W_{T(\text{optimum})} = 10.092 \frac{Pl}{V} \text{ grams}$$

Power distribution line weights as a function of line voltage for high power systems are shown in figure 4.2.2-2. Calculations are shown in Table 4.2.2-1.

TABLE 4.2.2-1: Calculated Distribution Line Weight as a Funtion of Line Voltage

μΩ-cm x 1 / VOLTS		10^1	10^2	10^3	10^4
	28	360 Kg	3,600 Kg	36,000 Kg	360,000 Kg
	100	101	1,010	10,100	
	250	40.4	404	4,040	
	750	13.5	135	1,350	
	1000	10.1	101	1,010	
	2000	5.05	50.5	505	
	5000	2.02	20.2	202	
	10,000	1.01	10.1	101	

4.2.3 Mechanical Forces

There are always mechanical forces to contend with when building a high current distribution system. Compressive forces and curling forces must be considered for a direct-current system. These forces become large for closely spaced, high current lines. The equations for these forces, the force between conductors, and the force between conductor and the Earth's magnetic field are given in the following:

1. Compressive force

   $F/l = 1.386 \times 10^{-7} I^2/W$

   F = Force in Newtons
   I = Current in Amperes
   W = Width of Sheet in Meters
   l = Length of Sheet in Meters

2. Curling Force

   $$F/l = (I/A)^2 \Big\{ 2t(w^2-t^2/3)\tan^{-1} t/a + W$$
   $$+ 2a \left[t^2 - ((W+2a)/3)^2 \log ((w+2a)^2 + t^2) / (w+a)^2 \right] \Big\}$$

Where

F = Force in Dynes
I/A = Current Density in Amps/cm^2
l = Length in cm
t = Thickness in cm
W = Width in cm
a = Width of Selected Area at the Edge of the Sheet

3. Force Between Conductors

   $F/l = (2 \times 10^{-7} I_1 I_2)/b$
   F = Force in Newtons
   $I_1 I_2$ = Conductor Currents in Amperes
   l = Length in Meters
   b = Conductor Spacing in Meters

4. Force Between Conductors and Earth's Magnetic Field

   $F/l = BI \sin \theta$
   F = Force in Newtons
   l = Length in Meters
   I = Current in Amperes

B = Magnetic Flux Density in Webers/Meter^2

θ = Angle Between the Conductor and the Lines of Magnetic Flux

High current transients and faults will subject the structural separators and standoffs between conductors and between conductors and ground to forces are great as 2.5 times normal operating forces. These forces will tend to pull together and curl flat conductors simultaneously (figure 4.2.3-1). In addition, large stresses will be imposed on the structural (high resistance) ground. This conductor movement will flex and stretch the conductors, placing heavy tensile loads on the mated connectors and line joints.

An example of the impact of one of these forces can be shown for the forces between conductors (eq. 3) where:

For a system with 1,000 amperes of current wires spaced 1 meter

F = 0.045 lbs/meter of wire length

For a 10 MW system with a 4 KV operating voltage

F = 0.28 lbs/meter

4.3 Wire and Cable Routing

Wire and cable routing will differ for each solar array and power utilization center configuration. Some general routing guidelines are delineated in this paragraph which may simplify problem areas on the solar array panels and within the high voltage modules.

4.3.1 Solar Arrays

Solar cell connections are unique to the type cell and size of the cell. For large panels each series cell group should be connected so as to produce near neutral electric and magnetic fields. For currents up to about 500A, the fields will not be

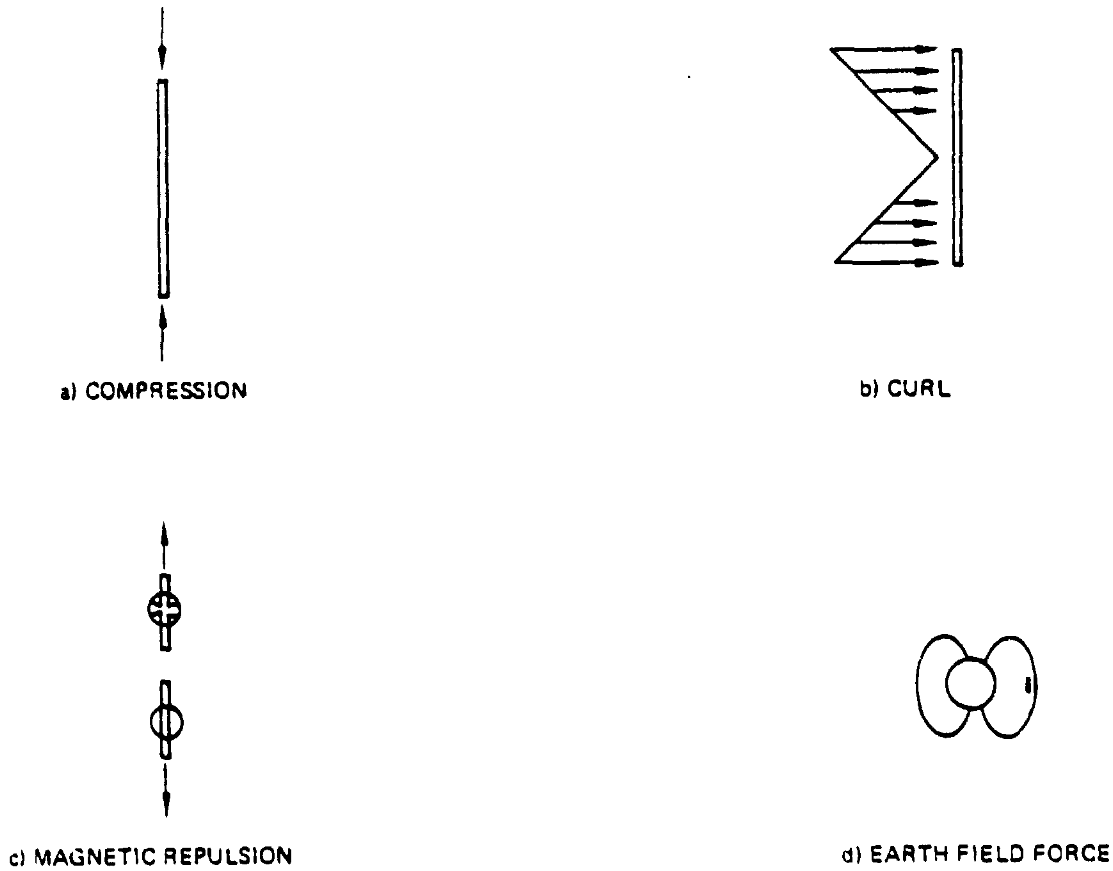


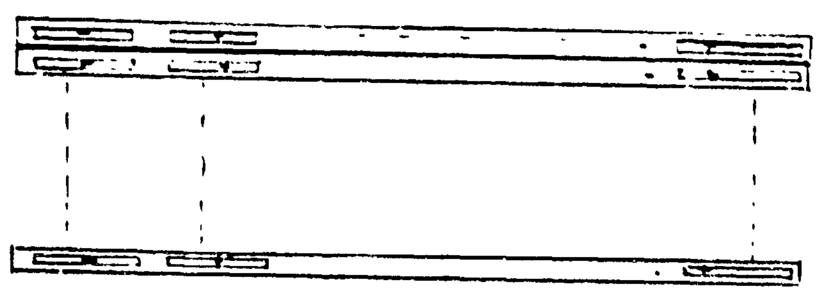
Figure 4.2.3-1: Sheet Conductor Loads

sufficient to significantly protect the array from plasma or particulate damage. Field neutralization can be accomplished by having the series cell strings fold back on themselves. In addition, the positive and negative power line connections should be kept close together.

Both power lines from a cell group should be connected to a single two-wire tape to produce a neutral magnetic field. Many two-wire tapes can then be brought to the edges of the solar array or along the center pole to continue the neutralization of electric and magnetic fields. This wiring technique produces a case of near zero magnetic and electric fields, allowing the plasma charge to build up on the array according to the quantity and charge of the incoming flux.

Flat, rectangular conductors in tapes are excellent for interconnecting the cell groups on a large panel producing up to 5 kilowatts of power. The near neutral electric and magnetic fields can be achieved and the tapes can be stacked on the edges of the panel.

Stacking several tapes will result in a large capacitance between conductors unless all negative conductors are positioned over each other. By placing the negative conductor s as shown in sketch A the capacitance can be kept to a minimum.



Sketch A

For instance, a tape made up of several flat conductors 0.075 cm thick spaced 0.1 cm apart in a dielectric media of $\varepsilon_r = 3.5$ have a capacitance of approximately 35 picofarads/meter for each pair of conductors. A 50 kw panel could conceivably have upwards of 50 conductor pairs in parallel or a capacitance of 0.0017 microfarad/meter. This capacitance will have some storage capability to decrease the effects of transients but will add to the power source energy to do greater damage in case of a short or arc. Both conditions should be considered in the wire routing design.

Conductors from the panels all have some resistance, therefore, line voltage drops become a problem. Distant panels could be operated at either higher voltage or the cells operated off the peak power point. Either way, the overall panel efficiency will be less than the solar array average. To decrease this problem all conductors from one to four panels could be paralleled and the power transferred to the utilization point by the two parallel pairs (one on each side of the array). The conductors to the distant panel group would have to be considerably larger in cross section area than those to the near (utilization point) group to compensate for voltage drops. A trade study should be made to determine the need for this kind of wire routing.

The larger conductors should be mounted for thermal cooling by radiation to space when possible. This will require special connections to the solar array panel. The flat conductor configuration could be continued to the rotary joint slip rings. From the rotary joint to the utilization point round or flat conductors could be used depending upon the voltage and heat transfer equipment used therein.

Wiring in high voltage equipment compartments should be of round conductors. For high power and high voltage round tubing is preferred. The tubing will hold its configuration during vibration and shock much better than insulated flexible conductors. Round tubing also has fewer problems with voltage breakdown, corona, and "clumping" or Malter effect than sharp edged square or small solid conductors which have high electric field stress at the corners or small radii. Tubular wiring should be considered for all open-construction type electrical/electronic equipment and for the interconnections between the high voltage modules. Screen-type shielding housing should be designed around those open constructed modules to reduce electromagnetic interference.

4.3.2 Electrical/Electronic Equipment Cables and Connectors

High-voltage, high-power cables and connectors for electrical/electronic equipment will be made of either flexible coaxial configuration or round tubing. The coaxial connectors must be machine or wrench torqued. Ample space between the connector shell and equipment must be available for portable hand tools used by the crew member. A minimum of 5 cm spacing is recommended.

High-voltage tubing will be made with solid taper metal fittings with smooth, rounded clamping nuts (corona ball configuration). Special tools with soft gripping surfaces (rubber) will be required to connect/disconnect the clamping nuts.

4.4 Wire and Cable Concepts

Conductors designed for high current (over 150 amperes) must have very low resistance, be able to radiate heat to space, or have external cooling. In space external cooling loops for wire bundles are impractical. Thus conductor designs should be such as to radiate heat to space or low resistance conductors should be considered for high current applications.

High current conductors to and from the rotary joint should be large, low resistance, rectangular conductors. Heat loss should be kept to a minimum because all heat from the rotary joint will be by active or passive radiators. Rectangular and/or tubular conductors can be designed for structural strength and rigidity to withstand the vibration and shock imposed on the spacecraft modules during Earth to orbit transport.

The physical shape of the conductor can be used to increase heat removal capabilities. A cylindrical conductor would have the lowest surface area to current density ratio. A method to achieve a larger surface area to current density ratio would be to use a rectangular cross-section. This concept can be extrapolated to the extreme case where very thin, wide conductors are used such as in the high power distribution system concepts. In these concepts (figure 4.4-1), to minimize satellite mass, conductor grade aluminum sheet was selected for the main and acquisition buses. Analysis of conductor operating temperature versus mass led to the choice of a conductor operating temperature of $100^{o}C$. A one millimeter conductor thickness was selected as the minimum gauge on the basis of handling and assembly. This leads to the result that the buses are 0.0158 centimeters wide for each ampere carried.

Holding the conductors in place are insulating standoffs spaced incrementally. The spacing between conductors will be large enough to prevent arcing if a conducting medium (outgassing, or ionized particles) enters the region between the conductors. By placing the conductors one above the other, the fields between the conductors are more favorable and the cooling of the conductors is optimized. Curving, or rolling the edges minimizes field stress concentrations which sharp edges cause. The conductor sheets can be brought up in rolls and unrolled as the installation proceeds. To provide some protection when astronauts would be maintaining the spacecraft while the electrical system was operating (similar to electric utility maintanance crews working on 'hot' power lines so as to avoid shutting down the system), a very thin conformal coating could be applied to the conductor. In a final design, tradeoffs would have to be made as to the coating thickness, the temperature effect, the effect on the conductor weight, and the relation to system revenue.

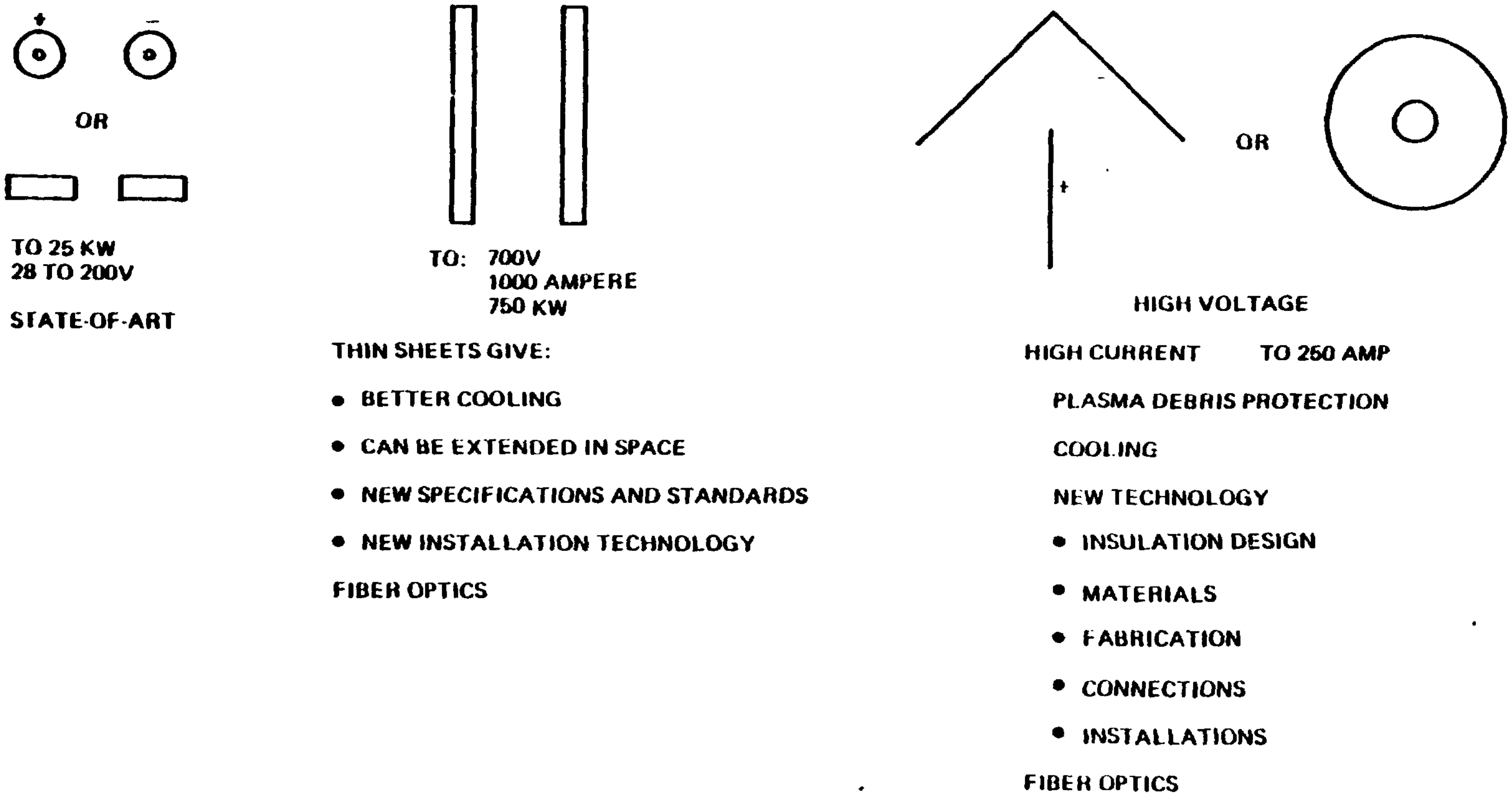


FIGURE 4.4-1: Conductor Configurations

4.5 Routing Techniques

Candidate structural configurations will influence routing techniques. For some concepts the electrical distribution and the electronics data systems can be placed in troughs in the structures which are deployed and assembled with the structure. In some cases, these can be in different troughs because of EMI/EMC. Or the cabling can be installed and fastened to the surface structure. When the electrical power cable requires a design such as shown in figure 4.5-1, the routing again varies because of the design of the standoffs to accommodate the high voltage in the system. The system to be grounded by the electrical conductor and all structural elements will be bonded to the ground bus. Each load location will be serviced by the electrical ground conductor to which all equipment boxes will be bonded. The significant difficulties will be encountered as the electronic data and electrical distribution systems encounter structural assembly joints, flexible joints, and rotary joints. Although cable lengths will be minimized, the structural design will dictate where cables will be fastened. There will probably be minimum crossing of cables over areas where no structure exists. This is caused by a lack of fastening for the cables, and a lack of handholds for an astronaut working during EVA duties. Larger structures will have more open areas than smaller area structures.

