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CRITICAL ENVIRONMENTAL INTERACTIONS DOCUMENTATION AND THEIR APPLICATION TO FUTURE AIR FORCE MISSION CONCEPTS

N.J. STEVENS
M.E. KIRKPATRICK
C.S. UNDERWOOD
R.P. STILLWELL

TRW, INCORPORATED
ELECTRONICS AND DEFENSE SECTOR
ONE SPACE PARK
REDONDO BEACH, CA 90278

Edited by:

CAPT C.J. FRUSHON
1LT M.J. GIGER
W.N. HALL

AIR FORCE GEOPHYSICS LABORATORY
SPACE PHYSICS DIVISION
SPACE SYSTEMS TECHNOLOGY BRANCH

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AIR FORCE GEOPHYSICS LABORATORY
AIR FORCE SYSTEMS COMMAND
UNITED STATES AIR FORCE
HANSCOM AIR FORCE BASE, MASSACHUSETTS 01731

"This technical report has been reviewed and is approved for publication"

William N. Hall
WILLIAM N. HALL
Contract Manager

Edward C. Jonson
EDWARD C. JONSON, Lt. Col., USAF
Space Systems Technology Branch Chief

FOR THE COMMANDER

Rita C Sagalyn
RITA C. SAGALYN
Space Physics Division Director

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19 ABSTRACT (Continue on reverse if necessary and identify by block number)

This report covers the second phase of the Spacecraft Environmental Interaction Investigation. The objective of this phase of the contract was to identify and rank interaction technical issues affecting military space system development. The goal of this phase was to identify potential detrimental interactions, to review existing military standards for adequacy in aiding designers, and to recommend standard and handbook development. These objectives and goals were met by considering environmental interactions with five specific large space system designs: Space Station in Polar Orbit, Geostationary Space Platform, Nuclear Powered Satellite, Orbital Transfer Vehicle, and Astronaut Life Support/Maneuvering Systems. The designs were reviewed from an engineering point of view using the environmental interaction technology data obtained from the first phase of this contract (Environmental Interactions Technology Status: Scientific Report No. 1, AFGL-TR-85-0043). From this review, the interactions were synthesized into a comprehensive list. Available Military Standards and Handbooks for each interaction were surveyed and recommendations were made for the development of needed documents.

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1.0 EXECUTIVE SUMMARY

A three year investigation has been conducted to identify and assess adverse spacecraft environmental interactions and to determine the availability of design and test standards to minimize these effects in future large and high powered spacecraft. The first phase of this investigation (identification and assessment of natural environmental interactions technology) was summarized in Scientific Report No. 1, "Environmental Interactions Technology Status".

The second phase of the investigation reviewed the adequacy of existing military documentation and recommended a development plan for producing any needed documents. This phase also assessed the impact of environmentally induced interactions on selected future system concepts. These concepts were based on the Military Space System Technology Plan (MSSTP) [1] to provide a cross section of operational conditions, system sizes, and environments. Results of the second phase are summarized within this report.

It has been found that available documentation is not adequate to define the measures required to provide immunity from environmentally induced effects. In some cases, the technology of these interactions is not developed adequately to prepare design and test standards. The interactions deemed critical for the future systems were grouped into nine major topics with ten subtopics (see Table 1.1). Recommendations ranging from the development of Pre-Handbooks (compendiums of available knowledge) for the immature technologies to Military Standards and Handbooks for the more mature technologies were assigned.

The environmental interactions identified in the first phase of this investigation were evaluated against future concepts to determine if system impact statements would be modified when considering a defined system. As a result, the impact rating of some interactions were downgraded. However, none of the impact ratings became more severe when specific concepts were used.

In addition, only the multiple body interaction in plasma environments was overlooked in the first phase of the investigation. This interaction is really an engineering extension of the already defined plasma interaction technology. This strengthens the belief that the environmental interactions review in Phase 1 was comprehensive.

The impact of environmental interactions on future space systems are summarized in Table 1.2. The rating scheme used in this report is explained in section 3.2.2.

Table 1.1
Summary of Recommended Documentation

Topic	Rating I	Rating U	Documentation PreHDBK	HDBK	STD	CAE
Space Environment Specification	5	5		X	X	
Environment Charging:						
- High Altitude	4	4		X	X	X
- Polar-Auroral	4	3	X			
- Multiple Object	4	3	X			
Radiation Effects:						
- Electronics Degradation	4	4		X	X	
- Material Degradation	4	3	X			
- Single Event Upsets	4	4		X		
Contamination:						
- Material	5	4		X	X	
- Surface Glow	3	3	X			
Atomic Oxygen Surface Erosion	5	4		X		
Micrometeoroid Impact	4	4		X	X	
Man-Made Debris Impact	4	3	X			
High-Voltage Interactions	4	3	X			
Environmentally Induced Stresses:						
- Thermally Induced	4	5		X	X	
- Electromagnetic	4	5	X			

Rating Key: I = INTERACTION IMPACT
U = UNDERSTANDING OF PHENOMENON

Documentation Key: PreHDBK = MILITARY PRE-HANDBOOK
HDBK = MILITARY HANDBOOK
STD = MILITARY STANDARD
CAE = COMPUTER AIDED ENGINEERING TOOL DEVELOPMENT

Table 1.2
System Impact of Environmental Interactions On Future Spacecraft Concepts

Interaction	PSS	EVA	GSP	NPS	OMV
Plasma Environment:					
- High Altitude Spacecraft Charging	-	-	4	-	4
- Polar-Auroral Spacecraft Charging	4	4	-	4	4
- Multiple Object Charging	-	4	-	-	4
- High-Voltage Systems Interactions	4	4	2	4	-
High Energy Radiation Environment:					
- Radiation Damage	4	4	4	4	4
- Single Event Upsets	4	4	4	4	4
- Hazard To Man-In-Space	4	4	-	4	-
Neutral Environment:					
- Atmospheric Drag	5	1	-	5	5
- Atomic Oxygen Surface Erosion	5	1	-	5	5
- Surface Glow	-	1	-	-	-
- Sputtering	2	1	-	2	2
Particle Environment:					
- Micrometeoroid Impact	-	2	2	-	-
- Man-Made Debris Impact	4	2	-	2	2
Solar Radiation Environment:					
- Coating Degradation	4	1	4	4	-
- Thermal Forces	3	1	3	3	-
- Biological Hazard	2	2	2	2	-
Self-Generated Environment:					
- Material Contamination	4/5	1	4/5	4/5	4/5
- Thruster Contamination	4/5	1	4/5	4/5	4/5
- Nuclear System Interactions	-	-	-	4	-
Electromagnetic Environment:					
- Motion Induced Electric Fields	3	1	-	3	-
- Current Generated Forces	4	1	-	4	-
- Magnetic Torques	4	1	-	4	-
- Induced Electric Field Torques	3	1	-	3	-

Note: (-) Indicates Interaction is Not Applicable to System

Key:
PSS = SPACE STATION IN POLAR ORBIT
EVA = ASTRONAUT LIFE SUPPORT/MANEUVERING EQUIPMENT
GSP = SPACE PLATFORM IN GEOSYNCHRONOUS ORBIT
NPS = NUCLEAR SPACE POWER SYSTEM
OMV = ORBITAL MANEUVERING/TRANSFER VEHICLE (OMV/OTV)

2.0 INTRODUCTION

Currently, the Air Force is contemplating future space missions that use very large, high powered spacecraft. Missions consisting of very large platforms exposed to the space environment and requiring multi-kilowatts of power for operation are in the planning stages. Space Based Radar (SBR) is just one of the many concepts listed in the Military Space Systems Technology Plan (MSSTP) that will necessitate a new generation of spacecraft operating in a variety of orbits from low earth to beyond geosynchronous. The natural environment may interact with these future, large systems in ways which have not yet been encountered. Therefore, past experiences cannot be used as a guarantee of interactions immunity.

If detrimental effects from these interactions are uncovered late in the vehicle design stages, then costly retrofits will likely be required. To avoid these possible retrofits, the Air Force Geophysics Laboratory (AFGL) has funded this Spacecraft Environment Interaction (SEI) investigation to summarize the adverse interactions and provide a guide for the documentation requirements needed to ensure that serious interactions can be avoided in these new designs. The SEI investigation has produced three reports to document the results: the Final Report [2], Scientific Report No. 1 [3], and this report (Scientific Report No. 2).

The final report of this SEI investigation reviews the methods used and the study results. Scientific Report No. 1 contains a detailed summary of the status of the seven environmental interaction categories. The report also covers the documentation available for the critical interactions and discusses the impact of these interactions on future military space concepts. These two reports, along with this one, should be used as a set to cover the major aspects of environmental interactions.

This report is divided into two sections. The first defines the critical environmental interactions and reviews the Military Standards and Handbooks [4] available to Program Managers for establishing design specifications, and to manufacturers for building spacecraft immune to these interactions. Recommendations for additional documentation based upon the maturity of the interaction technology are also presented.

The second section of this report summarizes the effects of these interactions on future space system concepts. These applications are not detailed studies of the interaction's impact, but are generalized reviews because the concepts are not defined well enough for a detailed impact study. The applications studies were used to assess whether or not the system impact rating would change when the interactions were evaluated for more specific design concepts and to verify that no interactions were overlooked.

2.1 REFERENCES

1. Information on the "Military Space System Technology Plan" is available from the USAF Space Technology Center, Kirtland AFB, NM.

2. Stevens, N.J., and Kirkpatrick, M.E., "Spacecraft Environment Interaction Investigation - Final Report", AFGL-TR-86-0214, ADA 179183, October 1986.

3. Stevens, N.J., Kirkpatrick, M.E., Chaky, R.C., Howard, J.E., Inouye, G.T., and Beran, E.W., "Environmental Interactions Technology Status", Scientific Report No. 1, AFGL-TR-85-0043, October 1986.

4. "Index of Specification and Standards", Part II Numerical Listing, Department of Defense, 1 July 1985.

3.0 CRITICAL ENVIRONMENTAL INTERACTIONS DOCUMENTATION

3.1 SCOPE

This section describes the availability of Military Standards, Handbooks, or other documentation that would provide guidance to program offices for obtaining a product immune to interaction hazards and to designers for controlling interaction effects. Twenty-four environmental interactions were defined in Scientific Report No. 1. These were then reviewed and assimilated into a group of fourteen critical interaction technology topics

Within this chapter, the critical interactions are discussed by severity of impact rather than using the environment categorization defined in Scientific Report No. 1. The approach used in this report: defines the interaction; briefly summarizes the known hazards associated with this interaction; reviews the status of the interaction technology; identifies available supporting programs, standards, handbooks, or other documentation; and recommends documentation that should be developed. In addition to these critical interactions, the need for a comprehensive Space Environment Standard is also discussed in this chapter.

Detailed discussions of these interaction technologies and extensive technical references are available in Scientific Report No. 1. Relevant military documentation was found by searching the Military Standards Index and reviewing the documents to determine if they are applicable to this study.

Because some environmental interaction technologies are believed to be mature and well-developed, they are not discussed in this report. These technologies include nuclear power systems, solar array degradation in radiation environments, biological hazards, and thermal control coating degradation.

3.2 DEFINITION OF TERMS

In order to discuss the technical issues, it is necessary to first define the terms and the rating scheme used within this analysis.

3.2.1 Terms Used

The five terms used in this report which need definition are: Military Standard, Test Specification, Handbook, Pre-Handbook, and Orbit.

3.2.1.1 *Military Standard*

A Military Standard is self-contained document, without referenced material, used to establish mandatory requirements for a specific design area or piece of hardware. Verification that the contractor has met these requirements is usually done by test, analysis, and/or inspection.

3.2.1.2 *Test Specification*

A Test Specification is a document that establishes the test objectives and the criteria to judge test results. Test specifications are used in contracts to establish requirements for successful test completion.

3.2.1.3 *Handbook*

A Handbook contains descriptions of phenomena, effects, and operational impacts that can be referenced in a contract. It may contain design recommendations and suggested test procedures. This document generally contains guidance and instruction and does not have the contractual strength of a Military Standard. Handbooks are usually written for mature technologies.

3.2.1.4 *Pre-Handbook*

A Pre-Handbook is a compendium of current information to increase the knowledge and understanding of designers and project managers and to enhance system designs. This document is prepared for developing technology topics.

3.2.1.5 *Orbit*

Three orbit acronyms are used throughout this report; LEO, PEO, and GEO. LEO (Low Earth Orbit) is an orbit less than 1000 km in altitude with a low inclination relative to the equator. PEO (Polar Earth Orbit) is an orbit less than 1000 km in altitude with an inclination greater than 60°. Finally, GEO (Geosynchronous Earth Orbit) is an orbit approximately 35,900 km in altitude (6.6 earth radii) where a satellite orbits synchronously with the Earth's rotation, remaining fixed over a point above the Earth's equator.

3.2.2 Rating Scheme

To permit a complete discussion of space environment interactions, the potential impact on system performance and the state of interaction understanding were rated on a scale of 1 to 5 (Table 3.1). This scale allows one to place interaction effects and technologies in relative perspective.

Table 3.1
Rating Scheme for Environment Interactions

System Impact	Maturity
1. Negligible	1. Negligible
2. Small	2. Slight
3. Moderate	3. Moderate
4. Large	4. Considerable
5. Catastrophic	5. Complete

3.3. SPACE ENVIRONMENT STANDARD

A complete definition of the environment in the orbit of interest is required to understand and control environmentally induced interactions. This is often achieved by searching through several references to identify high and low energy plasma, solar, magnetic, cosmic ray, micrometeoroid, and debris environments. The models usually give data in scientific terms which must be converted into engineering terms in order to evaluate the impact on a specific spacecraft design.