Routing over flexible joints will have to consider the amount of joint movement allowed, which will influence what type of transmission line joint will be made over the structural joint. The electronics data and electrical power distribution will have to have a flexible connection when crossing any such joint. Examples of two sheet conductor flex joint concepts are shown in figures 4.5-2 and 4.5-3. Several other concepts will be developed and trades for cost, mass, and assembly operation made.

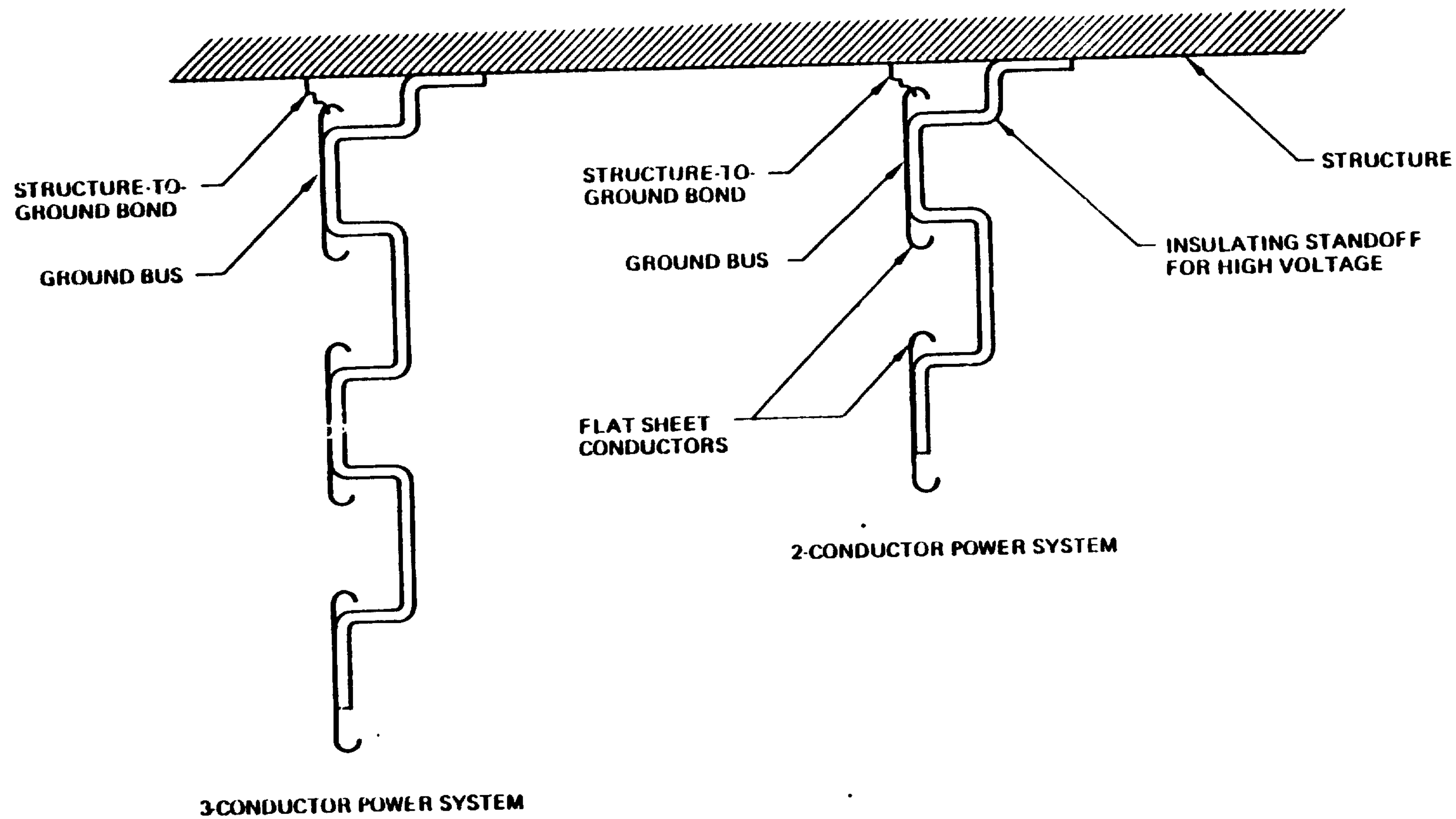


Figure 4.5-1: Power Distribution System Conductor and Mounting Concepts

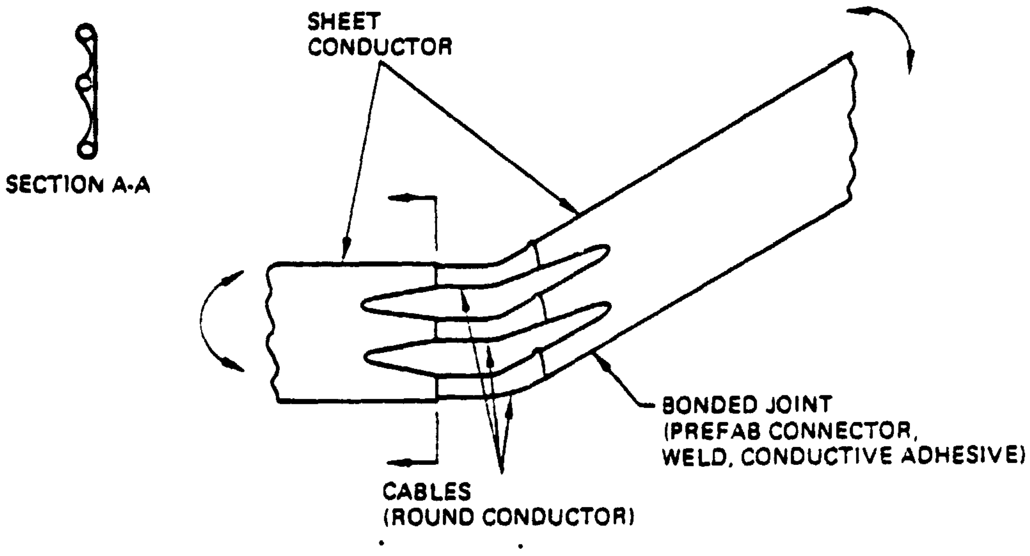


Figure 4.5-2: Flex Joint Concept for Vertical Movement Between Sheet Conductors

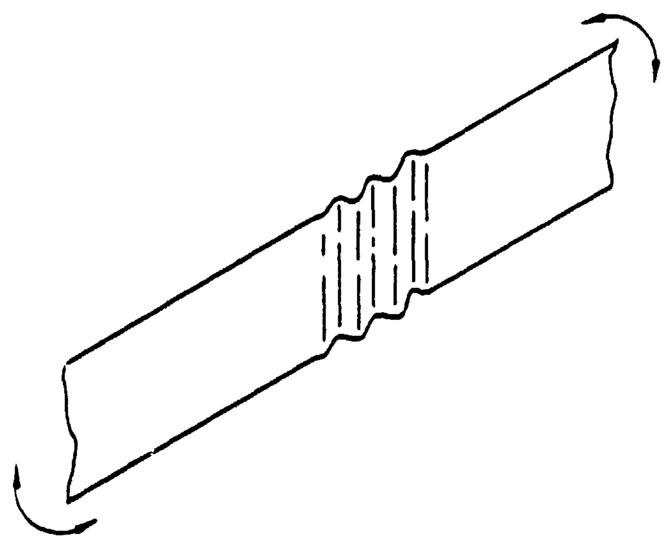


Figure 4.5-3: Corrugated Flex Joint for Sheet Conductor

4.6 Solar Array Cable Open/Fault Instrumentation

Each solar array wing on the spacecraft will be made of many panels. Each panel will have a number of series/parallel connected solar cells within a panel section. An open or shorted wire pair to one section can be determined by one of three candidate types of sensors. They are:

> differential temperature sensor, temperature sensitive paints, or an in-flight recorder of voltage and current from each section.

The differential temperature sensor appears best because it can sense temperature differentials of 0.5^{o}C and the backside differential temperature between operating and non-operating solar array sections will be at least 5^{o}C, depending upon the solar cell operating efficiency.

The problem with temperature sensitive paints is that they outgas and have to be replaced regularly, because of deterioration. Therefore, they are incompatible with spacecraft design objectives. An in-flight test set is bulky, requires installation of sensors on each wire pair, has to be monitored periodically, and lowers the system reliability. It also is not recommended.

Once a solar cell string has been determined to have failed, an impedance sensor can be used to determine if it is in the wiring of the cells or the wire pair. If it is in a wire pair, then one of the spare wire pairs can be attached in parallel. It must be remembered that the whole high voltage (> 32 volts) solar array section (or array) must be de-energized before work on measurements or repair can be initiated.

All spare conductors should be pre-tinned so that a solder diffusion bond can be made between the new conductors and the section. A handcarried, battery-operated bonding tool will be designed and used for this purpose.

5.0 CONNECTORS

A state-of-art- study was made to determine which connectors are available or have been used successfully on spacecraft in similar orbits and/or with comparable power levels. Since none of the space systems have yet been at the sizes and power levels of the systems in the proposed study, unavailable items will be identified. Conceptual designs will be made for the unavailable connectors and will be analyzed for performance with the types of spacecraft structures, sizes, and dimensions selected by the system structure designers.

5.1 Connector Survey

The objective of the survey was to determine existing electrical connector availability and to analyze, study and establish optimum connector designs for maintainable spacecraft substation interfaces.

A review was made of existing electrical connectors for new and old spacecraft systems to determine if state-of-art connectors are suitable for consideration for use in manned and unmanned maintainable spacecraft applications. Three specific areas were investigated:

- What state-of-art connector designs are available?
- What additional automatic, remote handling, latching devices are available?
- What new, low-insert force contacts are available?

5.1.1 Boeing Spacecraft Applications

The review of Boeing spacecraft applications disclosed that several connectors listed on the Inertial Upper Stage (IUS) preferred parts list offer design features which meet the objectives of the above study. The connectors are basically MIL-C-38999, series I, modified to meet the outgassing and lubrication requirements for spacecraft use.

No additional automatic latching devices were found, although the G & H Breech-Lok type of coupling may be suitable with remote latching. This coupling is used on MIL-C-38999, series IV connectors. This connector is in use as the Pod Umbilical connector for GSRS (Ground Support Rocket System).

There are three contact configurations now under evaluation which offer a 50 to 70% decrease in insertion force. These are: the Bendix brush contact, the Cannon tapered pin and the multi-wire mesh (Hypertac) contact from Industrial Electronic Hardware Corporation. These contacts configurations will be able to be used in existing connector designs.

There are some limited-use connectors (280-33019, Interstage and 280-33020, Umbilical) uniquely utilized which possess features that could meet basic design requirements needed for spacecraft interfaces.

Connectors meeting 40 M series specification requirements are performing satisfactorily in manned and unmanned spacecraft programs. This type of connectors should be considered for the near-term spacecraft of the 1980's. Some modifications may be required for the coupling mechanisms to meet the unmanned, automatic assembly in space requirements for the late and/or post 1980 to 1990 decade. The study by Martin Marietta (reference 9), includes data on the 40 M series connectors. Connector specialists working on the Inertial Upper Stage (IUS) spacecraft also agree that the 40 M series connectors are satisfactory for present and future spacecraft.

5.2 Connector Concepts

Cables and connectors for high voltage, high power, high current systems must be developed for minumum weight and efficient cooling. To achieve these design requirements, connectors must have low contact resistance to minimize resistance and heating and have good connections with simple lightweight mechanisms. The connector design concept is impacted by the spacecraft design. For instance, much of the spacecraft structure will be assembled in space. When the power, control, and communication wire and cables are made an integral part of the structure then all wires and cables on structural members must be mated simultaneously with the mating of structural members. Therefore, it is mandatory that the structural mating mechanism be partially engaged and the connector plate assemblies in alignment, prior to connector engagement.

When a multiple connector plate is used, the connector plate shall mate before the individual connectors. Then as the spacecraft and connector plate become fully engaged, the connectors will also be fully engaged.

5.2.1 Connector Modifications

Military specified connectors listed in MIL-C-38999, Series I, modified to meet the outgassing and lubrication requirements for space use, are adequate for most space hardware with power levels to 25 kilowatts. For spacecraft using large spacecraft structures technology (LSST) some new connector technology is needed to meet the requirements for higher power (50 kw to 2.5 MW), higher voltages (200 to 10,000 volts), and currents to 750 amperes. In some cases shielded connectors are required to suppress EMI from module to module within the power utilization equipment. Thus the need for new connector concepts.

5.2.2 Avionics Requirement For On-Orbit Assembly

Speical emphasis will be given to determine the capability of the spacecraft avionics equipment to meet the requirements involved in rendezvous and docking. Several modes of operation may be considered. These include using the Space-Ground Link Subsystem (SGLS), the Global Position System (GPS) satellites, and a completely self-contained on-board system. It was assumed that one goal of NASA would be to have a system that could not be interfered with and which would not radiate detectable signals except those preprogrammed to be transmitted.

The maximum range for rendezvous is set by the spacecraft three sigma error that is permitted by specification. A typical requirement is assumed for a spacecraft with a three sigma position and velocity injection accuracies of ±92 N. Mi. and ±78 FPS (both RSS values).

The study of each concept examined and its requirements to perform the rendezvous and docking mission is depicted in figure 5.2.2-1.

Radar systems of the type used in the Apollo program are very suitable to meet the mission requirements. Lasars, radar systems, and the microelectronic cameras recently introduced on the market were examined for close-in (200 ft or less) operation.

The following combination was selected:

- Apollo type rendezvous radar
- Microelectronic camera system for close-in relative attitude and position determination
- A mechanical "Soft Dock" first contact system
- A mechanical means of drawing the docking vehicle into a "Hard Dock" final position

The close-in docking system used in these six degree of freedom simulations is shown in figure 5.2.2-2

Requirements were established for the Rendezvous Radar and the Close-in Camera system. These are shown in figures 5.2.2-3 and 5.2.2-4.