NASA has several interaction models of interest to this investigation. In addition, others have developed models to study specific areas (e.g. cosmic ray environments for single event upsets). At present, however, a total environment model for predicting induced interactions does not exist. This proposed standard would be useful for imposing environmental requirements upon spacecraft systems early in their design phase. Such a standard could also list the environmental interactions of concern and identify threats that should be considered.

The maturity of understanding of the space environment is COMPLETE (5) and the impact to space systems is CATASTROPHIC (5). Thus, it is recommended that a Space Environment Standard, which would incorporate new or modified Military Standards to include space environment requirements, should be prepared.

3.4 ENVIRONMENTAL INTERACTIONS TECHNOLOGY ISSUES

In this section of the report the following interactions will be addressed: Environmental Charging, Radiation Effects, Contamination, Atomic Oxygen Surface Erosion, Micrometeoroid Damage, Impact with Man-Made Debris, High-Voltage System Interactions, and Environmentally Induced Stresses.

3.4.1 Environmental Charging Interactions

3.4.1.1 *Interaction Description*

Environmental charging includes all environmentally induced phenomena resulting in charging of dielectrics. There are three different categories in this interaction: High Altitude Charging, Polar-Auroral Charging, and Multiple Object Charging.

The first category covers geomagnetic substorm charging of geosynchronous spacecraft. It includes surface charging, dielectric bulk charging, and cable and circuit board charging phenomena associated with such substorms. The second category covers the effects on polar orbiting vehicles charged by auroral beams in the high latitudes. The third category covers interactions between two or more spacecraft charged to different potentials by the environment. Interactions between an astronaut, during Extra Vehicular Activity (EVA), and a large spacecraft would be an example of multiple object charging. Detailed reviews of these interactions (except Multiple Object Charging) can be found in Scientific Report No. 1 (Section 3.0).

3.4.1.2 *Hazard*

The hazards associated with these interactions are the same: environmentally induced surface potentials can disrupt sensor and instrument operations as well as cause enhanced contamination by ionizing atoms and molecules and attracting them back to the surface. Discharges resulting from differential charging between various parts of the spacecraft can generate electromagnetic noise. This results in electronic system logic upsets, sensor signal noise, electronic unit failures, and memory upsets.

Discharges can also result in physical damage to thermal control surfaces. One of the more serious consequences of these charging interactions would occur in spacecraft using automated systems. In this case, environmentally induced discharges could either directly initiate a stored command sequence or provide erroneous sensor signals, causing unplanned sequence operations. A discharge could also disrupt the memory so that a sequence would not begin as planned. The effect of these interactions is detrimental to spacecraft operations.

3.4.1.3 *Status*

As noted above, the impact of environmental charging on spacecraft systems is serious and was rated as having a LARGE (4) impact for all three categories. After ten years of studies, the level of understanding for High Altitude Charging is CONSIDERABLE (4). Uncertainties in coupling from discharge sites to electrical systems still have to be answered, but the technology is reasonably well understood.

Polar-Auroral Charging, a newly recognized phenomena, is still being studied. Information obtained from high altitude charging studies is useful, but more work must be done. Therefore, the level of understanding is rated as MODERATE (3). Multiple Object Charging is also a relatively new concept with unknown implications. Thus, it is also rated MODERATE (3).

3.4.1.4 *Supporting Programs, Standards, and Handbooks*

High Altitude Charging studies were conducted under a joint Air Force/NASA Spacecraft Charging Investigation agreement that included ground test studies along with a space flight experiment called Spacecraft Charging at High Altitudes (SCATHA). This investigation is essentially complete. Polar-Auroral and Multiple Object charging are actively being investigated by AFGL.

Applicable Military Standards and Handbooks are listed in Table 3.2. Note that there are no specific High Altitude Charging Standards or Handbooks, other than NASA and Aerospace documents. These documents should be upgraded to include information obtained in the past three years and published as a Military Handbook.

3.4.1.5 *Recommendations*

High Altitude Charging: The technology for this category of environmental charging is reasonably mature and is ready for the preparation of a Military Standard and Handbook. It is also sufficiently advanced to develop Computer Aided Engineering (CAE) tools. The use of such tools would significantly enhance the implementation of design requirements.

Table 3.2
Environmental Charging:
Applicable Standards, Handbooks, or Documents

MIL-STD-1541	EMC Requirements for Space Systems (Numerous EMC and EMI Standards/Handbooks)
MIL-STD-1540	Test Requirements for Space Vehicles
DoD-STD-1686	ESD Control Program for Protection of Electrical and Electronic Parts
DoD-HDBK-263	ESD Control Handbook for Protection of Electrical and Electronic Parts
MIL-HDBK-337 MIL-A-83377B MIL-B-5087B	Bonding and Adhesive Standards/Handbooks
NASA TP-2361	Design Guidelines for Assessing and Controlling Spacecraft Charging Effects
SD-TR-85-26	The Aerospace Spacecraft Charging Document

Polar-Auroral and Multiple Object Charging: The technologies for these categories are still being developed and therefore Standards should not be prepared. Pre-Handbooks, however, should be developed to provide available information to designers and managers so that they may make accommodations for potential hazards.

3.4.2 Radiation Effects

3.4.2.1 *Interaction Description*

Radiation interactions occur when spacecraft are subjected to high energy space radiation. Three aspects of this interaction are considered here. The first is a malfunction (temporary or permanent) of electronic components due to sensitivity to radiation. Radiation induced changes in surface or bulk properties is the second area discussed. The final aspect is the change in logic state in a digital electronic device triggered by a high energy particle (e.g. single event upset). A detailed review of these interactions was given in Scientific Report No. 1 (Section 4.0).

3.4.2.2 *Hazard*

Radiation damage degrades electronic component performance up to and including failure. Devices have different tolerances to radiation levels, thus, radiation shielding must be provided to protect the most sensitive devices. In composite materials, radiation can alter physical properties and cause possible state changes. Microcracking under long term radiation levels is also possible. The hazard from single event upsets is the potential for a system to malfunction due to spontaneous changes in logic state of a digital device within that system. In an automated system, mission failure could result.

3.4.2.3 Status

The impact of radiation interactions on system performance has been rated as LARGE (4). The understanding of radiation shielding requirements is essentially complete except for the new devices that are being developed. Hence, the understanding level is given as CONSIDERABLE (4). Since the radiation damage to composite materials is still being evaluated, the understanding rating of this interaction is given as MODERATE (3). Single event upset phenomena in devices are still being studied, but considerable progress has been made. Therefore, the understanding rating for this interaction is given as CONSIDERABLE (4).

3.4.2.4 *Supporting Programs, Standards, and Handbooks*

The Air Force is sponsoring a flight experiment, currently scheduled for launch in the 1990s, to investigate radiation effects in electronic devices. This experiment will be part of the Combined Release and Radiation Effects Satellite (CRRES).

Applicable Military Standards and Handbooks are listed in Table 3.3. Standards for radiation shielding of components are parts oriented; there are no systems oriented Standards. No Standards or Handbooks for composite material degradation presently exist. Finally, for SEUs, there are lists of part susceptibilities compiled and published by the Aerospace Corporation and NASA-JPL (see Section 4.3 of Scientific Report No. 1).

Table 3.3
Radiation Effects:
Applicable Standards, Handbooks, or Documents

MIL-STD-242	Electronic Equipment Parts: Selected Standards
MIL-E-5400	General Specification for Aerospace Electronic Equipment
MIL-E-558	Packaging of Electronic Equipment and Parts
DoD-E-8983	General Specification for Aerospace Electronic Equipment (Innumerable Specifications for Specific Parts, Connectors, and Modules)
——	Various Summaries of Parts Behavior by JPL and the Aerospace Corporation

3.4.2.5 *Recommendations*

A Military Standard should be prepared concerning the radiation interaction with electronic components. This standard should identify the approach to be used to protect systems from malfunctions and establish requirements for system operations. A Handbook should also be prepared comparing the various approaches to determine radiation shielding protection and recommend the preferred approach.

For composite material damage due to high energy radiation, a Pre-Handbook should be prepared summarizing the available information on this interaction. Finally, a Handbook for single event upsets, should be prepared which summarizes available data on parts susceptibility and establishes guidelines for protecting systems against upsets.

3.4.3. Contamination

3.4.3.1 *Interaction Description*

Broadly defined, contamination occurs when an outside agent interferes with the operation of a system. Under this definition, it is possible to include material contamination consisting of particles and vapors from outgassing and thruster operations, as well as contamination resulting from surface glow. Particulate and vapor contaminants deposit preferentially on cold spacecraft surfaces. Surface glow is a LEO phenomenon emanating from those surfaces facing in the velocity direction. Details on these interactions can be found in Scientific Report No. 1 (Sections 5.4 and 8.0).

3.4.3.2 *Hazard*

Degradation of thermal control and optical surfaces is one hazard associated with material contamination. This shortens the lifetime of spacecraft systems due to higher operating temperatures and reduces the performance of sensors that require cold temperatures for operation. Surface glow contamination would affect the performance of optical sensors. At present, it is believed that the emission is limited to the 4000 to 8000 angstrom wavelength range.

3.4.3.3. *Status*

For material contamination, the impact on system performance of sensors and other critical components can be catastrophic, other systems may only experience minor effects. Since ratings are given at the highest level, this interaction is designated as CATASTROPHIC (5). Due to the possible effect on the operation of optical sensors on spacecraft, surface glow contamination was judged to have a MODERATE (3) impact.

Material contamination phenomena are well understood and approaches to control its effects have been developed. Hence, its maturity was rated as CONSIDERABLE (4). The understanding of surface glow contamination is only rated MODERATE (3) since the effect was found only after the Shuttle first began to fly experiments. This interaction is still being evaluated.

3.4.3.4 *Supporting Programs, Standards, and Handbooks*

Two major Air Force programs are under way to understand interactions resulting from contamination. The first, directed from the Air Force Wright Aeronautical Laboratory (AFWAL), is focused primarily on material contamination. The second, directed from the Air Force Geophysics Laboratory (AFGL), concentrates on glow and other contamination effects. Work is also being conducted by the NASA centers and technology supported by the government is well coordinated. In addition to these efforts, aerospace industries are evaluating contamination control for specific projects.

The applicable Military Standards, Handbooks, and NASA Specifications for contamination are listed in Table 3.4. No Standard for a systems approach to material contamination exists. NASA has a Contamination Control Plan document for the Shuttle, but this is not applicable as a Military Standard that imposes contractual requirements for space hardware. In addition, Standards for surface glow have not been prepared because of the low level of understanding of this phenomenon.

Table 3.4
Contamination Summary:
Applicable Standards, Handbooks, or Documents

MIL-HDBK-4Ø6	Contamination Control Technology: Cleaning Materials for Precision Precleaning and Use in Clean Rooms and Clean Work Stations
MIL-HDBK-4Ø7	Contamination Control Technology: Precision Cleaning Methods and Procedures
MIL-STD-889	Dissimilar Metals
MIL-STD-1246	Product Cleanliness and Contamination Control Program
FED-STD-2Ø9B	Clean Room and Work Station Requirements, Control Environment
NASA JSC Ø77ØØ	Space Shuttle and Ground System Specification
NASA JSC SN-C-ØØ5	Contamination Control Requirements for Space Specification
NASA JSC SE-S-ØØ73	Space Shuttle Fluid Procurement and Use Control Specification
NASA JSC SE-R-ØØØ6	NASA JSC Requirements for Materials and Processes General Specification
NASA JSC-8131	Space Shuttle System Contamination Control Plan
NASA JSC SP-R-ØØ22	Vacuum Stability Requirements of Polymeric Materials for Spacecraft Applications General Specification
NASA JSC Ø8962	Compilation of Volatile Condensible Materials Data on Non-Metallic Materials
ASTM E-595	Standard Test Method for Total Mass Loss on Collected Volatile Condensible Materials for Outgassing in Vacuum Environments

3.4.3.5 *Recommendations*

A systems-oriented Military Standard and Handbook should be prepared for material contamination. These documents would codify the available information and identify what should be done for various mission categories. For surface glow contamination, however, the state of the art is too uncertain for any recommendation at this time, other than to prepare a Pre-Handbook. When more is known about the basic mechanisms involved, a Handbook should be prepared.

3.4.4 Atomic Oxygen Surface Erosion

3.4.4.1 *Interaction Description*

This interaction is caused by the impact of atomic oxygen with spacecraft surfaces facing in the velocity (ram) direction. It results in surfaces of some materials being eroded. In equatorial orbits, atomic oxygen is the dominant species for altitudes up to 650 km. A detailed discussion of this interaction was given in Scientific Report No. 1 (Section 5.3).

3.4.4.2 *Hazard*

Atomic oxygen surface erosion is a serious interaction because it erodes materials and coatings. A list of materials subject to severe degradation was given Scientific Report No. 1 (Section 4.2). This list indicates that material erosion may impact thermal control systems, by changing thermo-optical properties, as materials degrade and erode. Eroded material could also deposit on other surfaces, adding to contamination problems.

The use of unprotected Kapton as a structural element (e.g. solar array substrates) should be avoided in LEO applications, since the predicted erosion rate is about 2 mils in a year in a 300 to 400 km orbit. The erosion of certain metals (e.g. silver) indicates that this interaction would influence the performance of solar thermal dynamic and concentrator solar cell systems if silvered mirrors are used. In general, sensors and optics will be affected by both erosion and contamination.

3.4.4.3 *Status*

Impact of this interaction on system performance is rated CATASTROPHIC (5). The understanding of the interaction mechanism is rated as CONSIDERABLE (4).

3.4.4.4 *Supporting Programs, Standards, and Handbooks*

Ongoing Air Force and NASA sponsored programs to evaluate material responses and develop mitigation techniques for atomic oxygen erosion (Scientific Report No. 1, Section 5.3) are being supported by flight experiments.

There are no applicable Military or NASA Standards/Handbooks for this interaction. The phenomenon was identified when Shuttle flights began and materials exposed to the space environment were returned to Earth for inspection. Only after several days exposure to the environment, on the third Shuttle flight (STS-3), did it become apparent that surfaces were being eroded. The primary sources of information on this phenomenon are coming from Air Force and NASA investigations.

3.4.4.5 *Recommendations*

For this interaction, a Handbook should be prepared to describe the mechanisms which generate this erosion, list the material and coating susceptibilities, and outline possible test procedures to evaluate the effect of this interaction on system performance. Currently, knowledge about this interaction is insufficient for a Military Standard, since the requirements that must be imposed are not defined, acceptable mitigation approaches are vague, and verification techniques to demonstrate compliance with the requirements are uncertain.

3.4.5 Micrometeoroid Impact

3.4.5.1 *Interaction Description*

This interaction occurs when spacecraft are bombarded by micrometeoroids. Even though these particles have small masses, impacts with systems can result in significant damage because of their very high velocities. While designers have been aware of this environment for years, the seriousness of this interaction has only been realized with the retrieval of space hardware by the Shuttle. Since future systems will be larger, lighter, more automated, and complex, this interaction must be assessed carefully. A detailed review of this interaction was given in Scientific Report No. 1 (Section 6.1).

3.4.5.2 *Hazard*

High velocity micrometeoroid particles can penetrate and damage systems containing mirrors or sensors. These particles can also damage thermal control surfaces and could penetrate and rupture pressure vessels. The erosion of mirror surfaces can impact solar power generation concepts. Damage is caused by both primary surface penetration and secondary surface spalling or crazing.

3.4.5.3 *Status*

The impact on system performance for this interaction is rated as LARGE (4) and the level of understanding is rated CONSIDERABLE (4). Information on this interaction has been available for years, however, effects on larger spacecraft is a relatively new field.

3.4.5.4 *Supporting Programs, Standards, and Handbooks*

Studies are being conducted at NASA facilities to evaluate the performance of various materials when struck by hypervelocity particles. There are no applicable Military Standards or Handbooks for this interaction, only the three NASA reports listed in Table 3.5. These documents, coupled with open literature reports, are the only sources available to the designers to alleviate any concern for this interaction.

Table 3.5
Micrometeoroid Impact Interaction Summary:
Applicable Standards, Handbooks, or Documents

NASA-SP-8Ø13	Micrometeoroid Environment Model - 1969 (Near Earth to Lunar Surface)
NASA-SP-8Ø38	Micrometeoroid Environment Model - 197Ø (Interplanetary and Planetary)
NASA-SP-8Ø42	Micrometeoroid Damage Assessment

3.4.5.5 *Recommendations*

A Military Standard and Handbook should be prepared, since there is considerable knowledge about this interaction and no present documentation.

3.4.6 Man-Made Debris Impact

3.4.6.1 *Interaction Description*

This interaction involves the impact of man-made debris with spacecraft surfaces. Debris could include residual pieces from spacecraft (e.g. clamps, holders, tie wraps, etc.) or residual dust from rocket firings or explosions. These spacecraft remnants create hazards for present and future missions. Because man-made debris in space is increasing, the frequency of impacts is also increasing. Although the velocities of these particles are less than those of micrometeoroids, man-made particles are more massive. A detailed review of this interaction was given in Scientific Report No. 1 (Section 6.2).

3.4.6.2 *Hazard*

The hazards associated with man-made debris impacts are the same as those from micrometeoroids (i.e. surfaces are eroded and punctured). This is a serious concern for critical flight thermal control surfaces, mirrors, optical sensors, and pressure vessels.

3.4.6.3 *Status*

The impact of this interaction on system performance was rated as LARGE (4). Understanding of this phenomenon is MODERATE (3), since the debris environment is changing with each launch and models of have not been validated.

3.4.6.4 *Supporting Programs, Standards, and Handbooks*

A large driver for developing debris impact technology is the NASA Space Station. NASA is concerned about this interaction and is supporting the development of space debris models. *Orbital Debris Environment for Space Station* (JSC-20001) is the first such document on this topic. There are no applicable Military Standards or Handbooks.

3.4.6.5 *Recommendations*

For this interaction, a Pre-Handbook should be prepared. It is premature to develop Handbooks or Standards.

3.4.7 High-Voltage System Interactions

3.4.7.1 *Interaction Description*

This interaction results from the operation of high-voltage systems in a space plasma environment. It covers interactions due to operations of high-voltage solar arrays, as well as high-voltage power distribution and management systems. Any system in which biased conductors, surrounded by dielectrics, are exposed to the space plasma environment can be affected. A detailed review of this interaction can be found in Scientific Report No. 1 (Section 3.4).

3.4.7.2 *Hazard*

Power loss, due to coupling to the surrounding plasma, is the principal hazard. These losses are due to either plasma initiated breakdowns or direct current loops through the plasma. The operation of a high-voltage system in a plasma can also change the structure potential relative to the space plasma potential. This could disrupt instrument operations and cause erroneous data. In addition, the combination of the high-voltage system and the plasma can induce stresses in dielectrics.

3.4.7.3 *Status*

System performance impact, for this interaction, was rated as LARGE (4). The phenomenon has been studied for at least 15 years but the state of understanding can only be rated as MODERATE (3).

3.4.7.4 *Supporting Programs, Standards, and Handbooks*

Evaluation of this interaction has been supported primarily by the Air Force Wright Aeronautical Laboratories (AFWAL) and the NASA Lewis Research Center. The Air Force Geophysics Laboratory (AFGL) is directing the Jet Propulsion Laboratory (JPL) to build a Photovoltaic Array Space Power Plus Diagnostics (PASP Plus) experiment for spaceflight in the 1990s. This environmental interactions payload will address high-voltage interaction issues.

While there are no directly applicable Military Standards/Handbooks for this interaction, related documents (i.e. EMC and EMI Standards) exist. NASA has written a draft Environmental Interactions Handbook for the Space Station, but this document has not been released.

3.4.7.5 *Recommendations*

For this interaction the recommendation is to prepare a Pre-Handbook. However, the present state of knowledge is inadequate to provide positive information on this interaction phenomenon and mitigation techniques.

3.4.8 Environmentally Induced Stresses

3.4.8.1 *Interaction Description*

Environmentally induced stresses result from the interaction between the space (e.g. solar and electromagnetic) environment and the spacecraft. They range from thermal stresses caused by differential expansion of large flexible bodies, to stresses induced by spacecraft motion in the Earth's magnetic field. A discussion of these topics was given in Scientific Report No. 1 (Section 9).

3.4.8.2 *Hazard*

Hazards, associated with these induced interactions, arise when stresses are generated within flexible support structures. The result is mirror surface and solar array substrate distortion or spacecraft torques affecting attitude control systems.

3.4.8.3 *Status*

The potential impact of these interactions on system performance is considered LARGE (4). The understanding of the thermal effects is COMPLETE (5). The understanding of induced electromagnetic stress interactions is also rated as COMPLETE (5). It is the application of these known technologies to large space structures that must be evaluated carefully.

3.4.8.4 *Supporting Programs, Standards, and Handbooks*

The thermophysics community has programs to understand all phases of thermally induced phenomena. Results indicate that numerous torques and stresses can be generated. While each may be small, they can couple to produce magnified effects.

No thermal system Military Standards or Handbooks were found for spacecraft. However, Standards for airplane cockpits (MIL-T-81571) are available. In addition, there are no Standards or Handbooks for electromagnetically induced stresses.

3.4.8.5 *Recommendations*

Material on spacecraft thermal control should be organized into a Handbook. A Thermal Protection Standard and Handbook for spacecraft should be prepared for thermally induced stresses. Electromagnetically induced stresses should be evaluated for several configurations, to assess the impact in more detail, before any formal documentation can be prepared. Meanwhile, a Pre-Handbook summarizing these interactions and concerns should be produced.

3.5 SUMMARY

Available documentation for the environmental interactions critical to the design of future, large spacecraft, has been reviewed. Each interaction was rated for the impact it could have on the system and its level of maturity. In this review, it was found that existing contractual documentation (i.e. Military Standards and Handbooks) was deficient for handling environmentally induced interactions. Therefore, based upon the maturity of these interaction technologies, additional documentation should be prepared. These documentation requirements range from Pre-Handbooks for developing technologies, to Standards and Handbooks for the more mature ones. For established technologies, development of Computer-Aided Engineering (CAE) design tools is also recommended. The results of this review were shown in Table 1.1.

In addition to the interaction technologies, this review has also identified the lack of a system level Space Environment Standard. Environment models exist, but there are no standards to specify what models to use and how they should be implemented. The development of a Space Environment Standard is highly recommended.

4.0 APPLICATIONS TO FUTURE MILITARY SYSTEMS

4.1 PURPOSE OF APPLICATION STUDY

In this section, the possible impact of environmental interactions on future space system designs are reviewed. In this review, the consequences of the interactions are considered more for their engineering impact on spacecraft systems than as general technology impacts, as was done in the Environmental Interactions Technology Status Report (Scientific Report No. 1). Unfortunately, no future space system designs are sufficiently advanced to allow a detailed examination. Therefore, the concepts used in this review were derived from the Military Space System Technology Plan (MSSTP).

Rather than try to evaluate environmental interactions for all thirty-seven concepts on the MSSTP list, five generalized spacecraft were chosen to represent all the concepts and to cover interactions in a wide range of natural operating environments. The concepts chosen are: Space Station in Polar Orbit, Astronaut Life Support/Maneuvering System in Polar Orbit, Space Platform in Geosynchronous Orbit, Space Nuclear Powered System, and Orbital Maneuvering/Transfer Vehicle.

The interactions identified in Scientific Report No. 1 are used in this evaluation. If the impact of an interaction is the same for a second system or concept, a reference will be made to the previous discussion. After the interactions are discussed, they are rated for specific system impact, using the scale defined in Section 3.3.2 of this report.

The differences in the impact ratings derived for these spacecraft concepts, as compared to those obtained from the technology study, are identified in the narrative. Some technologies are sufficiently mature, thus only a short discussion is necessary. Others are so uncertain that little is documented about the interaction impact on system performance. Finally, those interactions whose impact may be modified as a result of this application study will be discussed in detail. Scientific Report No. 1 contains the detailed descriptions of the interaction technologies and references.

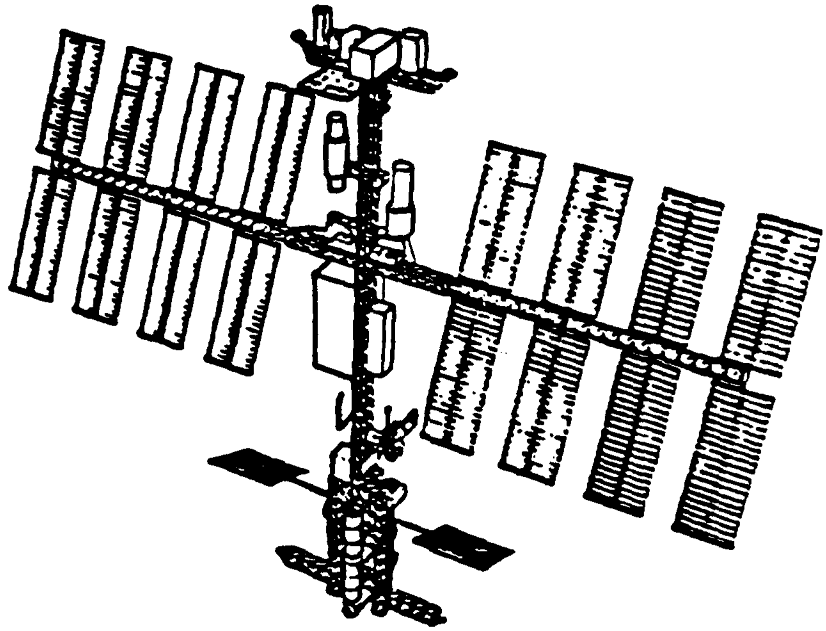
4.2 SPACE STATION IN POLAR ORBIT

4.2.1 Concept Description

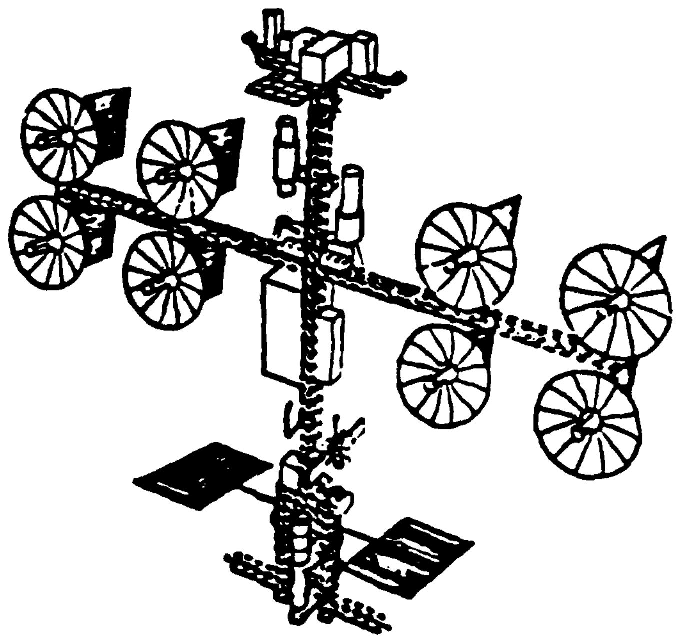
One technology concept chosen for this interactions study to focus upon was a space station similar to the one that NASA is currently designing (Figure 4.1). The overall dimensions of this space station were assumed to be 120 m from top to bottom and 100 m across the power sources. Both Photovoltaic Solar Array and Solar Thermal Dynamic power generation systems, with a load power of 300 kW, were examined (Figure 4.2 A and B).