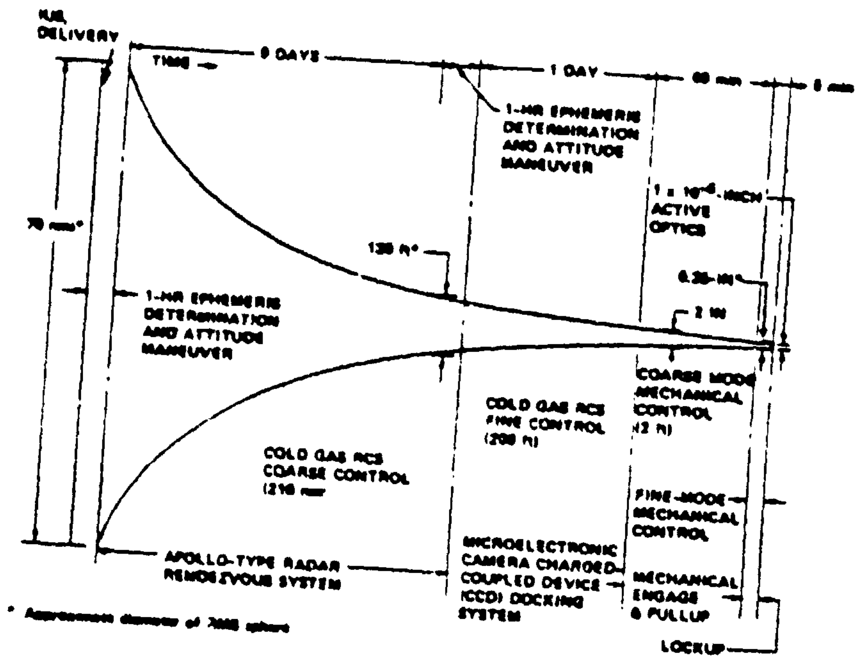


Figure 5.2.2-1: Docking Operation Schematic

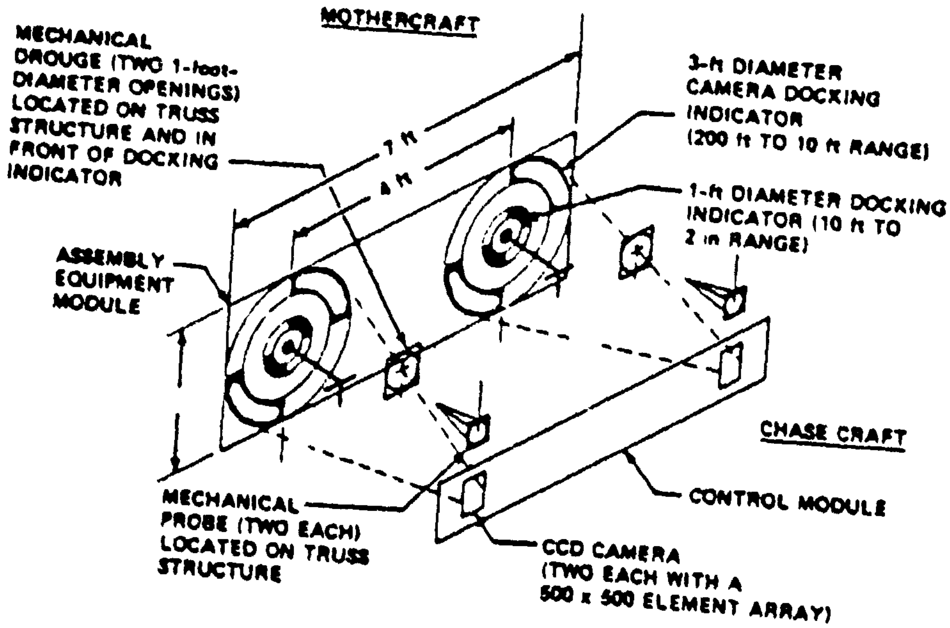


Figure 5.2.2-2: Micro-Electronic Camera System

- Range of radar system operation = 210 nmi to 200 ft
- Baseline data rate = 1 output/sec
  - Vary data rate to extreme of 1 output/min to show the effect on power required

  Let σ_R = 500 ft at 210 nmi

  σ_R = 50 ft at 200 ft

  σ_R = 0.5 ft/sec at 210 nmi

  $\sigma_{\dot{R}}$ = 0.1 ft/sec at 200 ft

  $\sigma_{\text{look angle}}$ = 0.7 deg at 210 nmi

  $\sigma_{\text{look angle}}$ = 0.5 deg at 200 ft

  $\sigma_{\text{look angle rate}}$ = 0.5 deg/sec at 210 nmi

  $\sigma_{\text{look angle rate}}$ = 0.2 deg/sec at 200 ft
- Key point:

  These requirements are based on Apollo rendezvous radar capabilities and are also compatible with handover requirements of the CCD camera docking system.

Figure 5.2.2-3: Rendezvous Radar System Requirements

Accuracy of pitch, yaw, roll angles as a function of distance.

	Distance (ft)	3σ
Required accuracy	200 1	0.5 deg 0.3 deg
Range information	200 1	8 ft 0.1 ft
Lateral information	200 1	5 ft 0.1 ft

Data transmission
0.050 sec each channel

Note: These requirements are based on results from a six-degree-of-freedom docking simulation with math model of CCD camera system. Camera jitter and pixel quantization errors are included in the simulation. Jitter includes the effects of vehicle structural flexibility.

Figure 5.2.2-4: CCD Accuracy Requirements

5.2.3 Connector Plate

An exploded view of the connector plate with alignment pins and the spacecraft structure docking probe-and drouge is shown in figure 5.2.3-1. It is necessary that the chase spacecraft connector plate be free floating so the alignment pins can insert properly. The "free float" plate would consist of movements in all three planes. There should be some spring loading for shock absorption during final mating. This should be limited to ± 5 millimeters movement in the X & Y planes. The spring loading in the Z-plane will keep constant pressure on the mating connectors.

Some high voltage, high current connectors will not require a connector plate. Due to their physical size and spacing they will be connected individually as the chase craft is docked to the mother craft. The connector plate will not require a latching device, the spring loading will hold the two plates together. The plates will contain feed-through type connectors that can be mated on both sides. All cables mounted on the structures will be connected to the connector plates.

Each connector on the chase spacecraft will be keyed and have limited movement to prevent pin damage.

5.2.4 Design Concepts

The connector design concepts described below must be mechanically and electrically analyzed for applicability on a spacecraft. These concepts only partially agree with NASA and military connector specifications and standards. Indeed, if they were made a part of a new, automatically assembled in space, spacecraft design, new specifications and controls would be necessary.

Concept number one is a multi-pin, cone-shaped connector. A diagram of the connector insert and shell is shown in figure 5.2.4-1.

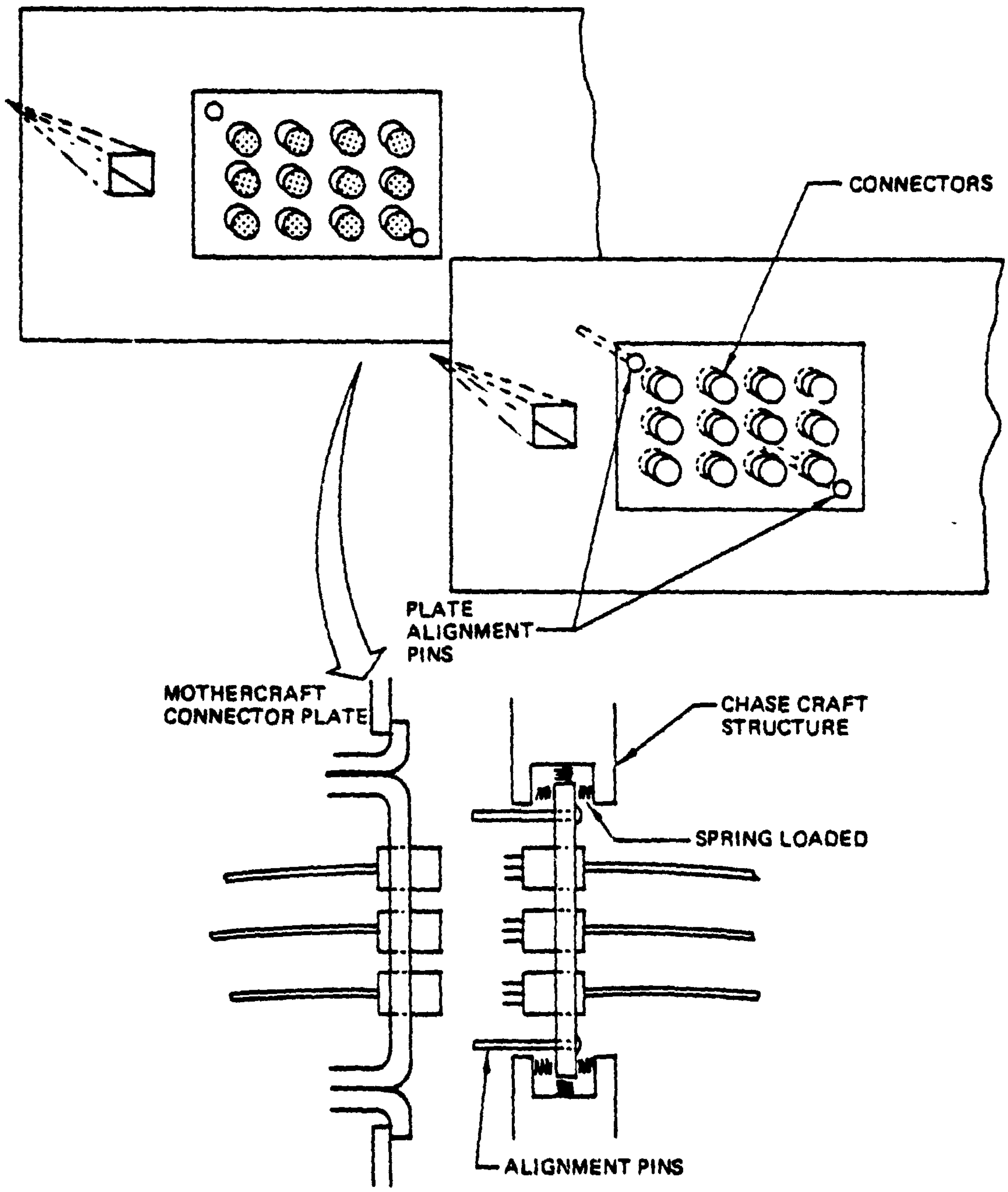


Figure 5.2.3-1: Connector Plate Alignment

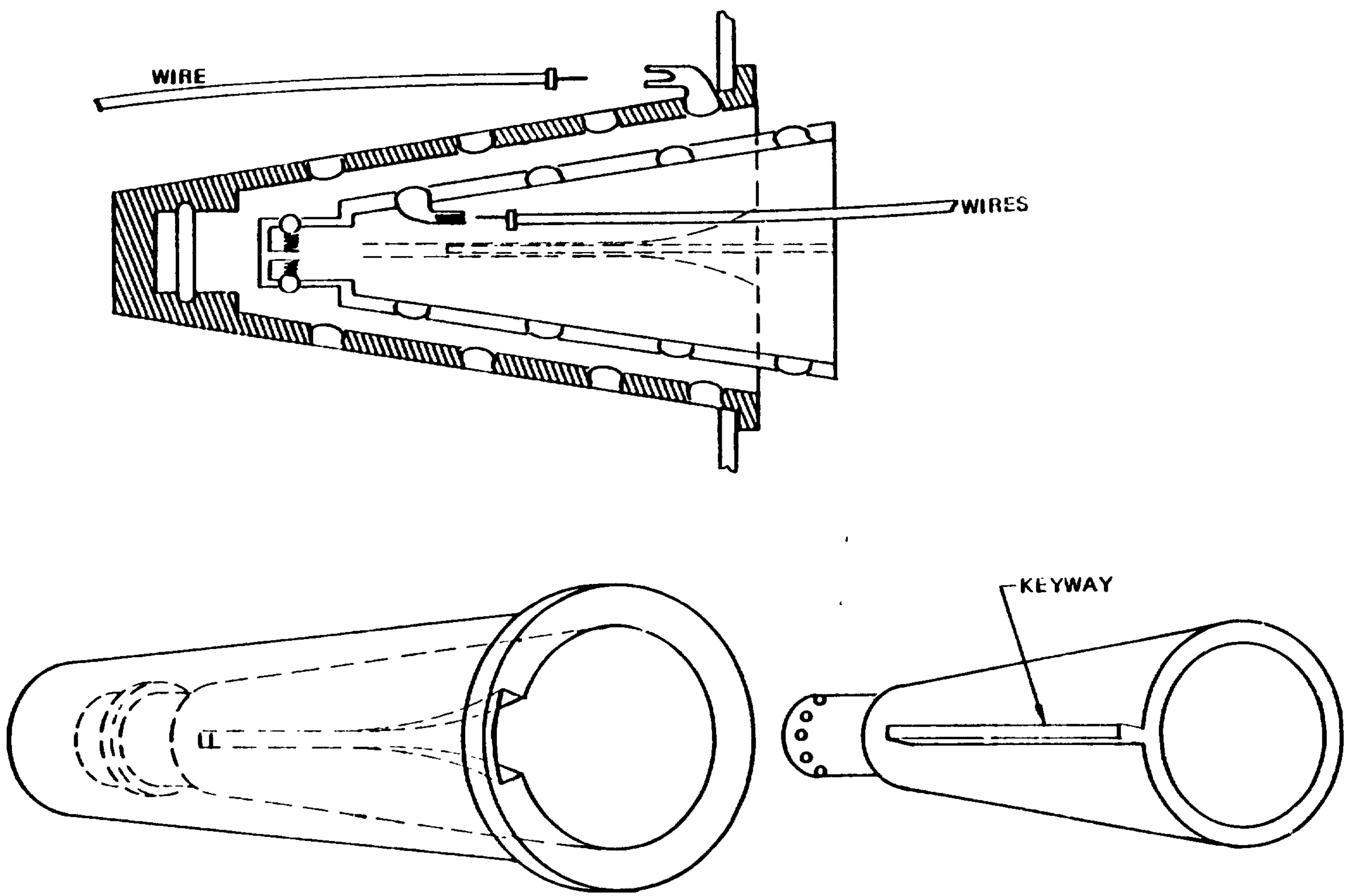


Figure 5.2.4-1: Multi-Pin Cone Connector

This unit is self aligning and may be connected/disconnected easily. An alignment slot on the side of the insert allows the insert to rotate up to $\pm 20^{o}$. The latching mechanism is similar to snap-lock fittings presently used and may be attached to the end of the insert end as shown, or made part of a snap-lock ring on the large shell of the connector. All wires will be attached on the inside via insert pins to the connector contacts.

This same cone configuration could be used for single conductor coaxial configurtions. In that case the coupling ring would require a spanner washer to apply positive pressure to the insert cone thourghout the life of the connector. A special tool would be required to fully mate the high voltage coaxial cone type connectors. Two American manufactures make connectors of the design shown in figure 5.2.4-2

A high voltage, high power, single rectangular bar connection concept is shown in figures 5.2.4-3 and 5.2.4-4. Guide fingers are used to align the connector into the proper mating snap locks. An automatic clamping device would be released onto two surfaces after the bars are in place. This unit could be connected/disconnected with a special tool.

A special tool for the circular connector is shown in figure 5.2.4-5. This tool turns at two speeds, 0.1 turn per sec. for snap lock connectors, and 1 turn per sec. for the high voltage coaxial connectors.

5.2.5 Concept Evaluation

The connector design concepts are evaluated in Table 5.2.5-1 with respect to impact on spacecraft design and EVA by astronauts or mechanisms.