The photovoltaic solar arrays would have a total surface area on the order of 3000 m^2 while the solar thermal dynamic collectors would need a total surface area of approximately 2000 m^2. This station was assumed to be in a 400 km polar orbit, primarily to estimate the interactions impact of this more severe environment. The

Figure 4.1 Space Station



(A) Photovoltaic Solar Arrays



(B) Solar Thermal Dynamic Concentrators

Figure 4.2 Space Station Power Concepts

stations operational lifetime was assumed to be 30 years. In addition, sensors, communications systems, and scientific payloads would be the mainstay of this concept.

4.2.2 Plasma Environment Interactions

4.2.2.1 *High Altitude Charging*

This interaction does not apply to the space station concept, since this system will not operate in high altitudes.

4.2.2.2 *Polar-Auroral Charging*

This interaction, described in Scientific Report No. 1 (Section 3.3), is applicable to both the photovoltaic and solar thermal dynamic power system options. Spacecraft in polar orbits charge from encounters with intense auroral current beams in the high latitude regions of their orbits. Data on this phenomenon has been obtained over the past few years from instruments on the Defense Meteorological Satellite Program (DMSP) vehicles in 840 km polar orbits. The particle instruments on these vehicles have shown that structure potentials can change rapidly.

For example, a DMSP charging event in which the vehicle potential reached -400 V is shown in Figure 4.3. A factor contributing to this charging was a pronounced decrease in the surrounding thermal plasma density. Rapid changes in potentials were a significant characteristic of this event. The whole event was over in about 20 seconds and the peaks lasted only seconds. There are no similar characteristics in the catalogs of charging at geosynchronous altitudes.

The -400 V structural potential experienced by DMSP on several occasions was not, by itself, catastrophic. The satellite apparently experienced no anomalous behavior. However, concern was more towards what such beams could do to very large structures. An initial study of polar-auroral charging concluded that charging levels would be increased for larger spacecraft. Subsequent studies, using more detailed computer programs, indicated that the same environmental conditions that charged the DMSP to -400 V would charge the Shuttle to about -3500 V. The larger space station could be charged to an even larger potential.

The main points are: (1) charging does exist in the polar regions; (2) the charging rates, both absolute and differential, are faster than those experienced in geosynchronous orbits; and (3) this phenomenon is size dependent. The consequences anticipated for this type of interaction are the same as those in spacecraft charging interactions: arc discharges, logic upsets, system failures, sensor malfunction (erroneous data), enhanced contamination, thermal coating degradation, and interruption of plasma scientific studies. The system impact for this interaction has been rated as LARGE (4).

4.2.2.3 *High-Voltage System Interactions*

This interaction was discussed in Scientific Report No. 1 (Section 3.4). Three

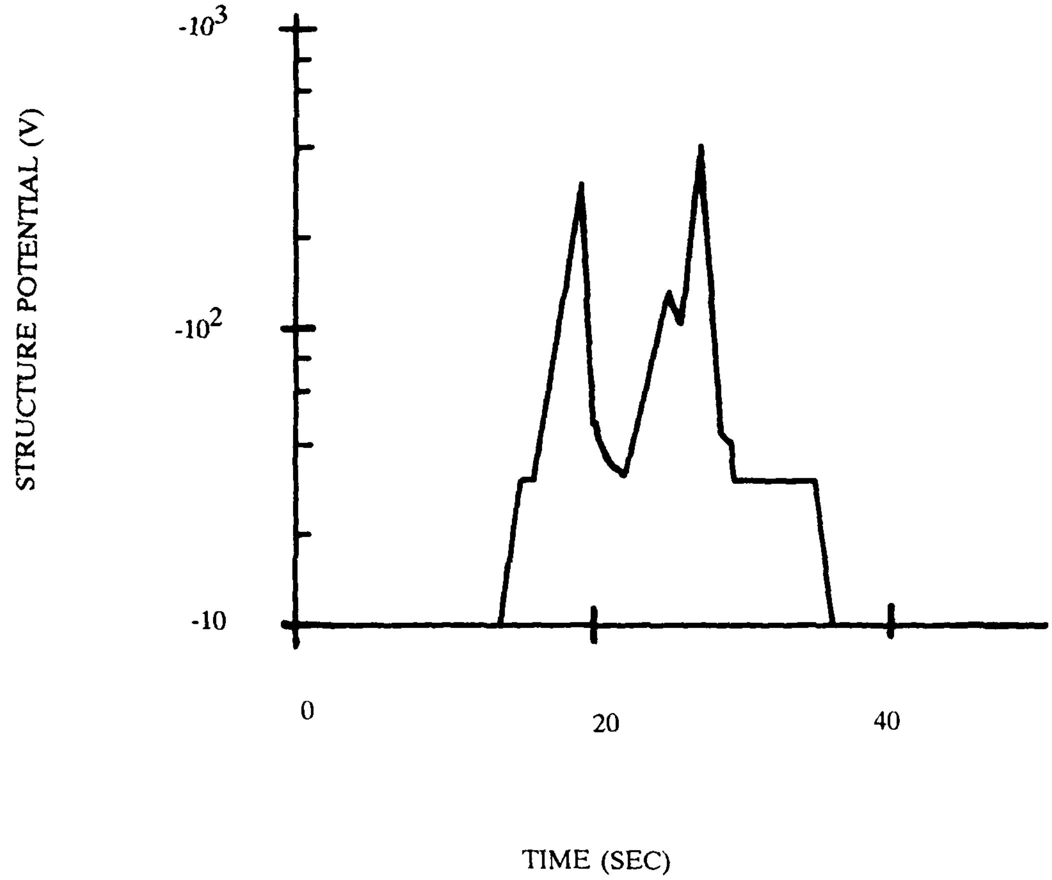


Figure 4.3 Charging of DMSP by Auroral Beam Event

high-voltage space station concepts will be described here: Planar Photovoltaic Array, Concentrator Solar Array, and Solar Thermal Dynamic Power System. In addition, some of the transmission effects will be highlighted.

4.2.2.3.1 Planar Photovoltaic Systems

The station using this power source option is assumed to have a 300 KW array to satisfy the electrical load requirements. The need for this much power means that the operating voltage of the array must be increased above the currently used 30 to 60 V, in order to minimize cable current losses and possible magnetic torque complications during high current operations. Once the voltage is increased, there are possible plasma interactions with the biased conductors at solar cell interconnects.

The station potential relative to the space plasma potential must be established. This value depends upon the plasma density, array operating voltage, and the connection of the array to the station power distribution system.

The environment of concern for this interaction is the ambient thermal plasma, or those particles having characteristic energies below 2 eV. The distribution of these particles is shown in Figure 4.4 for the assumed polar orbit. The plasma density can be expected to vary as the station moves in its orbit.

As the station orbits, the relationship between the velocity vector and the array normal can vary. This function is dependent upon the orbit chosen. If the station were in a midnight-noon orbit, it would experience both an eclipse and a velocity effect (Figure 4.5). At local noon and midnight, the velocity would be perpendicular to the array normal and the array would be in the "thermal" mode or have nominal plasma collection. At one pole, the velocity vector points away from the active face area normal creating a "wake" or reduced plasma environment for the biased interconnects. At the other pole, the velocity vector would be parallel to the active face normal, creating a "ram" or enhanced plasma environment in which the ions have a characteristic energy proportional to the vehicle velocity. The electrons are relatively insensitive to these velocity variations. A low inclination orbit would encounter similar variations. If a dawn-dusk orbit is chosen, the array is always in the thermal mode and there would be no eclipse or velocity effects.

If the array was configured to operate at 250 V in a 400 km noon/midnight polar orbit, and if the power system was electrically isolated from the station, the potential of the array relative to the space plasma potential would be as shown in Figure 4.6A. The largest voltage variation due to velocity effects occurs in the wake region.

When the power system is grounded to the station structure, the potentials relative to space would change. Unfortunately, the collection capability of large structures partially covered by dielectrics is not known. However, if the collection process is extrapolated from data obtained in tests using large metallic plates, then the behavior for positive and negative grounded systems would be as shown in Figures 4.6B and 4.6C. If the negative side of the power system is grounded, then the station structure potential

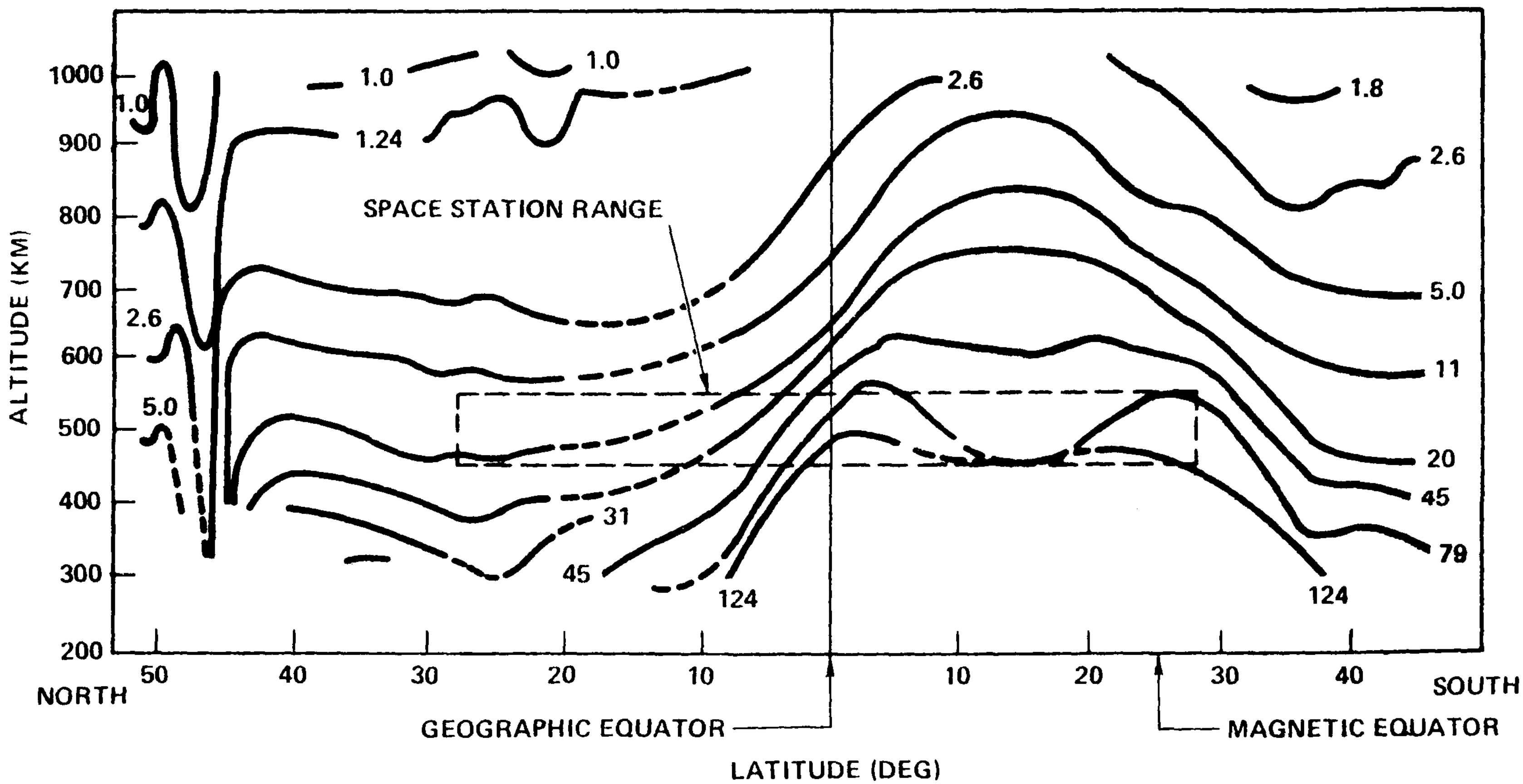


Figure 4.4 Plasma Environments (Units 10^4 cm^{-3})

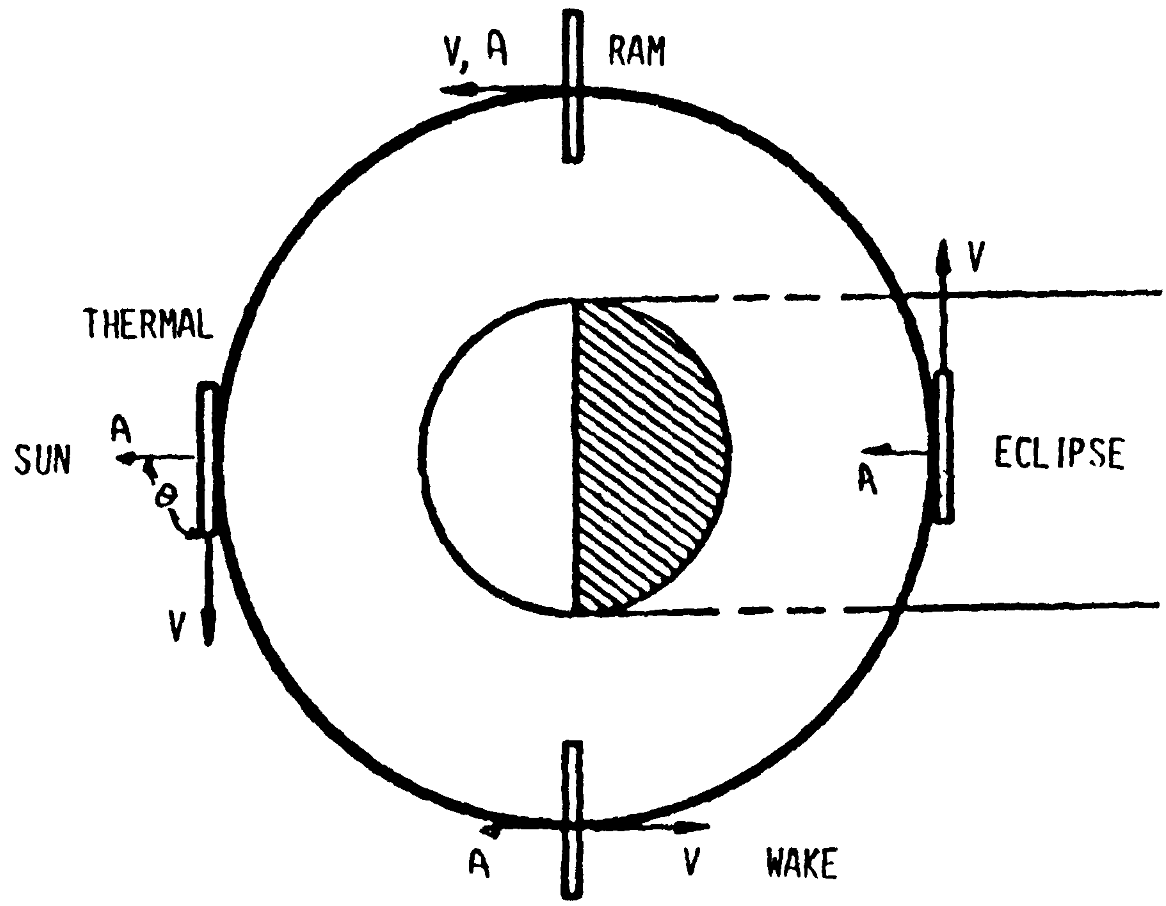
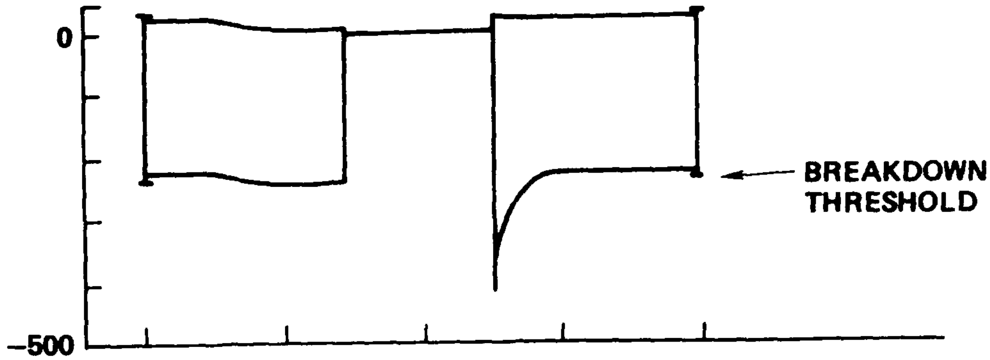
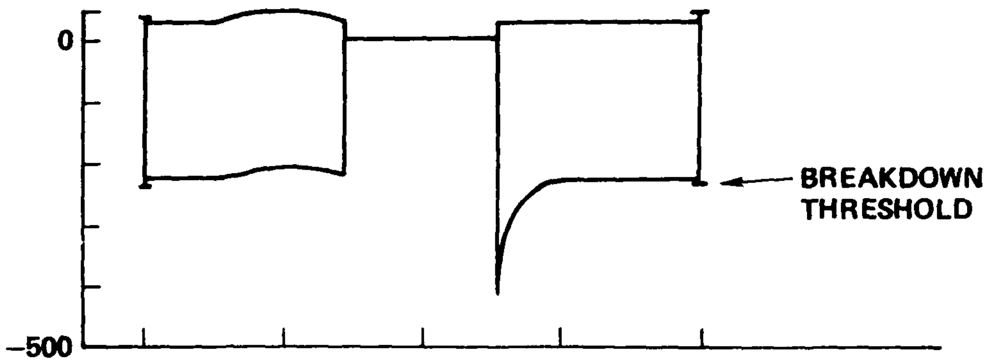


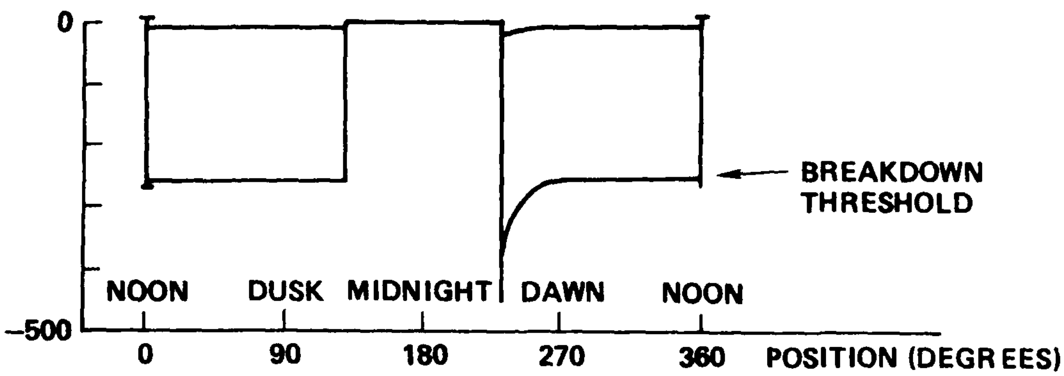
Figure 4.5 Changes in Environmental Conditions Over Orbit



(A) Array Isolated from Structure



(B) Array Negative Side Grounded to Structure



(C) Array Positive Side Grounded to Structure

Figure 4.6 Predicted Floating Potentials for High-Voltage Array relative to space plasma potential would be driven negative to approximately the operating voltage. If the positive side is grounded, then the station potential would be at the space plasma potential while the array voltage distribution would be driven negative.

An electrical system in space is usually indifferent to the potential between the structure and the space plasma. However, instruments and sensors may be sensitive to this relationship. Thus, the impact of voltage differential must be assessed.

Discharges should not occur in a 400 km orbit if operating voltages are limited to less than 250 V. If they do occur, there could be concern for the impact of this transient upon the power system performance. The result of a discharge analysis on a system operating at 250 V is illustrated in Figure 4.8. In this example, the solar array power system was configured with batteries, a solar array switching unit (SASU), and transmission line characteristics (Figure 4.7). A discharge was simulated simply by allowing part of the array to be shut off for five microseconds. It was found that the load fluctuations due to this discharge were negligible (i.e. less than 1%). This indicates that discharges may not be detrimental. However, the impact of repeated transients on the electronic components was not addressed. Over the long mission life, it must be assumed that there could be repeated discharges and that parts might fail.

The solar cells being proposed for use in these arrays are 8 x 8 cm. All of the plasma interaction data available to date is based upon 2 x 2 cm and 2 x 4 cm cells. Construction of the larger cells differ from the smaller, and the plasma interaction phenomenon may also differ. Testing of 8 x 8 cm cells in both ground simulation facilities and space flight experiments is strongly recommended before they are committed to space power systems. At this time, the impact of this interaction on system performance is rated as LARGE (4) due to the uncertainty in this technology.

4.2.2.3.2 Concentrator Solar Array Systems

Concentrator photovoltaic solar arrays have been suggested as an alternative to planar photovoltaic solar arrays, in order to reduce solar array area. As discussed in Scientific Report No. 1 (Section 3.4.2.3), the plasma collection process is limited to the light collector region of the concentrator solar cell. However, system aspects, of this type of interaction have not yet been evaluated. Data obtained shows that discharges still occur. These discharges may be more serious in this system since each cell could discharge separately. In planar cell arrays, a local discharge can relieve the stress in other parts of the array and reduce the probability of multiple discharges.

The effect of a discharge on a power system circuit using concentrator solar cells has not been evaluated. It is believed that the effects may be similar to those occurring on planar arrays. Based upon current knowledge, the impact of high-voltage interactions on system performance is rated LARGE (4).

4.2.2.3.3 Solar Thermal Dynamic Systems

Solar thermal dynamic power systems will probably generate AC power using an

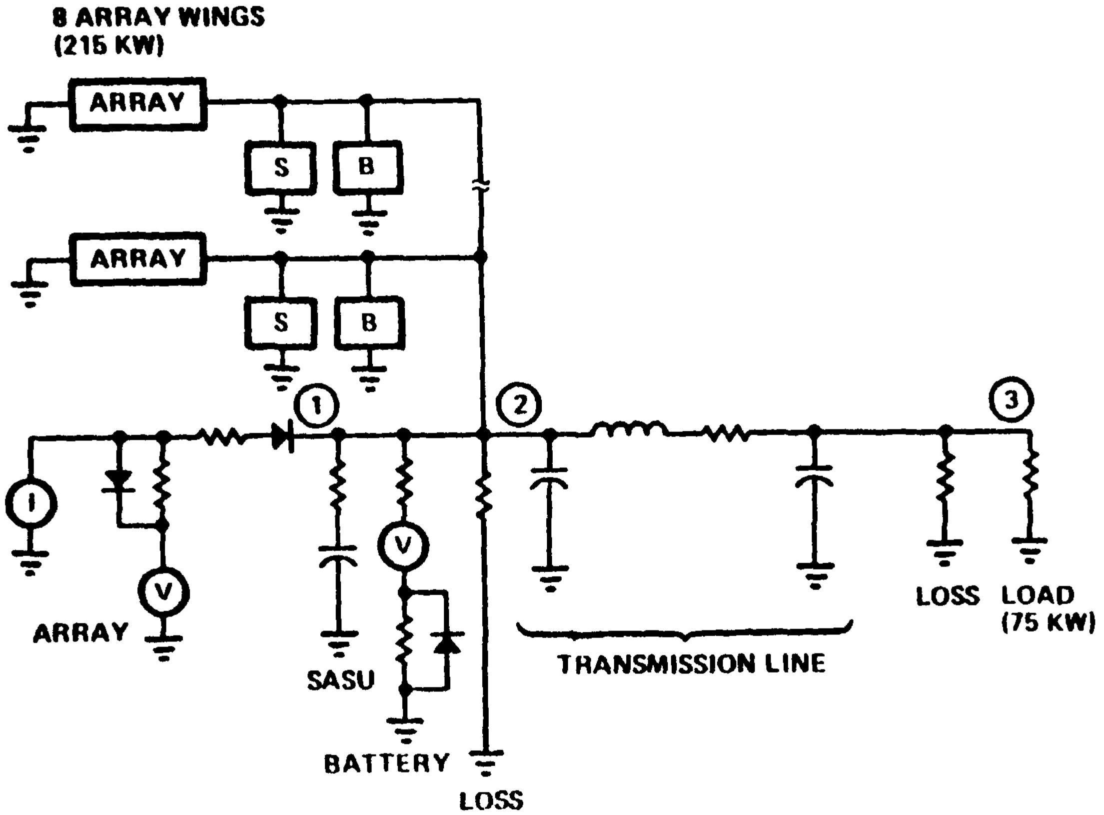
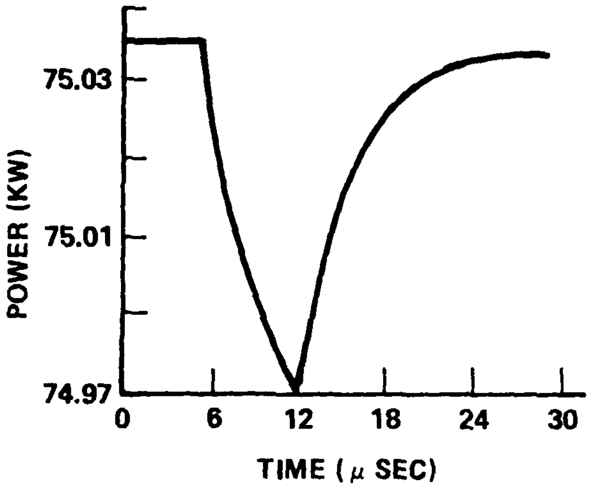
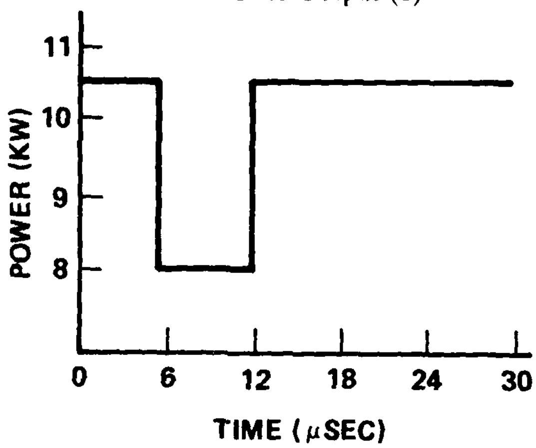


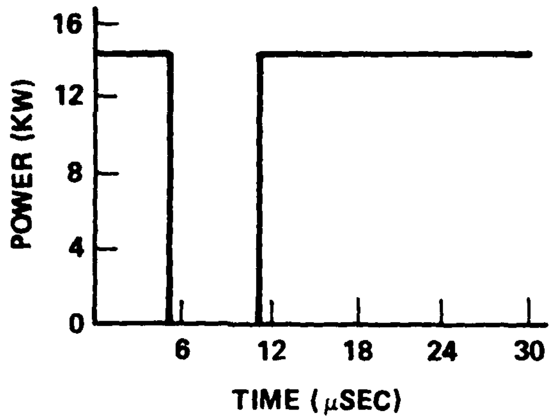
Figure 4.7 Discharge Transient Circuit Model (DC Generation/DC Transmission)



Blanket Power Output (1)



Wing Power Output (2)



Power Variation in Load (3)

Figure 4.8 Discharge Transient System Response alternator. The operating voltage could be as high as 440 V. In order to minimize high-voltage interactions, this should be a closed loop system (i.e. isolated from the plasma). However, if the system is grounded to the structure, any break in wire insulation would allow contact with the plasma. The behavior of an AC system exposed to the space plasma is not completely understood. Possible AC system transmission line interactions are discussed in the following section. Based upon the uncertainty of this type of system design, the impact of this interaction on system performance was rated MODERATE (3).