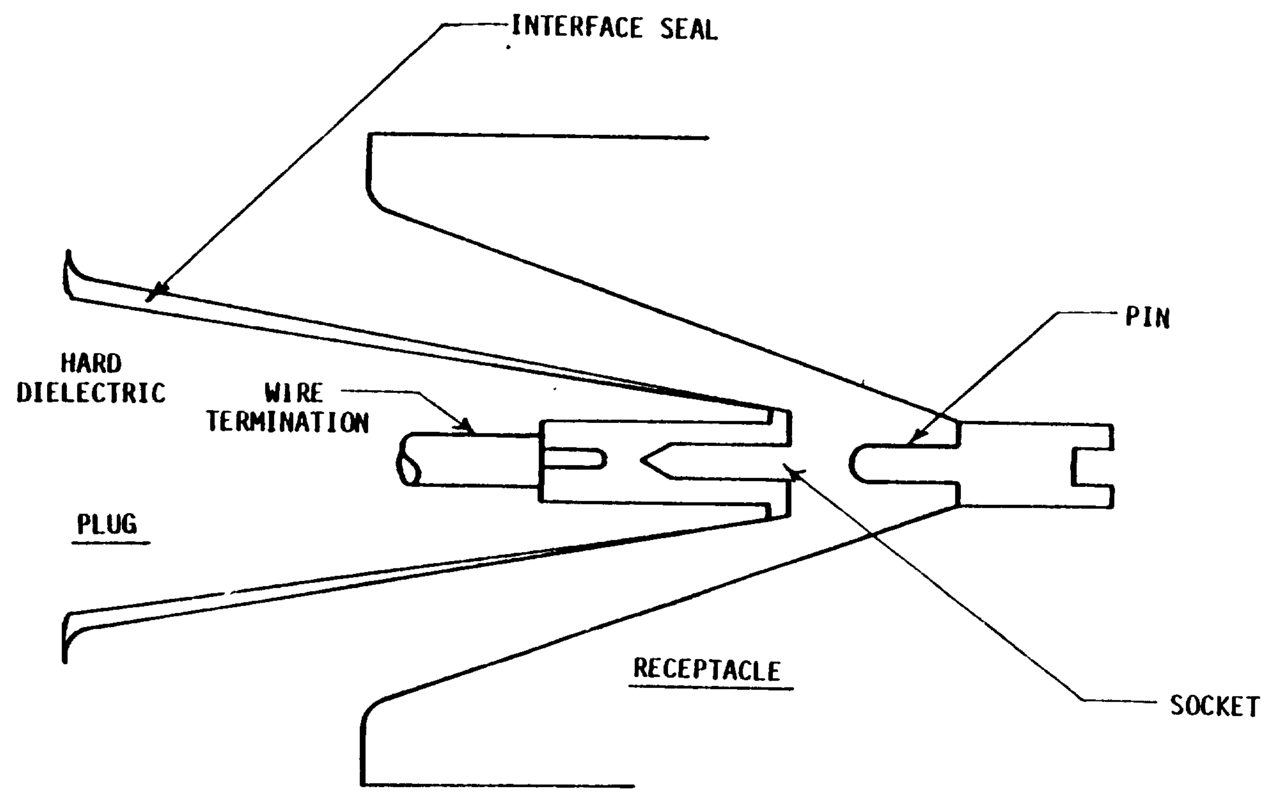


Figure 5.2.4-2: HV Cone Connector

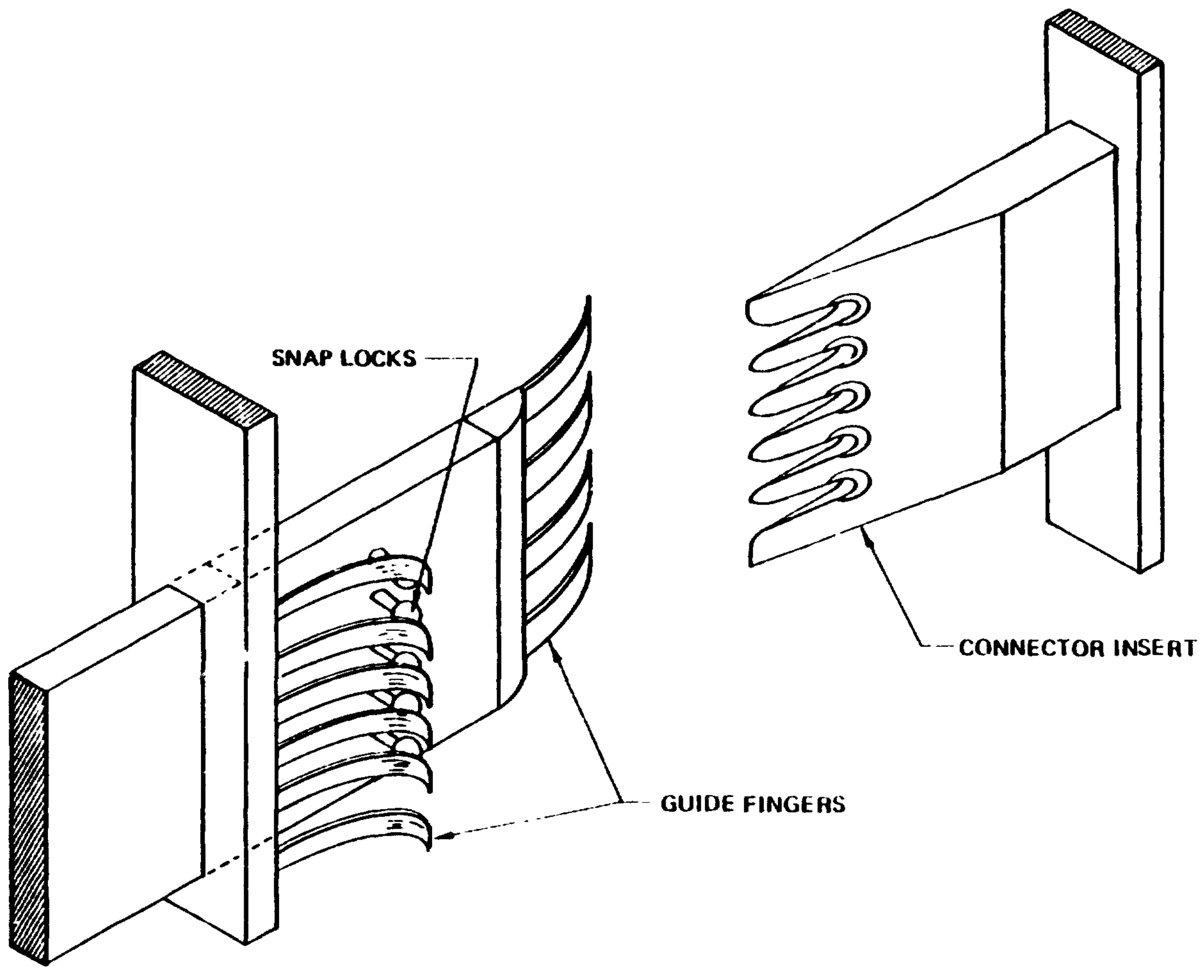


Figure 5.2.4-3: Rectangular Bar Connection

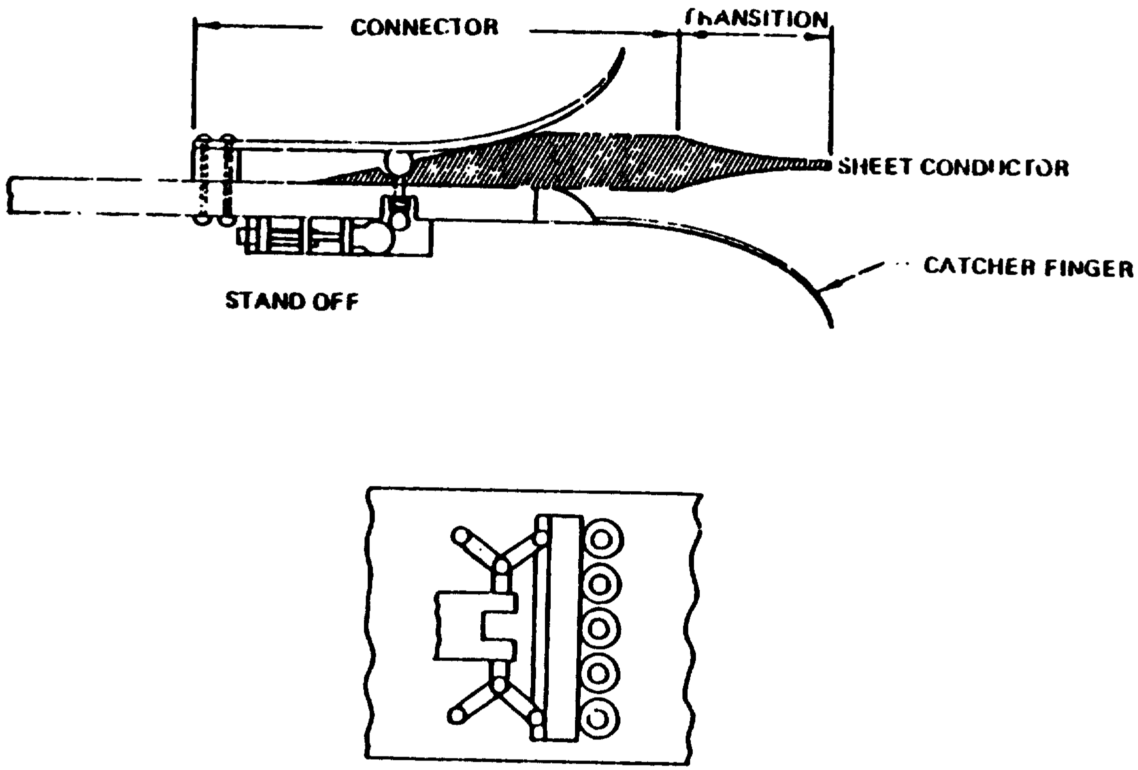


Figure 5.2.4-4 Sheet Metal Connector Detail

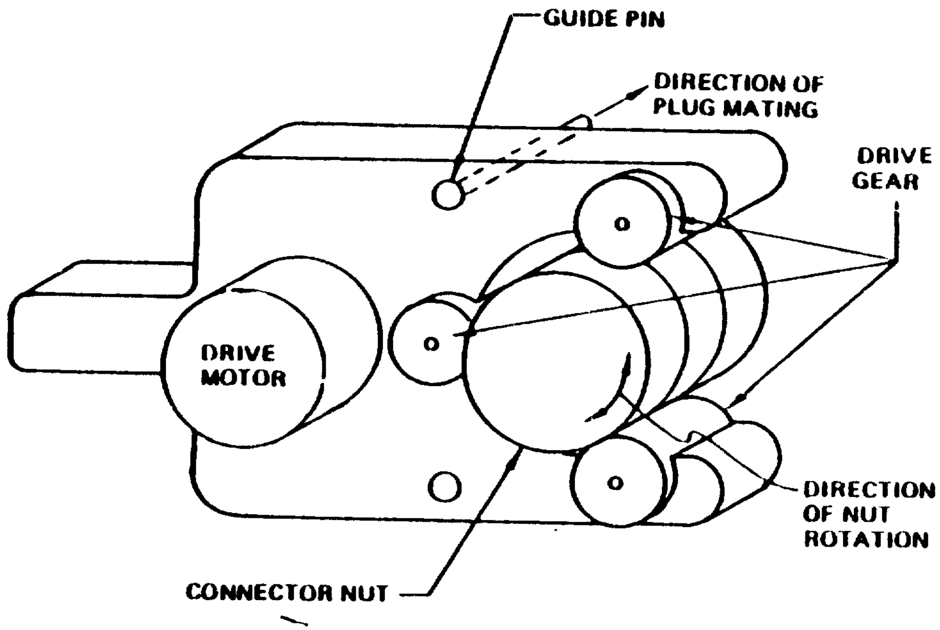


Figure 5.2.4-5 Connector Latching Tool

TABLE 5.2.5-1: CONNECTOR CONCEPT EVALUATION

Concept Figure	Voltage, Volts	Connector Current, Amperes	Concept Evaluation Advantages	Disadvantages
9	0-300V	0-150A	·Lowest Cost. ·Lightest weight. ·Many standard Mil-C-38999 connectors per plate. ·All connectors aligned and connected simultaneously. ·Connectors on one plate.	·Difficult to replace single connector. ·Single pin fault requires replacement of wire. ·Possible grounding and/or shell connection problems.
10	0-300V	0-15A	·Outgassing better than standard Mil-C-38999 connectors. ·Recommended for space use over Mil-C-38999.	·Increases cost/connector.
11	0-30kV	0-5A	·Simple connection. ·Lightweight. ·Low cost. ·Easily replaced by astronaut.	·Poor vibration/shock capability. ·Limited use. ·Good chance for arcing.
12	0-300V	0-25A	·Excellent alignment. ·Easily manufactured.	·Wire integral with the connector. ·Higher cost than Mil-C-38999. ·Heavier than Mil-C-38999. ·Redundant wiring required.

(Continued)

TABLE 5.2.5-1: CONNECTOR CONCEPT EVALUATION (Continued)

Concept Figure	Voltage, Volts	Connector Current, Amperes	Concept Evaluation Advantages	Disadvantages
13	0-150kV	0-500A	·Excellent alignment. ·Excellent shielding. ·Reliable - long life over 100,000 hours. ·Recommended for high voltage circuits where shielding is mandatory.	·High cost. ·Single conductor and shield. ·Suitable only for high voltage.
14	0-100kV	100-1500 Ampere	·High current application. ·Lightweight. ·Easily aligned. ·Low cost. ·Reliable - long life.	·Latching mechanism to be developed. ·Guide finger could act as points on high voltage circuits.

Connector concepts are shown in the following figures:

Figure 5.2.2-5	-	Connector plate alignment
Figure 5.2.5-1	-	MIL-C-38999 vented connectors
Figure 5.2.5-2	-	Single contact equipment connector
Figure 5.2.4-1	-	Multipin cone connector
Figure 5.2.4-2	-	HV cone connector
Figure 5.2.4-3	-	Rectangular bar connection

In summary, the design conecpts that should be further investigated are as follows:

Figure 5.2.2-5	-	Alignment blocks using Figure connectors are recommended for all low voltage applications
Figure 5.2.4-2	-	Single cone connectors are recommended for high voltage high current application where shielding is required.
Figure 5.2.4-3		Bar connectors are recommended for high current and high voltage applica- between power lines on adjacent solar panels.

5.3 Termination Methods

For the lower power ranges, the available connector technology can be utilized. For the large power ranges several power connector concepts were analyzed for, bus "bars" (rectangular bar conductors). Since photovoltaic systems tend to be large in area due to their low power density, several types of conductors could be used on a single system. Figure 5.3-1 illustrates a system in which power is collected from solar array sections with cable conductors. These power lines connect to a central feeder which could be a thin sheet or rectangular bar (figure 5.3-2). This determination is dependent upon central feeder transmission power (current/voltage) and length. The central feeders may then be connected to a main distribution line

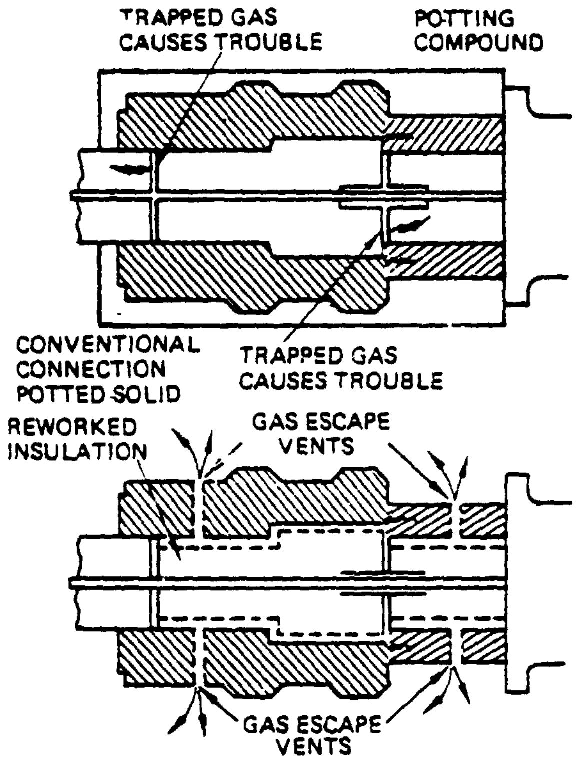


Figure 5.2.5-1: M.L-C-38999 Vented Connector

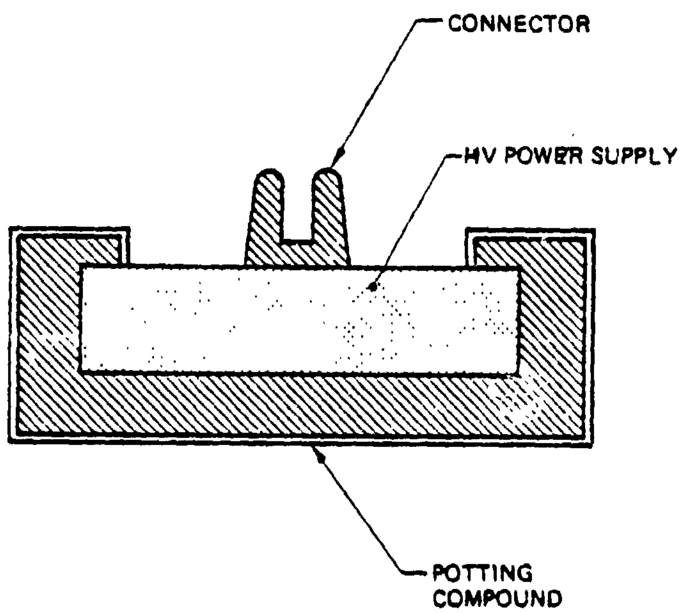


Figure 5.2.5-2: Single Contact Equipment Connector

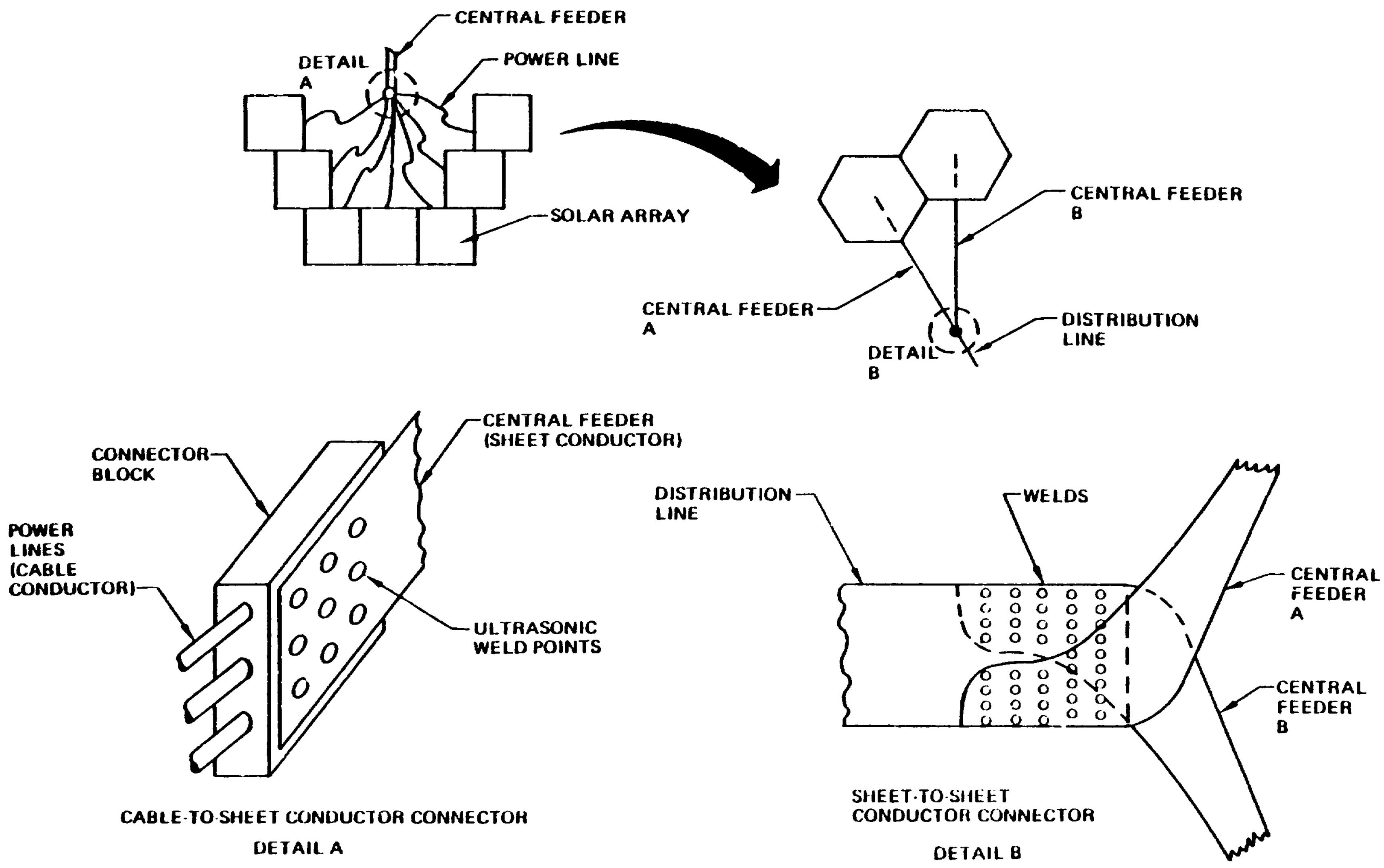


Figure 5.3-1: High Power Distribution System Terminations

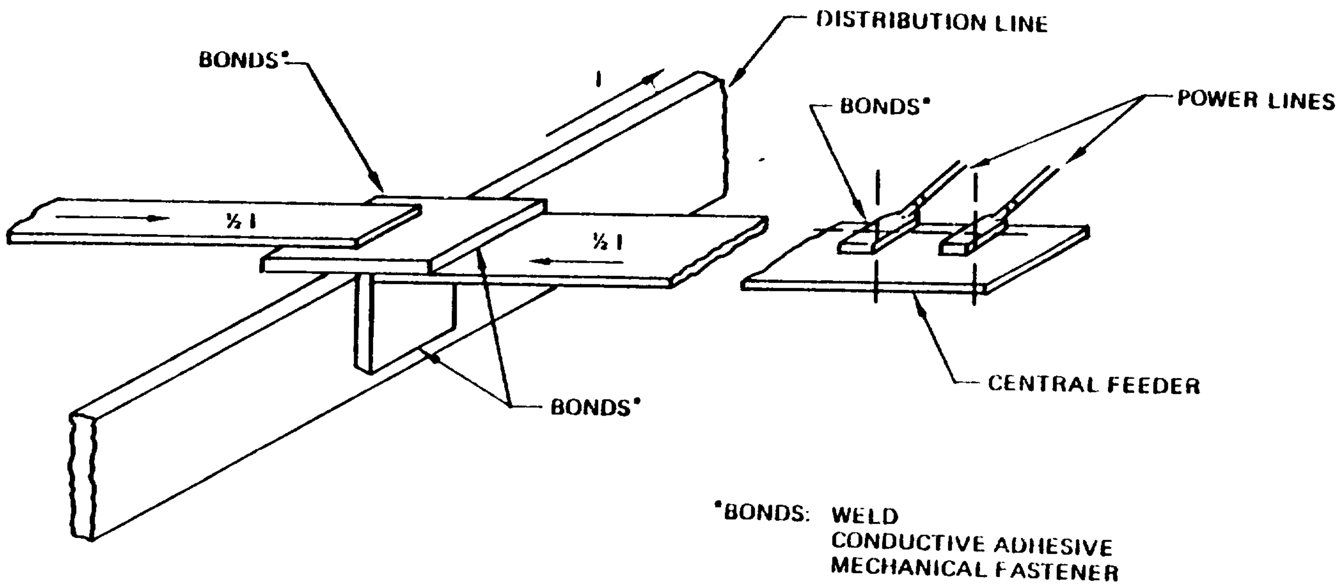


Figure 5.3-2: Rectangular "Bar" Conductor Termination Concepts

to supply some load, to be defined.

The connection of power lines, central feeders, and distribution line must be defined as to the solar array design involved, including insulation required for supports, manufacturing techniques, and joining techniques. For instance, the joining techniques used for a rectangular bar could involve a mechanical fastener (nuts/bolts, clamp, socket, etc.) whereas a thin foil connector may require some bonding operation (conductive adhesives, welding, etc.).

6.0 ENVIRONMENTAL CONSTRAINTS

Since the equipment will be operating in the space environment (hard vacuum, particulate radiation, plasma, solar insolance, zero-G ultraviolet, and meteoroids) its operational function should be compatible with all the conditions encountered. For safety reasons and time management considerations, the crew members should limit their EVA time during the deployment and assembly of the spacecraft.

6.1 Plasma and Particulate

The first objective of these studies is to quantitatively determine the amount of debris that will be collected by the high voltage circuit aboard a spacecraft and the plasma interaction on the high voltage system. The second objectives is to determine the effects of these environmental disturbances on the operation and life of the high voltage circuits.

Various environmental impacts significantly influence the performance of the power system of spacecraft. Although solar winds, plasma, and particulate debris differ in their mass, energy, and charge, they interact with the spacecraft in a similar manner. That is, plasma and particulates both have an electric charge; in addition, the particulates have mechanical interaction. This interaction with the power system results in effects such as:

- o arcing and surface damage,
- o power loss,
- o short circuiting of electrical elements,
- o degradation of paints and surface finishing.

These effects present serious problems for large spacecraft power system design due to:

- o High voltage cables and connectors exposed to the environment.
- o Excessive dielectric surface on the solar arrays

- Low density materials in the spacecraft structure
- Requirement for 10-20 years of service

These problems were investigated using the most recent relevant literature.

6.1.1 Plasma Problems

Extensive work has been done on spacecraft charging due to ambient plasma and solar winds, (references 18, 19, and 20). To date, most of the work has been done for small spacecraft. Out of the available analyses on large spacecraft, very few tackled the concepts for developing techniques for reducing plasma impact problems. This presents a challenging, interesting problem. The exposed cables, conductors, and connectors carry large currents which produce significant magnetic fields, in addition to the electric fields due to high voltage between conductors. In some cases, fields may be utilized in controlling the impacts of charged particles. This is achieved by selecting the proper configuration and design of conductors and cables for the spacecraft. As an illustration (reference 21) figure 6.1.1-1, shows the influence of the magnetic field of a conductor carrying a current in a moving plasma. The inertial force of the accelerated plasma or a gravitation field force a charge separation mode at the plasma boundary. Thus plasma is confined in the magnetic field around the conductor. Lee Parker (reference 22) had implemented this concept in his analyses for a solar power staellite. He considered a configuration where two cylinderical bus lines carry the current from a 2X20 km solar array as shown in figure 6.1.1-2. His analyses had indicated that electrons up to several Mev in energy are prevented from reaching a large fraction of the array surface. In the vicinity of the 100 kiloamp currents near the lower end of the array, the width of the protected portion of the array is 270 meters within which the life time of the solar cells would be prolonged by a possible factor of 5. Miller et. al. (reference 23) had conducted similar analyses, but considered a more complicated configuration for the current sources and buses for a section of the solar array (figure 6.1.1-3).

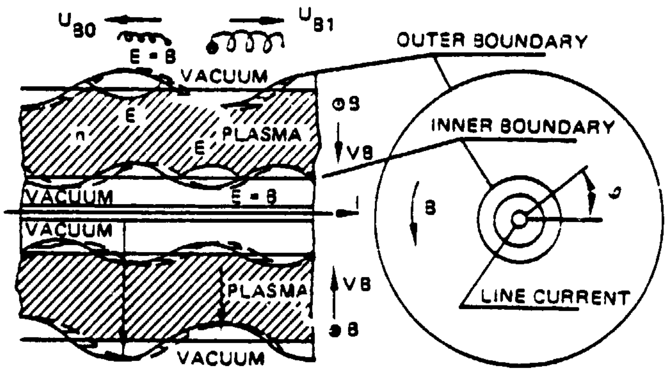


Figure 6.1.1-1: Cylindrical Shell of Plasma Confined in the Magnetic Field from an Axial Line Current.