4.2.2.3.4 Transmission Effects

In large space stations, the power system can be configured as a low-voltage DC source. Power can then be converted to high-voltage AC for delivery to the load. This is similar to commercial power transmission techniques used on Earth.

The interaction possibilities are plasma-induced breakdowns (discharges) or power losses due to coupling. Plasma-induced breakdowns have been discussed before and can occur especially if the lines are exposed to the plasma (e.g. at connectors). The effect on the load of a discharge, occurring in an array and coupled by an AC transmission line, has been investigated. It was found that the array discharge transient would not couple into the system well enough to perturb the electrical characteristics of the load. Damage to power system components by repetitive transient inputs was not evaluated.

Power loss is due to the system coupling to the space plasma at the ion resonance frequency (*Scientific* Report No. 1, Section 3.4.2.5). Since suggested operating frequencies are in the ion resonance ranges of the plasma, coupling losses could result. This would represent a power loss to the system. No one has adequately addressed the impact on system performance. Based upon the uncertainties involved with this interaction, impact on system performance has been rated as LARGE (4).

4.2.3 High Energy Radiation Environment Interactions

4.2.3.1 *Radiation Damage*

In the 400 km polar orbit, the station will be subjected to high energy radiation over the polar regions and through the South Atlantic anomaly region. This radiation level can become significant over the proposed life of this mission. It should be noted that over the 30 year life of the station, it will experience at least two complete solar cycles and several large solar flare events. Thus, radiation levels can be severe.

Electronic components have to be protected from radiation total dose and dose rate interactions. This can be done with shielding. Techniques needed to mitigate harmful radiation effects on electronics are known. In addition, composite materials used in the station could also be affected by this environment and should be examined.

4.2.3.2 *Single Event Upsets*

In addition to solar radiation, the station will be subjected to cosmic radiation

(Figure 4.9). This can result in single event upsets. Parts selection or other mitigation techniques must be used to prevent unwanted electronic upsets.

4.2.1.3 *Radiation Hazard to Man-In-Space*

Radiation is a definite hazard for man in the polar environment. EVA equipment may protect man from the normal environment, but may not protect him from solar flare events. In this case, he must return to the protection of the station. Therefore, the station must be designed to provide a safe haven, not only for the electronics, but for man under the most severe radiation fluxes.

The system impact for all of these interactions has been rated as LARGE (4). However, the interactions can be tolerated with adequate design care. This category was discussed in detail in Scientific Report No. 1 (Section 4.4).

4.2.4 Neutral Environment Interactions

While these interactions were discussed in Scientific Report No. 1 (Section 5.0), the impact of Atmospheric Drag, Atomic Oxygen Surface Erosion, Surface Glow, and Sputtering interactions on a polar space station need further discussion.

4.2.4.1 *Atmospheric Drag*

Atmospheric drag is more significant for large spacecraft with large surface areas such as photovoltaic arrays, solar thermal dynamic collectors, or thermal radiators. The neutral environment is mainly comprised of atomic oxygen, helium, molecular nitrogen, atomic hydrogen, and argon. The neutral particle density at 400 km is approximately 3.6×10^{-15} gm/cm^3. In the ram direction, the density exceeds the ambient conditions due to the bow wave effect of a large structure "pushing" through the environment and building up a higher concentration of particles. As a large body moves through the neutral environment in LEO, it experiences a drag force given by:

$$F_D = 1/2\, C_D\, A\, \rho\, V^2$$

where: C_D = Drag Coefficient; ρ = Neutral Particle Density (m^{-3})
A = Cross Sectional Area (m^2); V = Spacecraft Velocity (m/sec)

In LEO, the neutral particle density and the spacecraft velocity will be relatively constant. Therefore, the spacecraft cross sectional area is the only variable which can be used to minimize drag. In a gravity gradient orientation in polar orbit, there are two extreme orientations for the station that would have different effects on drag. The noon-midnight orientation, shown in Figure 4.10A, would have the solar arrays or solar collectors creating a maximum drag effect with minimums at the equator. The dawn-dusk orientation is preferable from a drag perspective (Figure 4.10B), since it would always have the arrays or collectors at a minimum drag orientation.

The end result of atmospheric drag is orbital decay, which requires periodic firing

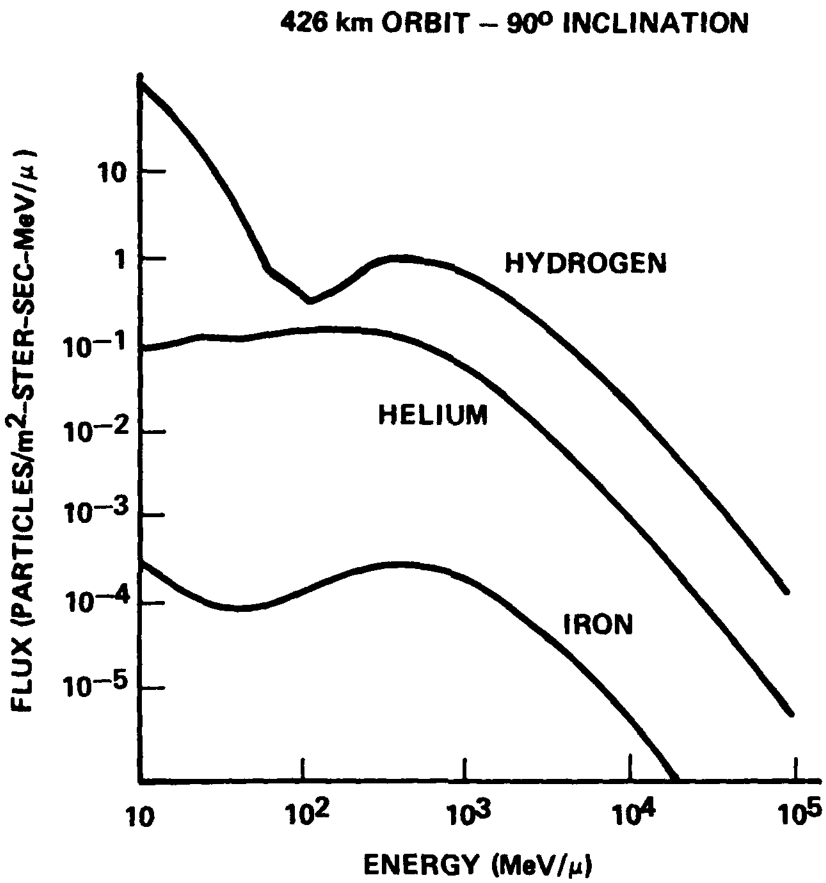
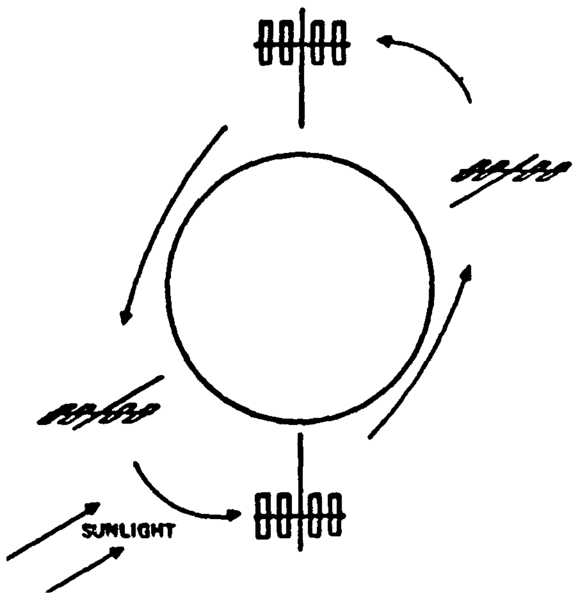
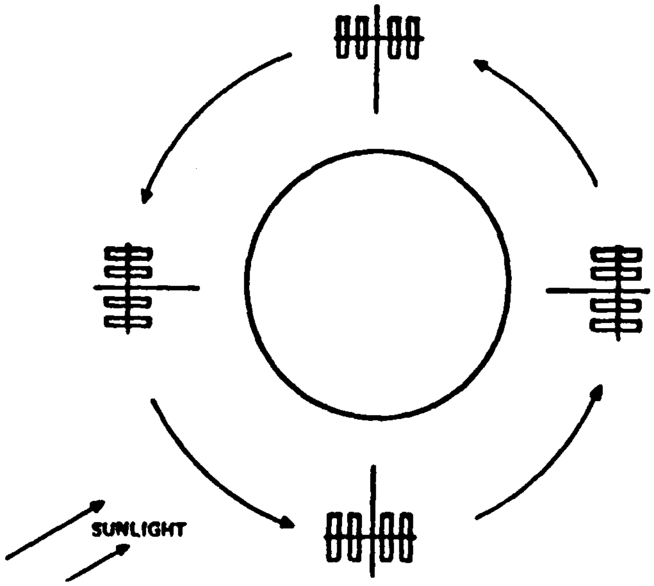


Figure 4.9 Maximum Cosmic Ray Environment: Free Field During Solar Minimum



(A) Maximum Drag at Noon and Midnight



(B) Minimum Drag Orientation

Figure 4.10 Atmospheric Drag

of booster rockets in order to maintain the same orbit. Spacecraft whose cross-section is not symmetric about its center of mass experience another effect of orbital drag; torque about the structures' center of gravity (Figure 4.11). This torque would be destabilizing without the use of on-board gyros and thrusters.

These drag interactions have been long recognized in orbiting spacecraft and are correctable. The system impact of these interactions is rated as CATASTROPHIC (5) since they could jeopardize the mission unless corrected.

4.2.4.2 *Atomic Oxygen Surface Erosion*

The neutral particle environment in LEO altitudes consists mainly of atomic oxygen ($\approx$ 80%) formed by photodissociation of molecular oxygen. At 400 km the oxygen atom number density varies from 2.5×10^7 cm^{-3} to 5.0×10^8 cm^{-3}. For surfaces facing in the ram direction, the effective oxygen atom energy is determined by the vehicle velocity and has a value of about 5 eV.

Material effects attributed to atomic oxygen interactions were seen on the first shuttle flight and identified as an erosion problem on the third flight. Since then, atomic oxygen has been the subject of several experiments on the shuttle. Experiments on flights STS-3 through STS-8 have demonstrated that many spacecraft materials experience surface erosion and adverse changes in optical and physical properties after exposure to the neutral environment.

Those materials that are most reactive to atomic oxygen include Kapton, Mylar, Kevlar, silver, osmium, and carbon. Table 4.1 shows the predicted surface losses of several materials. Changes seen during exposures ($\approx$ 40 hours) to the ram environment include: polymer samples changing surface morphology from a smooth to a rough surface; a change of optical properties from specular to diffuse surfaces; increased surface conductivities; and measurable mass losses.

Table 4.1
Atomic Oxygen Mass Loss Rates

Material	Surface Erosion (mils/yr)
Kapton	2.4
Mylar	2.8
Teflon	0.025
Silver	8.4
Aluminum	0.0003
Black Conductive Urethane	4.6
Chemglaze Z302	3.1
Carbon	3.1
Indium Tin Oxide	0.0003
Epoxy	1.9

Note: Assumed Flux = 2×10^{21} cm^{-2}yr^{-1}
Altitude = 500 km
Solar Cycle Maximum

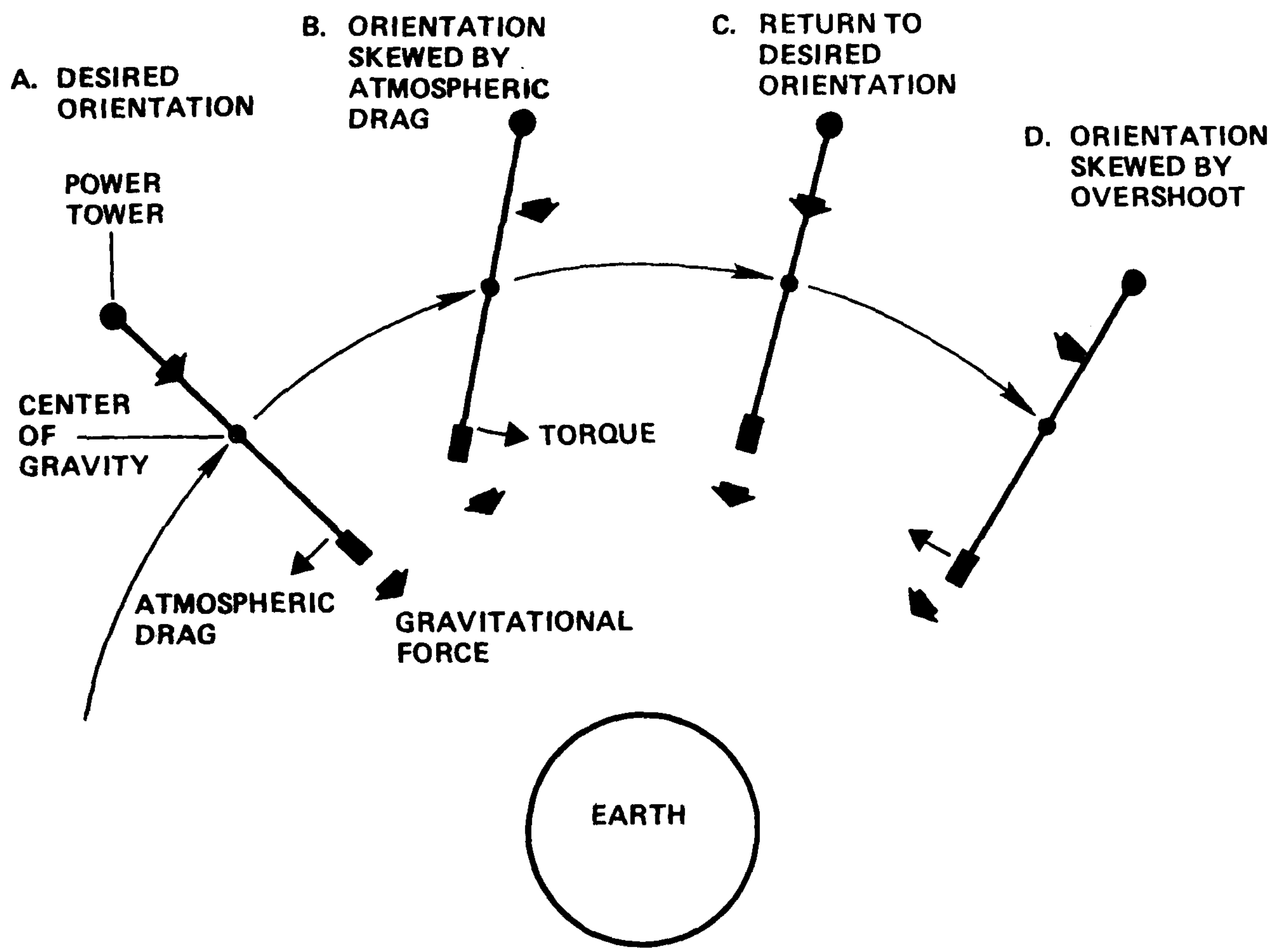


Figure 4.11 Drag Torques on Long, Unsymmetric Space Structures

One recent dramatic demonstration of atomic oxygen surface recession occurred on STS-8. There, a 0.5 mil disk of mylar that was exposed to a flux of 3.5×10^{20} atoms/cm^2 completely disappeared (Figure 4.12A). All that remained of the sample was the portion that was covered by the sample holder (Figure 4.12B). Analysis after the flight showed that some of the Mylar had deposited back onto the sample holder. Thus, atomic oxygen erosion is a source of contamination.