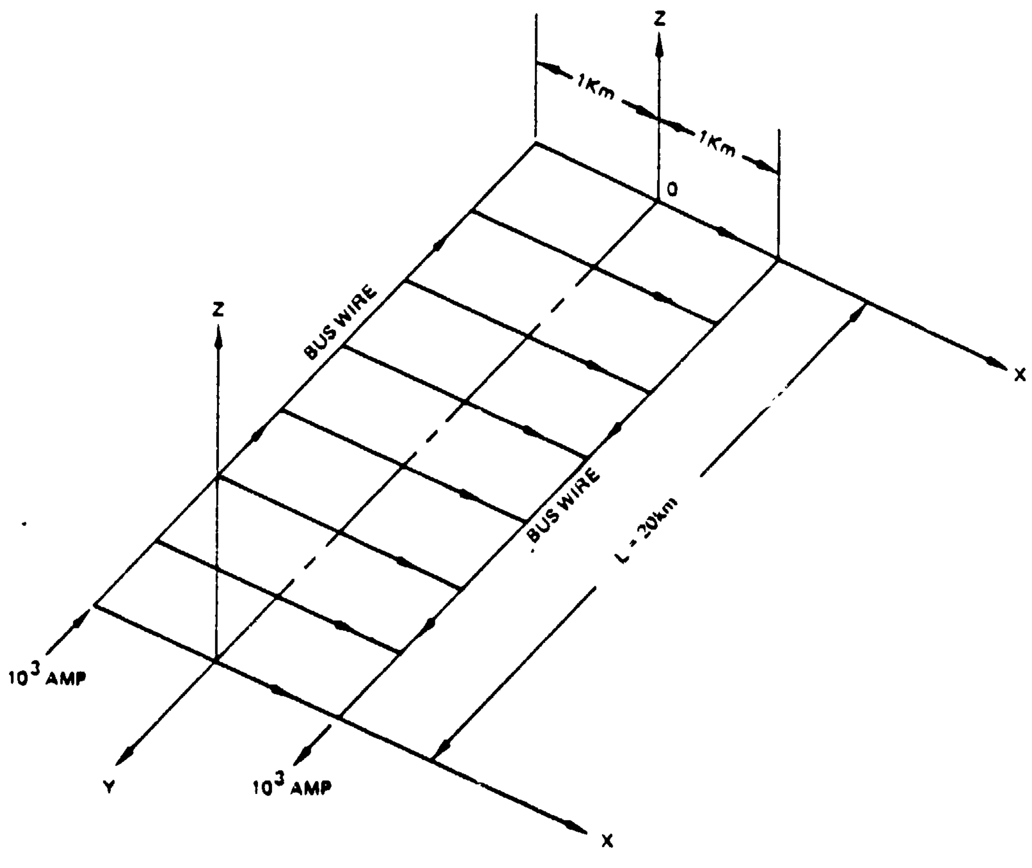


Figure 6.1.1-2: Current Distribution in Solar Array Panel Analytical Model

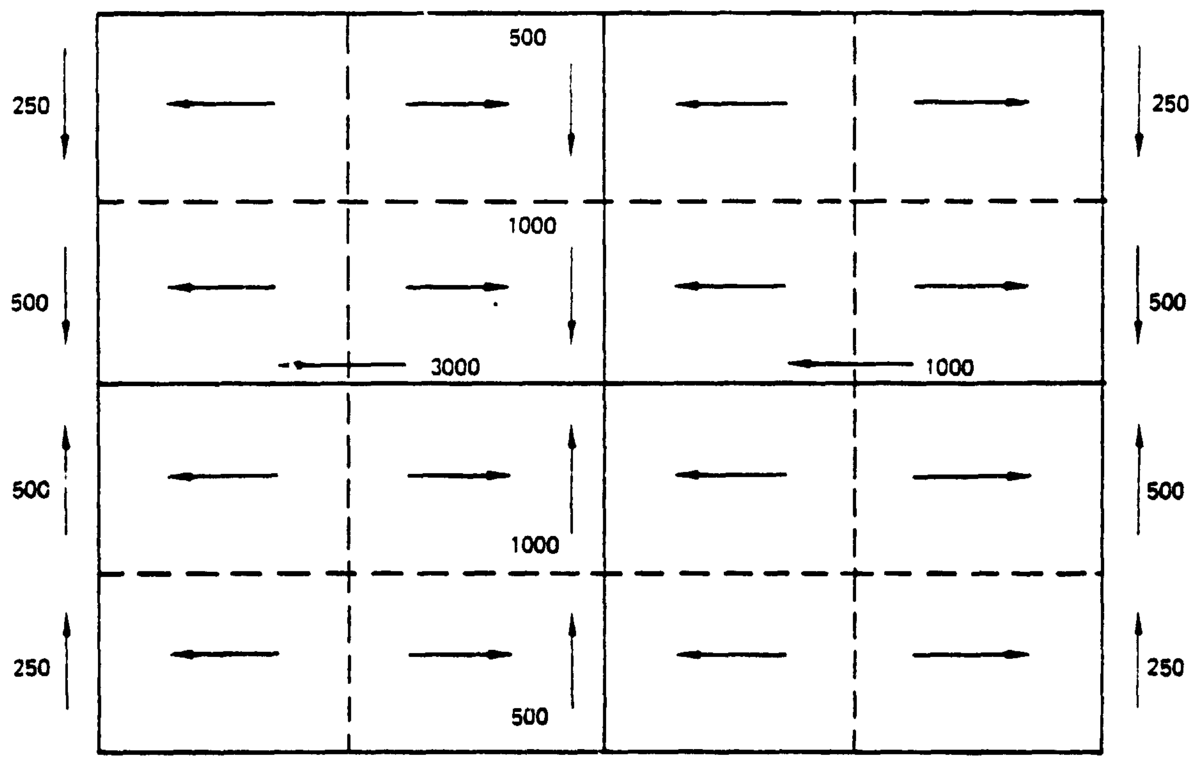


Figure 6.1.1-3: SPS Current Flow

The magnetic field due to this current distribution was determined over the array. Their analyses had indicated that for a normally incident plasma, most of the panel is not screened from electrons with energy greater than 1 Kev except along the crossed midlines. This indicated that the presented arrangement may promote differential charging and electron avalanche; for protons the cutoff energies are 1/1836 of these for electrons, hence most of the panel is not protected against protons with energy greater than 1 eV. Thus the analyses of Parker and Miller et. al. indicate the complexities of the shielding problem. Unlimited numbers of configurations and designs for the electric conductors, cables, and connectors may be implemented. Each has its shielding effects with varied advantages and disadvantages; more analyses are being pursued to find the optimum configuration and design for the conductors, cables, or connectors. In figure 6.1.1-4, a preliminary analyses on flat conducting bus lines instead of cylindrical ones is presented.

6.1.2 Particulate Problems

No serious consideration has been given for the impact of particualte debris on spacecraft. This may mislead designers who might assume that the impact of particulate debris is insignificant. Analyses have indicated, however, that the impacts of particulate debris on large spacecraft are appreciable. Problems are caused by debris, such as cosmic dust, striking the spacecraft at speeds greater than about 5 km/s. The impact of such a particle upon the spacecraft surface causes a shock wave to propagate from the collision interface compressing and heating the effected areas. The spacecraft surface at the collision is melted and the particulate is mostly vaporized, causing cratering and fragmentation in the spacecraft surface. The experimental work of Gault and Heitowit (reference 24) showed that an aluminum projectile with velocity of 6.25 km/s and impacting upon basalt surfaces induced ejection of mass approximately 370 times the mass of the projectile. Dohnanyi and Marcus (reference 25 and 26) had indicated, however, that a large range of experimental results on ejection generated from impacts into rocks are approximated by

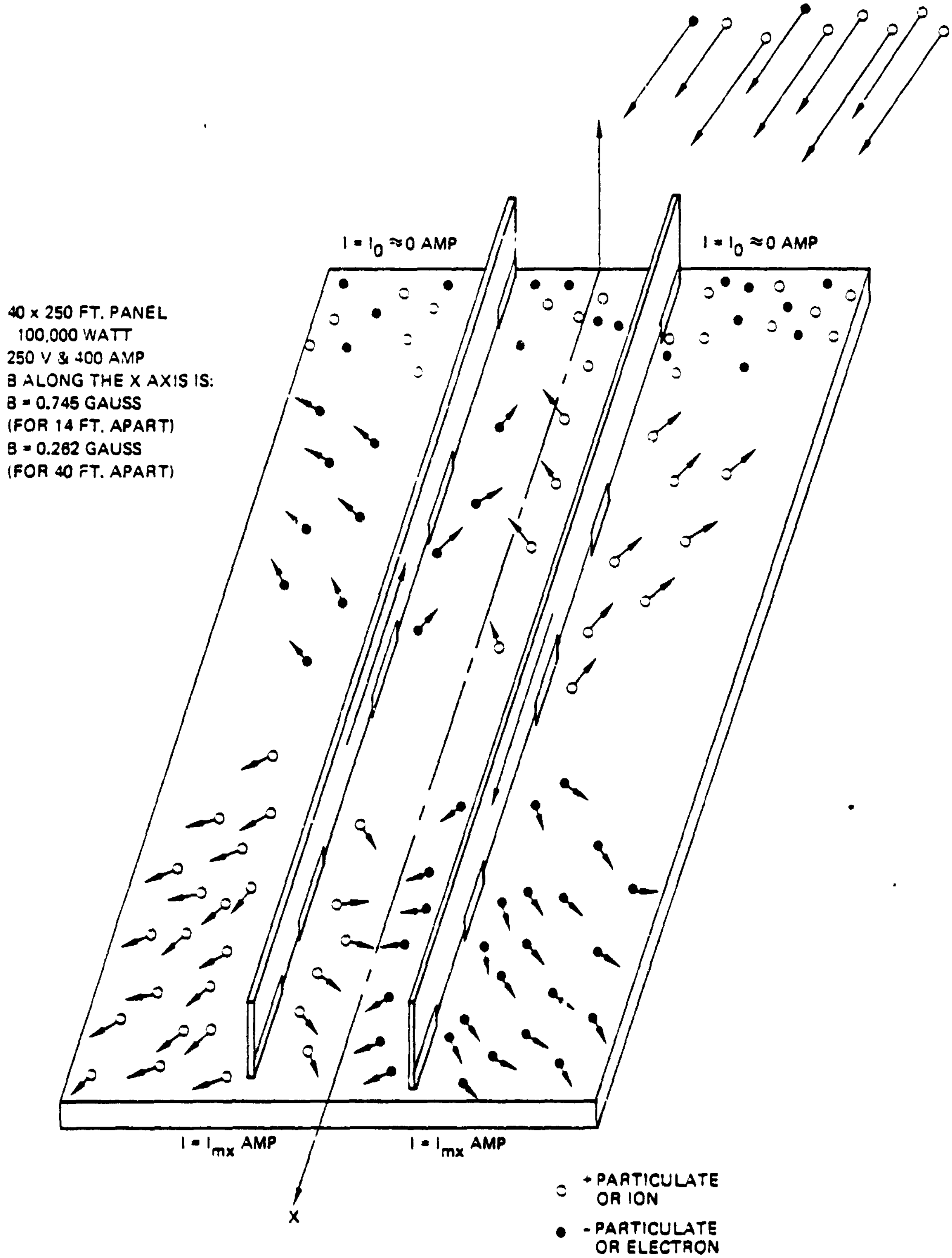


Figure 6.1.1-4: Schematic Presentation of Particulates — Solar Panel Interaction Under the Influence of the Magnetic Field Generated by the Flat Bus Lines

$M_e = 5v^2 M_p k$ where M_e and M_p are the masses of ejection and projectile, respectively; v is the velocity of the impacting particle in km/s, and k is a normalization constant of 1 s^2/km^2. Thus, for a speed of 6.25 km/s, the above equation yields about half the amount suggested by Gault and Heitowist. Therefore, applying this formula yields a conservative estimate of the ejection that is generated as a result of cosmic dust impacting upon the spacecraft.

Using the following values:

- o Cosmic dust average speed = 8 km/s (reference 27)
- o Cumulative flux value = $3.3 \times 10^{-3} m^{-2} sec^{-1} (1\pi sr)^{-1}$, (reference 27)
- o NASA recommendation for micrometeoroid mass distribution (reference 28)
- o Solar panel area = 1 km^2

The above formula yields a value of 430 gm/day of ejected mass. This is almost 10^4 times more than the ejection produced by continuous substorm proton flux (reference 24), indicating that the impacts of particulate debris are serious and require development of techniques to minimize their impacts.

6.1.3 Environmental Effects on Wire Routing

Plasma and debris cause problems for high voltage lines, that is, open lines with over 300 volts between conductors. Plasma, being a semi-conducting medium, causes a minute current flow between the conductors, resulting in a power loss which increases with increased voltage and increased flux.

Debris, whether from space or from the spacecraft, will collect on high voltage conductors as shown in figures 6.1.3-1 and 6.1.3-2. Debris can be collected on the high voltage conductors by 1) direct collision of the space debris with the conductor,

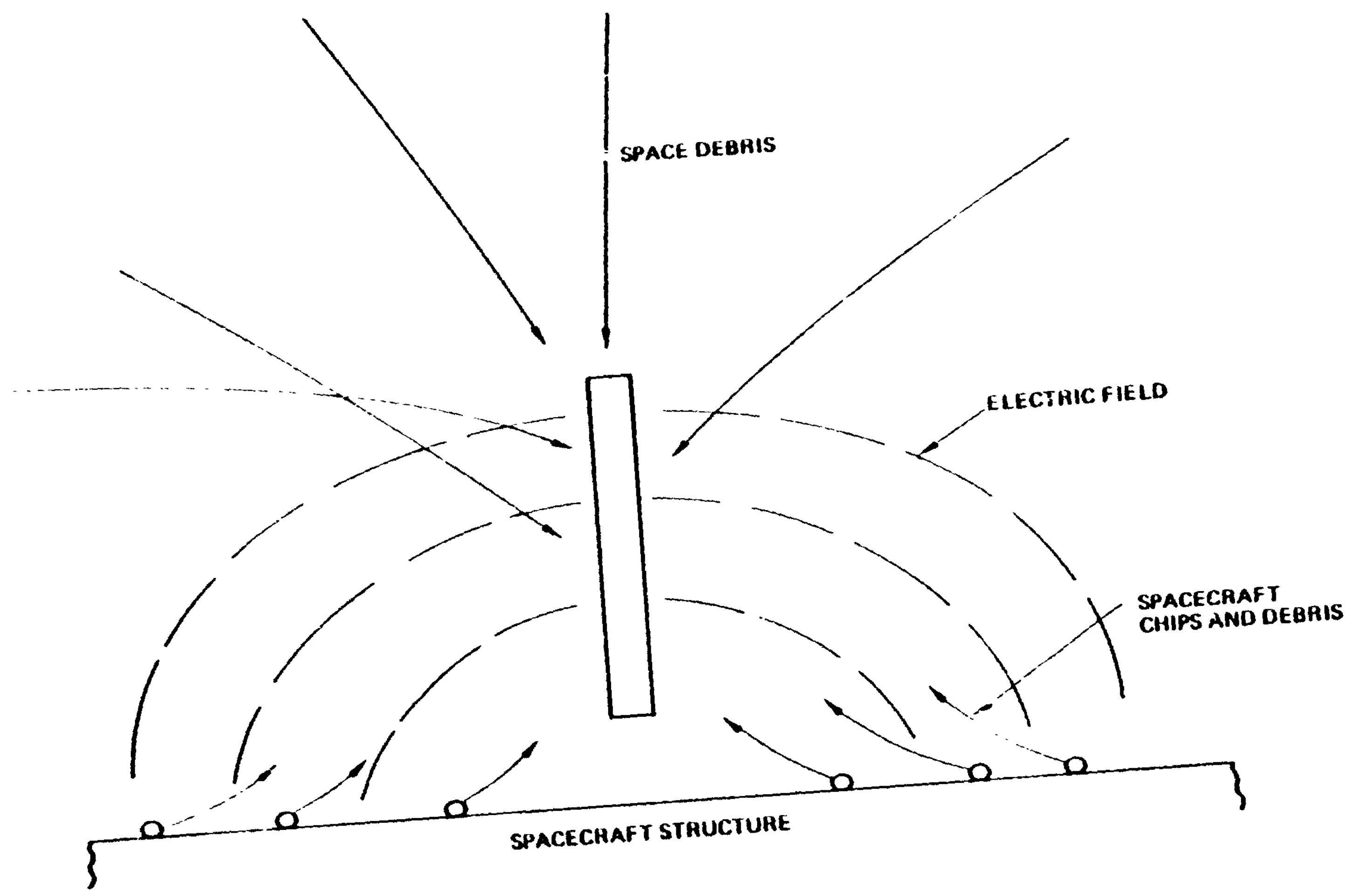


Figure 6.1.3-1: Particulate Collection to a High Voltage Conductor

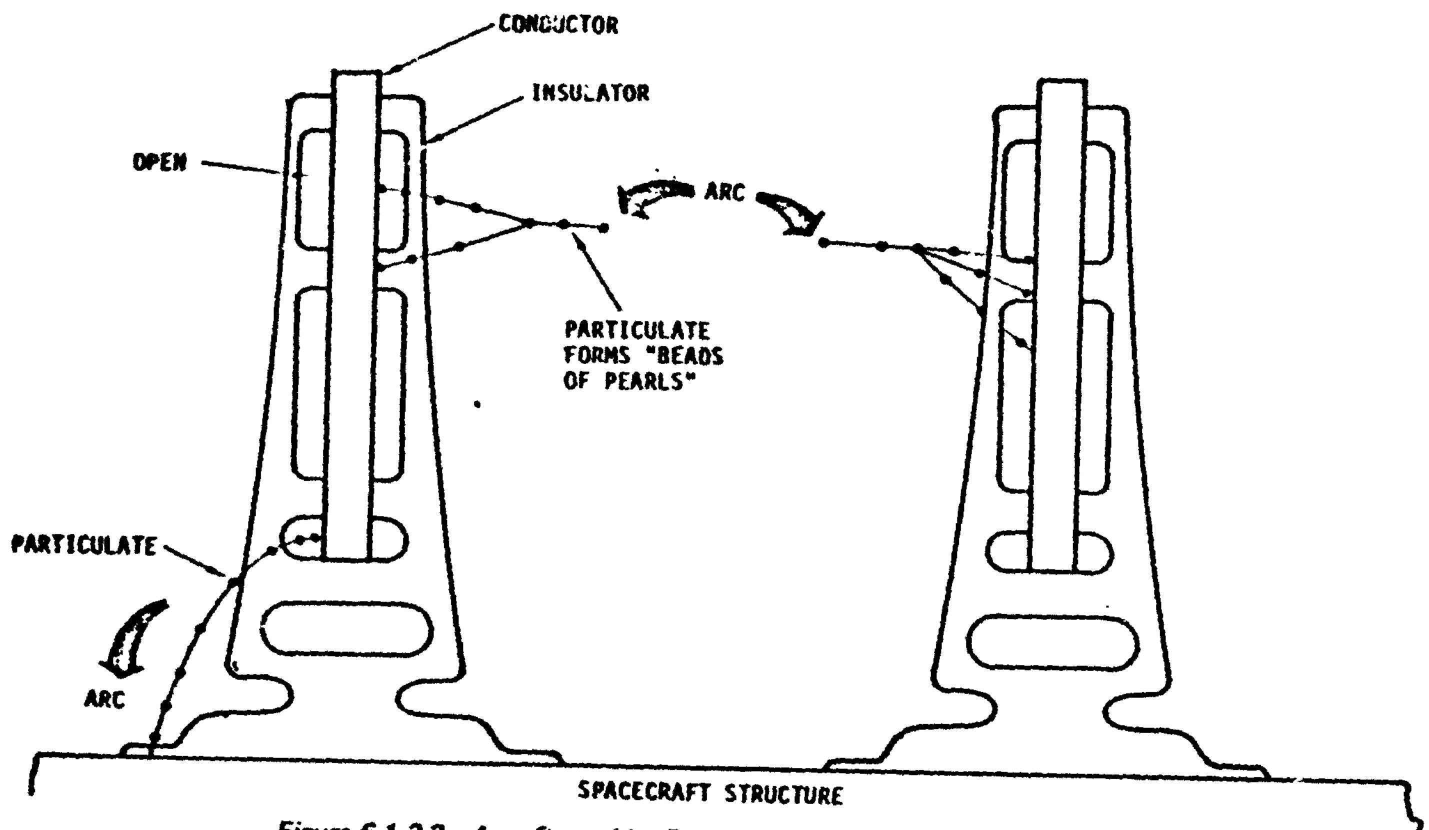


Figure 6.1.3-2: Arcs Caused by Particulate Bridge Between Conductors

or 2) electromagnetic field forcing the debris on the spacecraft surface to the conductor as shown in figure 6.1.3-1.