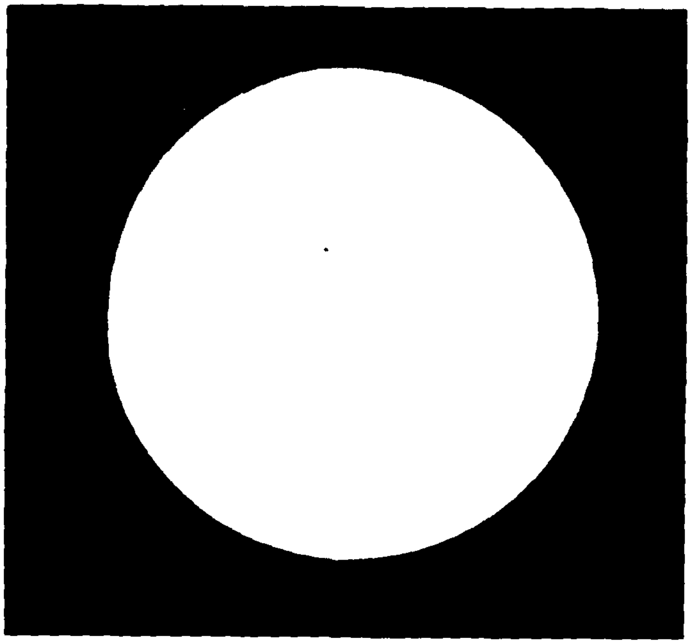
These mass losses and physical property changes are of particular importance to a space station, because of its many large surfaces and its expected long lifetime. The solar photovoltaic design will use large arrays of solar cells mounted to a flexible substrate. Typically, Kapton ($\approx$ 2 mils) is used as a lightweight substrate and is periodically exposed to the ram environment during the orbit. The interconnects between solar cells are made of thin silver alloy foil ($\approx$ 3 mils) and are also exposed to the ram environment. These interconnects are therefore very susceptible to atomic oxygen erosion effects.

The solar thermal dynamic concept will also have components damaged by atomic oxygen. The solar collector is be a highly polished surface of either aluminum or silver with a protective coating. Although aluminum seems to be non-reactive, a candidate coating (normally magnesium fluoride) can be reactive and will require additional testing. Silver, as shown in Table 4.1, is a highly reactive metal and will require a substantial protective coating that will not crack or allow the silver to be exposed to the oxygen environment. The solar collector will also be susceptible if protective coatings, which must remain functional over long periods of time at elevated operating temperatures, fail.

The effects of atomic oxygen surface erosion are based upon Shuttle experiments of limited exposure (< 40 hours) and small sample sizes. Extrapolation of these results to materials with larger surface areas must be done carefully. To date, the only large article returned from space has been a thermal blanket sample from the Solar Maximum spacecraft. As shown in Figure 4.13, the 2 mil Kapton outer layer is still intact, although discolored, after about 4.2 years in a 500 km orbit. The cut-out areas in the blanket are places where samples were removed for other studies. If this blanket was facing in the ram direction, the Kapton should have completely eroded, based upon extrapolation of data from previous Shuttle missions. This discrepancy has not been resolved.

Ideally, mitigation of atomic oxygen erosion would necessitate avoiding the use of reactive materials and orienting reactive surfaces away from the ram direction. Unfortunately, these solutions are not always possible and it is therefore necessary to use protective coatings such as metallic films, Teflon, metal oxides, or fluoro-polymers for organic films and composites. Glass resins can also be used on thermal control coatings and optical paints, but surface charging in the auroral regions may result.

Because these interactions are relatively new, much remains to be investigated. Based upon the possible severe surface erosion due to this interaction and the uncertainty of applications to large areas, the impact on system performance has been rated as CATASTROPHIC (5).



(A) Ram Exposed Mylar Disk

(B) Mylar Disk UV Control Sample

Figure 4.12 Mass Loss on Mylar Disk

Figure 4.13 Solar Maximum Thermal Blanket

4.2.4.3 *Surface Glow*

The shuttle glow phenomenon, first observed during the early shuttle flights, was observed to be a diffuse low-visual-intensity layer of enhanced luminosity on the leading edges of the vertical tail and aft engine pods (Figure 4.14). This glow has been measured on subsequent shuttle flights and was found to extend about 10 cm from surfaces and is predominantly in the 4200 to 8000 angstrom region. Surface glow could be intense enough (10^5 rayleigh) to interfere with optical sensors on-board the station.

Since this interaction would impact only instruments operating in the spectral region where glow is produced, a definitive list of instruments to be used on the station would be required before the system impact could be determined. Otherwise, this interaction would not apply.

4.2.4.4 *Sputtering*

Neutral particle sputtering of surfaces and coatings can occur in this concept. However, atomic oxygen erosion could mask this effect. Therefore, the role of sputtering is unclear and thus its impact is rated SMALL (2).

4.2.5 Particle Environment Interactions

The interactions in this environment category consist of bombardment of spacecraft by micrometeoroid and debris particles. These interactions were discussed in detail in Scientific Report No. 1 (Section 6.0).

Only interactions between this environment and the station power systems are discussed. For the planar solar array power system, the array area is assumed to be about 3000 m^2 with 6 mil solar cell cover glass. This thickness can be penetrated by micrometeoroids having a mass greater than 10^{-8} grams and by debris particles having masses greater than 10^{-7} grams. Using current models, this would indicate that about 35 micrometeoroids and 5 debris particles per square meter per year could penetrate the cover glass. For the thirty year mission, 3.6 million particle impacts could be anticipated (with about an order of magnitude uncertainty in the number).

Penetration would cause a spalling pattern that can result in the cover glass becoming opaque over an area at least forty times the initial impact area. This damage could result in the loss of one to three square meters of the total array. This should not cause a significant power loss. However, the particles could damage diodes or other components on the array wing. This can cause more serious performance degradations, the extent of which is uncertain.

For the solar thermal dynamic power system, it is assumed that the mirror reflecting surface is a thin coating with a protective overlay. Particles with a mass as low as 10^{-12} grams could penetrate these layers. Approximately 775 micrometeoroid and 9900 debris particle impacts per square meter per year can be expected. The required size of the reflectors and the thirty year mission duration suggests that approximately 640

Figure 4.14 Shuttle Glow on STS-3

million penetrations can be expected (with about an order of magnitude uncertainty). The surface damage estimates range from 0.05% to 10% due to impacts and resulting spalling patterns. This large range is a result of uncertainty in the damage area caused by these impacts.

Mirror damage is not limited to normal incident penetrations. Particles can strike at an angle causing long scratches in the mirror. In addition, protective coatings can be penetrated, exposing underlying materials to other interactions which can cause further deterioration of mirror performance. Finally, removal of the reflective coating can cause the mirror temperature to rise, further damaging the reflective capability of the mirror.

The effects of particle bombardments on concentrator arrays would be similar to those on solar thermal dynamic mirrors. Both of these systems require sun pointing accuracy and mirror reflectivity to function properly. Based upon the uncertainties in the particle models and the unknowns in the puncture interactions, this interaction must be rated as having a LARGE (4) impact on the station system performance.

4.2.6 Solar Radiation Environment Interactions

The interaction technology for this category was discussed in Scientific Report No. 1 (Section 7.0). These interactions can cause coating degradation, thermal forces, and hazards to astronauts in the space station scenario.

4.2.6.1 *Coating Degradation*

Thermal control coating degradation is another concern on large space structures with long exposure to the solar radiation environment. Ultraviolet radiation (UV) is primarily responsible for both surface and bulk damage to thin flexible coatings and paints that are used for thermal control. Typical effects of UV radiation on materials include outgassing, cracking, pitting, embrittlement, shrinkage, and discoloration. These effects can cause degradation of mechanical properties such as tensile strength, strain, and modulus of elasticity, as well as optical and thermophysical properties such as transmittance, reflectance, and absorptance. In addition, material electrical property changes are possible.

In all cases, although the study of coating degradation is quite mature, it will be necessary to account for these changes in material properties during the long lifetime of a station. Since this interaction can have a serious impact on system performance, if temperatures are not maintained within operating ranges, it was rated LARGE (4).

4.2.6.2 *Thermal Forces*

Distortion due to differential thermal heating is the primary interaction concerning thermal forces. This distortion can induce stresses in the structure that can lead to mechanical fatigue and microcracking in lightweight composites. Differential expansion can cause bowing in solar arrays or distortion in solar thermodynamic mirrors, thus causing power disruption.

The forces induced by solar radiation are well known, however, they must be accounted for in the station design. A MODERATE (3) system impact rating has been assigned to this interaction.

4.2.6.3 *Biological Hazard*

Since the space station will be manned, the structure must prevent solar radiation from harming personnel inside the station. Solar radiation effects on EVAs will be discussed in Section 4.3. This interaction is rated as having a SLIGHT (2) impact.

4.2.7 Self-Generated Environment Interactions

Self-generated environment interactions were discussed in Scientific Report No. 1 (Section 8.0). However, material and thruster effluent contamination and nuclear system interactions will be applied to the space station concept.

4.2.7.1 *Material Contamination*

In this discussion, contamination will refer to all foreign materials in orbit that can degrade the performance of critical spacecraft systems. Other sources of contamination which can affect spacecraft before launch will not be discussed here. The main sources of on-orbit spacecraft contamination are from: outgassing materials (from both external surfaces and internal sources), thruster effluent (from orbit insertion rockets, attitude control thrusters, and ion thrusters), and optical contamination (primarily from surface glow). Surface glow effects were discussed in Section 4.2.4.3 and will not be repeated here. These various types of contamination will have an increasing impact as instruments and sensors get more complex, system operations become more critical, and design lifetimes increase.

Material contamination results from particles and volatile, condensible materials that will outgas from warm, non-metallic adhesives, insulators, and paints under space conditions. Exterior sources include: thermal control paints, dielectrics, thermal blankets, and lubricants. Interior sources include: the spacecraft structure, electronic box coatings and components, and venting from crew quarters.

A major concern of material outgassing is that the volatile outgassing products will condense on cold surfaces such as thermal radiators, solar cells, and low temperature sensors. Material deposition changes surface properties, resulting in changes to operating temperatures and degradation of optical sensors. Although a material's outgassing rate slows with time, the amount of material that can be outgassed and redeposited on surfaces can be significant.

Although cold surfaces are the primary targets for outgassing products, a spacecraft charged by the environment can lead to electrostatically enhanced contamination over the entire spacecraft. Under this mechanism, outgassing vapors are ionized either by photoionization, collisions, or strong electric fields. The charged particles are then attracted and deposit on the charged surfaces. These deposits may remain after environmental charging has been neutralized. This phenomenon was studied as part of the Spacecraft Charging At High Altitudes (SCATHA) satellite and was found to have a significant impact.

Contamination is a configuration-dependent phenomenon, and its impact would have to be determined for a specific design with a specific payload. In general, it has a serious impact on spacecraft system performance and has been assigned a system impact rating of LARGE (4). If the payload includes infrared (IR) sensors, the impact rating would be raised to CATASTROPHIC (5).

4.2.7.2 *Thruster Effluent Contamination*

Thrusters operate by ejecting quantities of gases and, often, particulates (either liquid or solid). While most chemical thruster products completely escape from the system, small fractions do not and can return to the spacecraft's surface. The quantity of matter ejected, which can include both ionized and neutral particles, is so large that the small fraction that returns can be appreciable. Solid thrusters also present problems since they include aluminum to prevent unstable burning. Aluminum oxide particles formed during combustion can also deposit on spacecraft surfaces. Thus, the impact of thruster effluent on a space station system performance is also rated as LARGE (4).

4.2.7.3 *Nuclear System Interactions*

This interaction is not applicable to this concept, since it was assumed that there will be no nuclear systems on the station.

4.2.8 Electromagnetic Environment Interactions

Electromagnetic interactions result from the velocity of the space station through the Earth's magnetic field and current loops within the space station. This interaction was discussed in detail in Scientific Report No. 1 (Section 9.0). In this application, motion induced electric fields, current generated forces, magnetic torques, and induced electric field torques will be further discussed.