Once the debris have entered the field of the conductor, they will become charged and polarized to form a seta (hair like) growth on the conductors. Some of these charged particles will form bridges or "strings-of-pearls" between high voltage and low voltage or ground planes as shown in figure 6.1.3-2. The field stress at the ends of the seta is very high, so any plasma in that region can and will initiate an arc between the conductor and ground plane on another conductor, initiated at the highly stressed seta. It is important to shield the conductors and high voltage lines from debris whenever possible.

The question always arises about the benefit of insulated surfaced conductors. The debris will attach to the insulated surface. For thinly insulated surfaces, less than 0.5 mm thick, the field will be greater than the dielectric breakdown strength of the insulation. Therefore, an insulation breakdown would be inevitable.

Shielding can best be done with:

1) Coaxial conductors, with the high voltage conductor inside,
2) Side by side or layered conductors to produce zero field effects,
3) Heavily insulated conductors.

The layered conductors, as shown in sketch A, have the overall advantage since the external magnetic field is neutralized, they are inexpensive, and they are easily placed on the spacecraft. Insulated conductors will have many pin holes and the insulation will be subject to arcs since it will be exposed to the environment. Coaxial configurations are expensive and much more difficult to join in space.

6.2 Gases and Contaminants in the Environment

Corona and voltage breakdown of a gas are affected by temperature, charged particle radiation, ultraviolet irradiation, and particulate (space or spacecraft debris) contamination. Other contributing factors include electrode materials, electrode shape and finish, type of insulation, outgassing, electrical field stresses, electrode spacing, and applied voltage.

Contaminants can be foreign gasses, dust particles, oxides, and salts. For example, helium used for leak detection, if entrapped, reduces the breakdown voltage. If mechanical or electrical stressing should cause the insulation to crack internally, the crack can fill with helium rather than nitrogen or other pressurizing gas. When the helium partial pressure is between 13.3 N/M^2 (1×10^{-1} torr) and 2.66×10^3 N/m^2 (20 torr) it can ionize, generating partial discharges within the void, and possibly result in insulation failure.

Dust particles can contribute to local stress by making a plane surface ha· several points on the electrode. Likewise, oxides and salts, which are present in the air during the assembly, storage, transportation and launch, can deposit on the surface of the insulation. Eventually these deposits lower the electronic work function of the metal electrode permitting voltage breakdown at much lower voltage.

The vacuum in deep space is a good insulator because it contains few charge carriers and the mean-free path far exceeds the gap between electrodes. The volume within most spacecraft, however, contains gas atoms and charged particles from these sources:

a. Outgassing from nearby materials
b. Sublimation of nearby surfaces
c. Trapped air within the components
d. Gas-filled voids in insulation
e. Spacecraft leakage gases

As a result the interelectrode gap can appraoch the minimum corona initiation or voltage breakdown from the high-vacuum side of the Paschen Law curve. These gas sources can be minimized during design and manufacturing. However, they must be quantified so that the pressure/temperature profile in critical gaps can be calculated to permit design of a corona-free system. Breakdown voltage for selected pure metals in uncontaminated vacuum is shown in Table 6.2-1. The breakdown voltage between contaminated electrodes may be as much as an order of magnitude voltage lower than that between the pure metals and/or alloys.

Table 6.2-1: Vacuum Breakdown of Metal Electrodes

Material (polished surfaces)	Vacuum Breakdown Voltages (kilovolts for 1 mm gap)
Steel	122
Stainless Steel	120
Nickel	96
Aluminum	41
Copper	37

Unpressurized high-voltage conductors in the presence of outgassing from a spacecraft or even from high-voltage insulation are also vulnerable to corona, partial discharges, and arc-over. If there is outgassing, then the turn-on of high voltage circuits must be delayed until the pressure due to outgassing has decreased to a satisfactory low level.

6.2.1 Space Environment

Most satellites operate in space where the gas pressure (less than 10^{-4} N/m^2 makes the theoretical dielectric strength of the volume of gas greater than 3×10^5 volts/cm, a value 10 times the dielectric strength of dry air at sea level. This is because there are few carriers and the mean free path exceeds the gap length between closely-spaced electrodes.

The spacecraft and its modules and circuits slowly outgas after the spacecraft is inserted into orbit as the spacecraft is boosted from its launch pad into orbit the external pressure decreases rapidly from Earth sea-level pressure ($1.013 \times 10^5 N/m^2$) to less than $1 \times 10^{-5} N/m^2$. This pressure change takes only a few minutes, but the pressure next to the outer surface and inside the spacecraft will remain at a higher value throughout the life of the spacecraft, due to the outgassing of electrical, structural, and other materials.

6.2.2 Internal Gas Pressure

During boost, gas escapes rapidly from the spacecraft interior as the altitude changes from sea level to thirty kilometers into space. During this interval, there is little differential pressure between the inside and outside of the spacecraft. As the spacecraft continues on its projected course, its internal pressure will be greater than the external pressure due to the slow outgassing, through small orifices, tubes and cracks, of gases entrapped in electrical and thermal insulations and structural materials.

For a spacecraft which has no outgassing products, the flow of gas can be calculated by the Clausing equation (reference 28).

$$C = 3.638 \, A_k \left(\frac{T}{m}\right)^{1/2} \quad \text{cc/s}$$

Where: C is the flow conductance for the orifices
A_k is the area of the k orifices in square centimeters
T is the internal temperature in $^{\circ}K$

m is the mass of the gas molecule in grams

This equation is used for estimating the flow of gas from chamber to chamber in a multiple chamber vacuum system.

Scialdone calculated and measured spacecraft compartment and equipment outgassing rates (reference 29). He shows that the depressurization time constant τ , which is the time for the pressure to decrease to 36% of its initial pressure, is V/C, where V is the volume in cubic centimeters and the conductance C is:

$C = 1/4 \bar{\nu} Ak$

$\bar{\nu} = (8 kT/\pi m)^{1/2}$ cm/s ($\bar{\nu}$ is the molecular flow speed and k is the Boltzman constant).

Gases such as air and nitrogen have a time constant of about 0.4 second when bled through a 1-cm^2 opening in a 10-cm radius steel sphere (reference 36). NASA experience (reference 29) has shown that a 0.1 sec. time constant insures adequate outgassing around high voltage circuits.

These equations work well with known outgassing port sizes, spacecraft volumes, and non-gassing parts. However, most spacecraft have thermally insulating coatings, fiberous insulation, electrically insulated parts, semi-shielded and electromagnetically-shielded boxes, and boxes within modules. In addition, compressed gases for orbit keeping jets are carried and released. With these many gas sources, it is often better to qualify the design by testing the completely assembled spacecraft in a thermal vacuum chamber than to measure the real internal and external spacecraft pressures and the outgassing rate.

6.2.3 Outgassing Through Insulation

Some high voltage conductors and connectors may be inside temperature-controlled compartments. The outgassing time for these compartments will be much longer than for the unpressurized compartments.

To show the effect of thermal blankets on outgassing an experiment was made in a space chamber. In the experiment, a thermal blanket having 100 layers of superinsulation was placed across the center of a vacuum chamber. Gas flowed through the intersticial spaces in the insulation. During the first fifteen minutes of pump-down, the pressure in the chamber dropped from sea level to 10 N/m^2, with the gauge on the thermally insulated side of the chamber following the pump pressure within 5%. As the pump pressure dropped further, the pressure at the insulated side of the chamber decreased very slowly, (figure 6.2.3-1).

In the Skylab Apollo Telescope Mount, the outgassing area was about one square centimeter per liter volume, the value recommended for adequate spacecraft outgassing when high voltage experiments or equipment are on board. The resulting pressures, reported in References 31 and 32, are summarized in figure 6.2.3-2.

Cuddihy and Moacanin, (reference 33) in measuring the outgassing rates of polyurethane foam, used for electrical/electronic insulation, found that the calculated value based on the reported permeation constant of the measured value agreed within a factor of 2. The diffusion coefficient, D, for a foam is calculated with the equation:

$$D = Pe\left(\frac{RT}{M}\right)\frac{P_o}{P}\left[\frac{1}{(1 - P/_{P_o})^{1/3}} + 1 + (1 - P/_{P_o})^{1/3}\right] \mathrm{cm}^3/\mathrm{s}$$

where:

P is in the foam density in g/cc
Pe is the permeation constant in $mm/s/cm^2/torr$
P_o is the density of the bulk polymer in g/cc
R is the gas constant

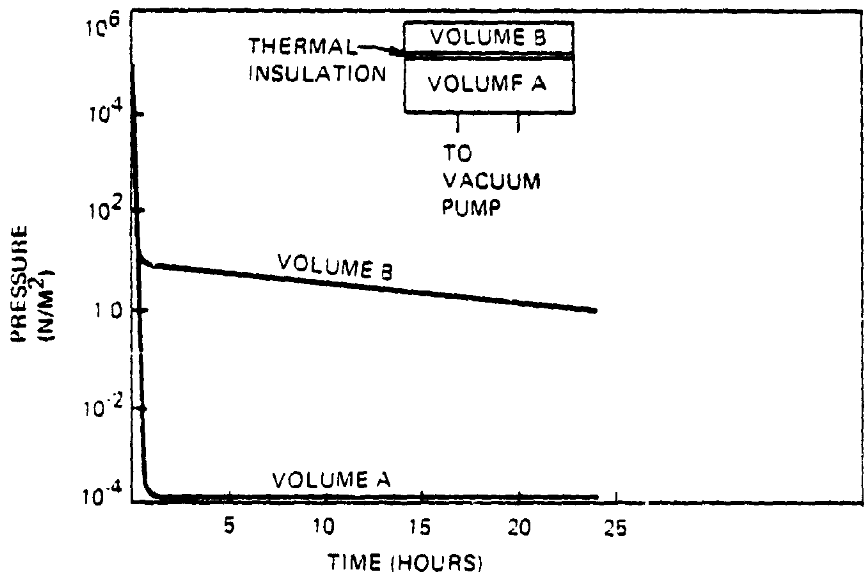


Figure 6.2.3-1: Effects of Thermal Insulation on Outgassing Rate

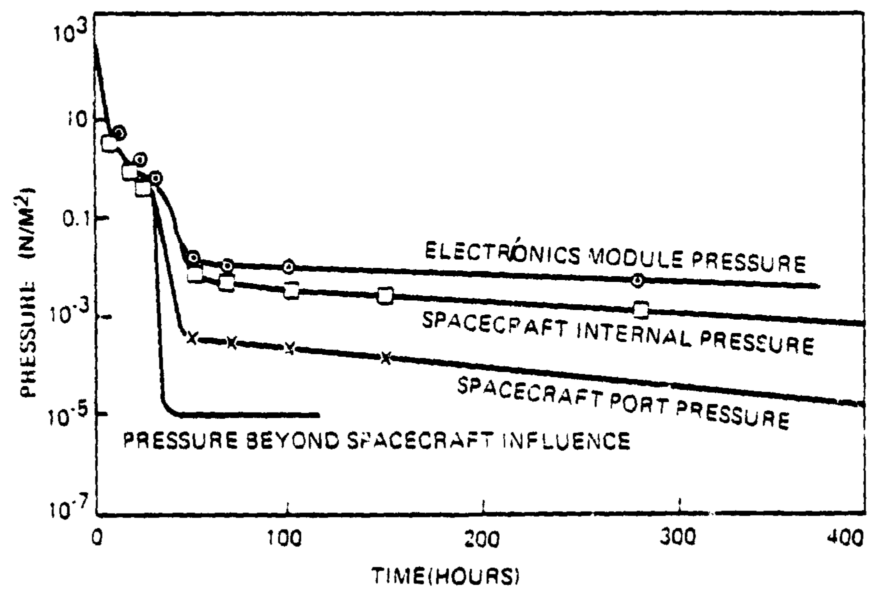


Figure 6.2.3-2: Gas Pressure Inside the Apollo Telescope Mount

They also found that for sufficiently long outgassing time, the weight loss of the gas, when plotted as a function of the thickness of the solid, A, eventually becomes linear with a slope of $(\pi/2A)^2D$.

The superinsulation in the Apollo Telescope Mount and polyurethane foam tests show that the spacecraft internal pressure may be significantly greater than the external pressure for several days after orbit insertion. Furthermore, outgassing products within a high-voltage module may keep the pressures much too high for safe, reliable operation of high voltage circuits, making it advisable to delay their TURN ON. Likewise, the outgassing products of the spacecraft and reaction control propellants increase the pressure in the vincinity of the spacecraft.

6.2.4 Astronaut Suit Outgassing And Safety

When a suited astronaut performs work outside the pressurized cabin, a <u>pressure dome</u> around the astronaut will result. The pressure near the suit surface will rapidly decrease to 1.33×10^{-2} N/m^2 at one meter distance from the astronaut and to $1.33 \times 10^{-4} N/m^2$ at ten meters distance. This implies that all high voltage equipment, circuits, and lines must be de-energized or grounded beofre the astronaut enters the area. Enough time must be programmed for the system pressure to decrease to an acceptable level before the equipment is turned on, otherwise voltage breakdown will occur (figure 6.2.4-1). The pressure level is a function of particulate and plasma conditions at the time of turn-on. Likewise, the astronaut and his equipment must be clear of the high voltage area.

6.2.5 Connectors And Degassing

High voltage connectors have given so much breakdown trouble that there is a tendency to not use them at all. All the junction points are soldered with the possible exception of very high voltage points. Even then rudimentary connectors such as that shown in

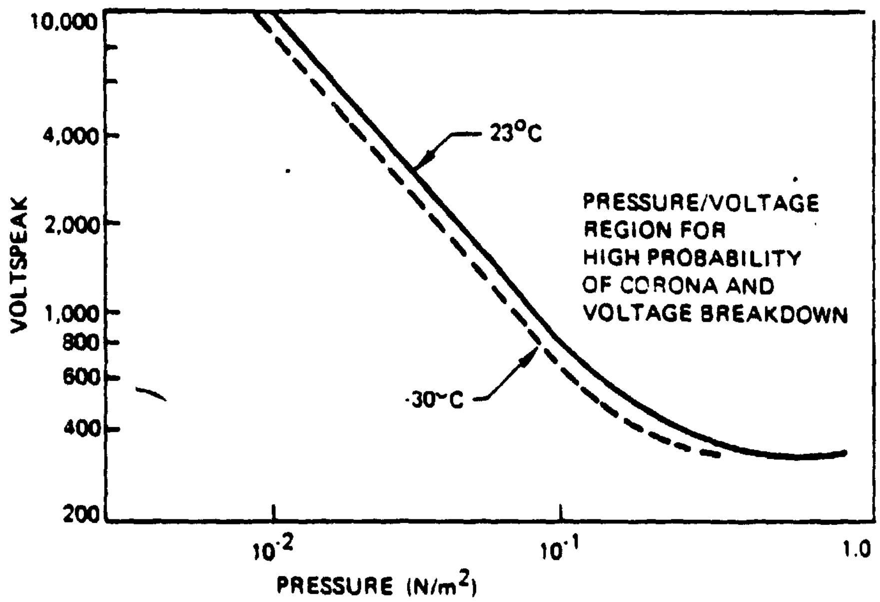


Figure 6.2.4-1: Paschen Curve for High Voltage Assemblies

figure 5.2.5-2 are used. The unit requiring 15 kilovolts rests directly on the rounded central stub where the voltage is developed, thus obviating the need for leads.

Of course, some equally successful designers do use connectors. The key to success is adequate venting. No gas can be allowed to be trapped in the connector body in such a way that it can leak out slowly. If this happens a critical pressure environment can result for the high voltage leads in the connector long after one would have expected all gas to have diffused into space. Some connectors are vented by drilling holes through the body and insulation to permit ready egress of the gas. An adequately vented connector is the bottom one shown in figure 5.2.5.-1.

Figure 5.2.5.-1 also shows a failed connector. This was a commercial connector that was simply potted after the cnnnection was made, with a nearly gas tight seal around the electrical leads. The gas slowly leaked out during the vacuum test. When the pressure around the leads became low enough a breakdown occurred. The arc-over drew large currents and eventually destroyed the connector.

6.3 High Temperatures

High temperatures can be very damaging to insulation. When insulation is exposed to an over-temperature (i.e., temperature above its rated temperature) it will:

a. Outgas and pressurize voids and volumes.
b. Become more conductive, which increases the probability of surface creepage and flashover.
c. Lose part of its integrity. The insulation dielectric strength decreases as temperature increases, creating cracking and treeing and more chance of breakdown or arcing.
d. Suffer degradation due to atomic oxygen which is more active than ozone. At temperatures above 450^{o}F ozone dissociates into atomic oxygen.

6.3.1 Hot Spots

High temperature in the area of the highest electric field, rather than the ambient temperature, increase the quantity of partial discharges. "Hot spots" increase the outgassing rate of the insulation, thereby generating voids, minute cracks, and small pressurized enclosed volumes which enchance partial discharges and arc-over.

These "hotspots" can be the result of contaminants on the surface of the electrodes or insulation. These contaminants are things like dust particles, oxides, and salts that are present in the air during assembly, storage, transportation, and launch. The breakdown voltage between contaminated electrodes may be as much as an order of magnitude lower in voltage than that between pure metals or alloys.

These "hotspots" on bare contaminated electrodes can become hot enough (500^{o}C) to cause thermionic emission, which would further enhance the discharge and lead to catastrophic breakdown sooner.

7.0 EXTRA VEHICULAR ACTIVITY (EVA)

An astronaut or crew member must be well versed in high voltage safety prior to working on or near high voltage or high power equipment and systems. To merely disconnec a high-voltage or high-power module is inadequate and could be dangerous. Conductors must be grounded when disconnected by either manual or automatic means and a visual indication of the grounding must be confirmed. The man must then wait at least 3 to 5 minutes before entering a compartment or touching the hardware to be assured all static and stored charges have been neutralized by the grounding action. Disconnecting low-voltage, high-current leads (greater than 32 volts) may result in as serious burns to the crew member as touching a high-voltage wire.

Any time a crew member enters a compartment or approaches a high-voltage or high-current distribution line or equipment such as a rotary joint or power breaker, he must be assured that that portion of the system is de-energized. Then, he should attach a "grounding wand" (shorting bar with flexible attachment cable) to the nearest grounding joint and touch the terminals of the lines or module to be serviced. Any sparking upon conta is an indication that the unit was charged and the grounding wand must be in place before further service by the crew member. Finally, before entering a compartment or cul-de-sac containing equipment it must be assumed that all the equipment may have a charge. Therefore, it is important that all discharge points be checked for discharge within the region. Then the crew member is ready to do maintenance, repair, and inspections.

7.1 High-Voltage High-Power Cables/Connectors

High-voltage, high-power cables and connectors on the solar arrays and to the electronic compartments will be capable of supplying several hundred amperes. These lines will be inspected by a crew member or by a remote video system. By whichever means of inspection, the crew member must inspect for connector mating and latching. This includes grounding connections as well as power connections.

Connectors must be designed to automatically latch when the solar array sections are mated. Disconnecting a solar array section requires much more effort. First, the connectors must be unlatched by either a hand-pull device operated by the crew member or by a mechanized unlatching device activated by the crew member. All small connectors assembled on a single board also must be unlatched prior to removal of the solar array array section. The crew member must be cognizant of all operations, grounding of high voltage lines before and after disconnect, and safety during the EVA.

7.2 Data and Communication Systems

. The only danger associated with data and communication lines is that they may be in the general vicinity of high-voltage or high-current lines. The capacitive coupling between the lines may induce high-voltage into the low-voltage circuits. Such voltages can damage sensitive low-voltage sensors and circuits. Therefore, all low-voltage equipment inputs and conductors must be shorted during maintanance/repair to safeguard crew members from lethal induced voltages. Connectors on the connector plates may be connected/disconnected with speical tools and/or manually operated latching devices designed for the connector systems.

Hand operated latching mechanisms for portable equipment are too bulky and heavy to install on a multiple connector plate. Circular connector or rectangular connectors with flexible leads and diamond-knurled clamping rings over center latches (suit case type) should be used. Connector spacing should be 3 to 5 cm apart, and staggered to eliminate cable interference between connectors.

Portable magnetic disconnect mechanisms can be designed and built to latch/unlatch large connectors and connector plates. Screw thread alignment is not recommended for large plates on connectors.

Each connector should have its own alignment mechanism. The alignment pin must have axial and angular orientation for ease of connect/disconnect by the crew members as shown in figure 5.2.4-2.

8.0 PERSONNEL PROTECTION

Safety of personnel shall be a prime consideration in the design, development, test, and operation of systems, subsystems, and equipment. Every effort shall be made during the early stages of design to obtain a high degree of inherent safety through the selection of appropriate design features, proven qualified components, sound engineering practices, and operating principles. Hazards which cannot be eliminated through early design shall be reduced by using appropriate safety, caution, and warning devices as part of the subsystem. When built-in protection is not technically feasible, technical manuals and operation instructions should clearly identify the situation.

8.1 Dangerous Conditions

Any circuit carrying a voltage considered dangerous must be suitably protected. Generally, voltage in excess of 32 volts is regarded as a dangerous voltage. Also, it is usually desirable to provide protection for lower voltages or other hazards where normal maintenance functions must be performed in crowded locations with limited access.

Current rather than voltage is the most important variable in establishing the criterion for shock intensity. Three factors that determine the severity of electrical shock are:

1. Quantity of current flowing through the body.
2. Path of current through the body.
3. Duration of time that the current flows through the body.

The voltage necessary to produce the fatal current is dependent upon the resistance of the body, contact conditions, and the path through the body. Probable effects of shock, for personnel of normal health, are shown below:

Current values (milliamperes)

AC 60 Hz	DC	Effects
0-1	0-4	Preception
1-4	4-15	Surprise
4-21	15-80	Reflex action
21-40	80-160	Muscular inhibition
40-100	160-300	Respiratory block
Over 100	Over 100	Usually fatal

While the current is the criterion that determines shock effect, the voltage on an exposed metal part is the only criterion that is practical for a designer to consider.

8.2 Dangerous Voltages

Equipment utilizing dangerous voltages shall be protected by means of a removable guard or barrier. All conductors, contacts, terminals and like devices having potentials between 70 and 500 volts shall also be protected by way of a cover provided with holes for test probes. In this case, terminal numbers should be placed on the protective cover. Assemblies operating at potentials greater than 500 volts shall be in an enclosure which is interlocked is such a way that that high voltage is always removed when the enclosure is opened. This does not require that necessary low voltages assemblies be excluded from these enclosures.

8.2.1 High Voltage Power Supplies

High Voltage supplies designed to deliver polarizing voltages to items which use less than 10 milliamperes shall be designed to minimize short circuit current output capability whenever this is consistant with the intended function of the supply. The energy storage in filter capacitors shall be a minimum for the required ripple level. A desirable type of supply is an r-f oscillator-rectifier supply which delivers no persistent current to a low resistance load.

8.2.2 Interlocks

Interlock switches shall be of the two-piece type, whereby the electrical circuit is broken by physical separation of the two parts. Bypass devices, when provided, shall be of a manually operated type and shall conform to one of the following:

a. Bypass devices of the switch type shall be such that returning the chassis to the operating position or closing the door, cover, or plate will automatically open the bypass switch and leave the interlock in position to function normally. The circuit must be turned on following an interlock closure.

b. Other types of bypass devices shall be equipped with a handle or otherwise insulated to provide maximum protection to personnel and shall be such that the equipment cannot be replaced or the door, cover, or plate closed without removing the device.

8.2.3 High Voltage Protection

When the operation or maintenance of equipment employing potentials in excess of 1000 volts could require that these voltages be measured, the equipment shall be provided with test points so that all high voltages can be measured at a relatively low potential level but in no case shall the potential exceed 100 volts to ground. This may be accomplished by the application of voltage dividers or other techniques such as the use of sfety-type panel meters and multipliers.

8.2.4 Warning Markings

All removable barriers or guards shall contain a warning to indicate the highest voltage which may be encountered.

All contacts, conductors, terminals, and like devices having potentials in excess of 500 volts shall be clearly maked "DANGER HIGH VOLTAGE (maximum voltage applicable) VOLTS." The letters shall be capitals, clearly legible, color white or aluminum with red background. (American Standard z351-1959). Covers or doors within units protecting components having dangerous voltages shall be clearly marked and labeled per MIL-STD-454A.

8.2.5 Protective Devices

The design shall incorporate methods to protect personnel from accidental contact with voltages in excess of 30 volts while operating a complete equipment. Means shall be provided so that power may be cut off while installing, replacing, or interchanging a complete equipment, assembly, or part thereof. Personnel shall be protected from capacitor-type discharges when changing fuses or tubes. The main power ON-OFF switch located on the equipment (clearly labeled as such) shall cut off all power to the equipment.

8.3 Safety

A potential personnel shock hazard results from the "recovery charge phenomenon," where a latent charge builds up in any capacitor-like structure such as a long conductor, after it has been discharged. Therefore, high voltage protective caps and cable or transmission line shorting taps having integral shorting contacts should be installed on all open high voltage connectors and cables when not in use. These protective caps and taps should be installed on each high voltage cable and equipment prior to shipment and should remain on until installation is complete.

The influence of the plasma on high voltage breakdown was evidenced in a test by Burrowbridge on the Orbiting Solar Observatory - Harvard College Observatory Spectroheliograph (Reference 34). During this test, a plasma exposure at an elevated pressure--10^{-3} torr, and an ion flux of 5×10^{11} ions/cm^2 -sec, which corresponds to an orbital ion density of approximately 10^6 particles/cm^3, produced arcing in the high voltage system. These arcing conditions were not duplicated in a plasma-free environment until a pressure of 10^{-2} torr was reached. This increases the number of available ions, which increases the collision cross-section and lowers the anode voltage required to produce and sustain an avalanche (gas breakdown). It would be reasonable to assume that this effect is operative at lower pressures, in which case a plasma environment could induce breakdown when the pressure was normally "too low" to sustain it.

9.0 REFERENCES

1. "Structural Attachments for Large Space Structures", NAS8-32654, MMC.

2. "Advanced Technology Laboratory Program for Large Space Structures", NAS1-14116, Rockwell, (Parts 1 and 2, May 1977; Part 4, March 1978).

3. "Large Space Structues, Configuration, Packaging and Responses Studies", NAS1-13967, Boeing, Sept. 1978.

4. "Structural Efficiency of Long Lightly Loaded Truss and Isogrid Columns for Space Application, "NASA Tech. Memo 78687, M.M. Mikulas, Jr., NASA-Langley, July 1978.

5. "Process Identification Study for Space Cured Composite Structures", NAS1-14887, LMSV, October 1977.

6. "Comparative Analysis/Evaluation for SPS Large Space Structures", NAS8-33062, Vought.

7. "Space Fabrication Demonstration System", NAS8-32472, GAC.

8. "Large Space Erectable Structures-Building Block Structures Study", NAS9-14914, Boeing, April 1977.

9. "Analytical Study of Electrical Disconnect System for Use on Manned and Unmanned Missions", NAS8-31971, MMC.

10. "Orbital Construction Demonstration Study", NAS9-14916.

11. "Satellite Power System (SPS) Concept Definition Study", NAS8-32475, Rockwell.

12. "Simplified Thermal Estimation Techniques for Large Space Structures", NAS1-13967, Boeing, October 1977.

13. "OAST System Technology Planning", Stanley R. Sadin, NASA Conf. Pub. 2058, Future Orbital Systems Technology Requirements 1978.

14. Ibid.

15. "Analytical Study of Electrical Disconnect System for Use on Manned and Unmanned Missions", NAS8-31971, MMC.

16. "Satellite Power System (SPS) Concept Definition Study", NAS8-32475, Rockwell.

17. "Power Management and Control for Space Systems", Robert C. Finke, Ira T. Myer, Fred F. Terdan and N. John Stevens, NASA Conference Publication 2058, Future Orbital Power Systems Technology Requirements, May 1978.

18. "Spacecraft Charging Technology", NASA Conference Publication 2071, AFGL-TR-79-0082 (1978).

19. "Proceedings of the Spacecraft Charging Technology Conference", NASA TMX-73537, AFGL-TR-0051 (1977).

20. "Spacecraft Charging by Magnetospheric Plasma", Progress in Astronautics and Aeronautics, Volume 47 (1975).

21. Bo Lehnert, "Dynamics of Charged Particles", North-Holland Publishing Company (1964).

22. Parker, L. and Oran, W. A., "Magnetic Shielding of Large High-Power-Satellite Solar Arrays Using Internal Currents", Spacecraft Cahrging Technology, NASA Conference Publication 2071, AFGL-TR-79-0082 (1978).

23. Miller, E., Fischbein W., Stauber M. and Suh P., "Environmental Interaction Implications for Large Space Systems", Spacecraft Charging Technology, NASA Conference Publication 2071, AFGL-TR-0082 (1978).

24. Gault, D. E. and Heitowit, E. D., "The Partition of Energy for Hypervelocity Impact Craters Formed in Rocks", Proc. 6th Hypervelocity Impact Symposium, 2, PP 419, Firestone Rubber Company, Cleveland, Ohio/USA (1963).

25. Dhonanyi, J. S., "Interplanetary Objects in Review: Statistics of their Masses and Dynamics," Icarus, 17, pp. 1-48 (1963).

26. Marcus, A. A., "Speculation on Mass Loss by Meteroroid Impact and Formation of Planets", Icarus, 11, pp. 76-87 (1969).

27. Fechtic, H., Grun, E. and Morfill, G., "Micormeteoroids with Ten Earth Radii," Planet Space Sci., Vol. 27, pp. 511 (1979).

28. West, G. S., Jr., "Space and PLanetary Environment Criteria Guidelines for Use in Space Vehicle Development", 1977 Revision, NASA Technical Memorandum 78119 (1977).

29. Dustman, S. and Lafferty, J. M., Scientific Foundations of Vacuum Technique, John Wiley and Sons, Inc., N. Y., N. Y. 3rd Printing, 1965, P. 93

30. Scialdone, J. J., "The Outgassing and Pressure in a Spacecraft", Proc. of Inst. of Environ. Sciences, Washington, D. C., 1974, P. 164.

31. Sutton, J. F., and Stern, J. E., "Spacecraft High Voltage Power Supply Construction", NASA Tech. Note TN D7948, Goddard Spaceflight Center, Greenbelt, Md., 1975, P. 39

32. Dunbar, W. G., "Skylab High Voltage Systems Corona Assessment", 11th Electrical/Electronics Insulation Conference, Chicago, Illinois, 1973.

33. Scannapeico, J. F., "The Effects of Outgassing Materials on Voltage Breakdown, Proceedings of the Second Workshop on Voltage Breakdown in Electronic Equipment at Low Air Pressure, NASA, Jet Propulsion Laboratory, Technical Report 33-447, 1969.

34. Cuddihy, E. F., and Moachanin J., "Outgassing Rates in Polymeric Foams", NASA, Jet Propulsion Laboratory, Technical Report 32-840.

35. Burrowbridge, D. R., "Testing of High Voltage Spacecraft In a Simulated Ionospheric Plasma, NASA, Goddard Space Flight Center, Greenbelt, Md., March, 1969.