4.2.8.1 *Motion-Induced Electric Fields*

The station is assumed to have a conductive structure and will pass through varying magnetic fields, thus inducing oscillating electric fields within the structure. The electric field (the vector product of the velocity and magnetic field) peaks over one pole, has a value of zero at the equator, and reverses polarity over the other pole. Due to the small value of the magnetic field and the finite velocity, this field ranges between $\pm$ 0.37 V/m. For a 120 meter long structure, the maximum voltage induced is 0 to 44 volts (end-to-end).

While the electric fields are neither excessive nor cause for grave concern, this oscillating behavior of the electric field can effect plasma phenomena. It will influence the floating potential of the structure (relative to the space plasma potential) and this could disturb sensor measurements. The plasma coupling for a high voltage solar array would be influenced by this oscillating structure voltage if the array is tied to the structure. This might make the array sufficiently negative to induce arcing.

Generally, this electric field has not caused serious impacts. Previous satellites have been too small to generate sufficiently large electric fields. A Shuttle experiment did observe an 8 volt distribution from tip to tail, corresponding to the vehicle $\mathbf{V} \times \mathbf{B}$ field. Extrapolation of this data to station-sized objects indicates a need to be concerned about this interaction and the effects it may induce. This interaction has been rated as having a MODERATE (3) system impact.

4.2.8.2 *Current-Generated Forces*

Great care is usually taken to prevent the generation of current flow magnetic fields on satellites. These precautions are necessary to prevent interactions between the Earth's magnetic field and the satellite induced fields. However, with the advent of a large, high-powered station, there are additional concerns that must be addressed.

Consider the solar array power system shown in Figure 4.15. The circuits are laid out in a serpentine pattern so that the magnetic fields cancel. Yet in an overall perspective, the circuit forms a current loop. A proposed design calls for 16 cells in series per block, with 26 blocks in a circuit. There will be 36 circuits per blanket. The physical size of this system creates a large loop that can interact with the Earth's field. This produces a small torque ($\approx$ 0.6 μN $\cdot$ m) rotating the station in one direction over the North Pole and the opposite over the South Pole. In a conceptual design of the NASA Space Station, there were 16 identical blankets, causing the current loops in the upper and lower blankets to be in opposite directions, thus producing opposite torques. The concern here is not that the individual torques are prohibitive, but that the number of torques are large and the total influence must be considered.

The current flowing in this system will be large. This means that there will be forces pushing the wires apart. In the solar array shown in Figure 4.15 all the circuits are in parallel, generating an increasing current from bottom to top. The separating forces range from about 1.5 μN at the bottom to about 1 mN at the top. This would be identical in all blankets.

The current flowing from the array to the station could be as high as 1000 amps. This would produce separation forces on the order of 1.0 N. In addition, if these currents are not balanced, the flow could interact with the Earth's magnetic field and thus increase drag. Individually, these forces are probably not large enough to cause concerns, but collectively they may. Thus, the impact of these interactions on system performance is rated as LARGE (4).

4.2.8.3 *Magnetic Torques*

Magnetic torques can also be caused by the solar array layout. A proposed solar array configuration, shown in Figure 4.15, calls for a number of cells in series to form a

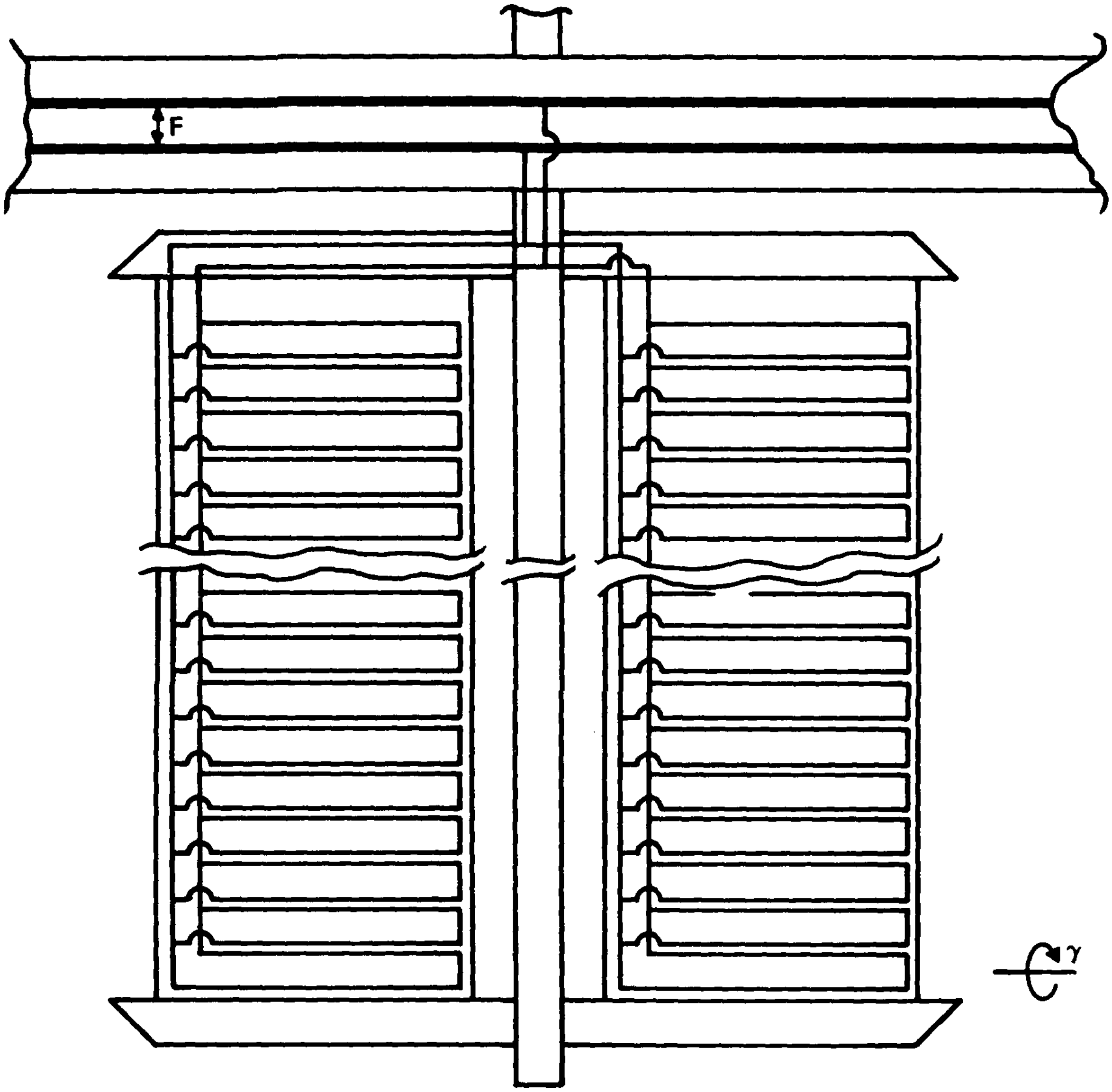


Figure 4.15 Motion and Current Induced Effects in a Solar Array Power Source

block. A number of blocks in series are required to generate the desired operating voltage. Enough voltage blocks are combined in parallel to produce the current for the design power level. The physical size of this system creates a large loop that can interact with the Earth's magnetic field. There would be a small torque associated with each voltage block ($\approx 0.6\ \mu N \cdot m$) trying to rotate each loop on the flexible substrate. This rotation would vary as the magnetic field changes during the orbit.

If the current flows are not matched in the wings on both sides of a station, a magnetic torque can exist which would tend to rotate the station. The separation between the wings is large (on the order of 75 meters in this station concept) and the blanket has many current loops which can cause imbalances. The impact of these torques on system performance has been rated as LARGE (4).

4.2.8.4 *Induced Electric Field Torques*

If the station is modeled as an elongated object, it is possible that charge distributions could be imposed on the non-symmetric ends, creating an electric field torque ($\rho \times \epsilon$). This effect has been investigated and was found to be relatively small ($\approx 0.4\ \mu N \cdot m$). However, oscillation period is related to the orbit frequency, thus there is the possibility that this torque could match a structure resonant frequency, producing an uncontrollable oscillation. This oscillation could be initiated by overshooting drag corrections and possibly enhanced by electric field torques. Due to the uncertainties in this interaction, an impact rating of MODERATE (3) has been assigned for its effect on system performance.

4.3 ASTRONAUT LIFE SUPPORT/MANEUVERING EQUIPMENT

4.3.1 Concept Description

This concept involves the Astronaut Extravehicular Activity (EVA) equipment. The equipment consists of the Extravehicular Mobility Unit (EMU), the Primary Life Support System (PLSS) and the Manned Maneuvering Unit (MMU) (Figure 4.16). While this equipment allows the astronaut to be temporarily self-contained and completely free from the parent vehicle, it also makes him an independent spacecraft subject to environmentally induced effects. Thus, the EVA equipment may be susceptible to electronic upsets when not used in low altitude, equatorial orbits.

While the equipment is built with reliability in mind, there is always room to improve system performance. The astronaut EVA equipment is a relatively small system, whose exposure to the space environment is short compared to other concepts in this applications study. However, when in the dynamic polar-auroral region, environmental interactions unique to EVA equipment should be expected.

4.3.2 Plasma Environment Interactions

These technologies were discussed in Section 3.0 of Scientific Report No. 1. EVA equipment polar-auroral charging and high-voltage interactions will be discussed here.

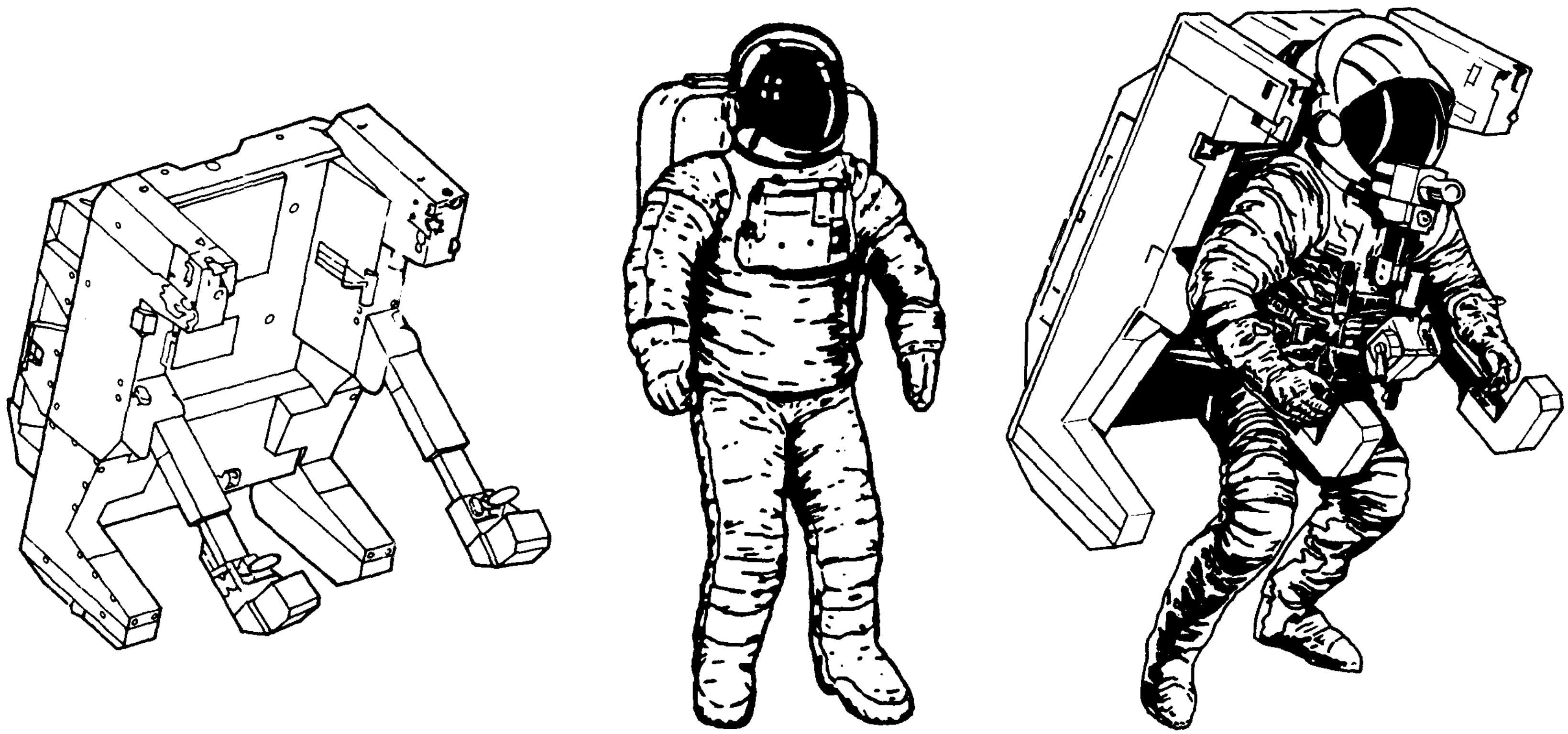


Figure 4.16 Astronaut EVA Equipment

4.3.2.1 *High Altitude Spacecraft Charging*

For the EVA equipment concept, this interaction is not applicable since there are presently no plans for astronauts to perform EVAs at high altitudes.

4.3.2.2 *Polar-Auroral Spacecraft Charging*

In low altitude, high inclination orbits two aspects of this interaction must be considered: charging of the astronaut EVA equipment by polar-auroral beams, and the interaction between the astronaut and any large structure being serviced. As stated previously, the astronaut is like a small spacecraft which may interact with the environment like any other spacecraft. The equipment presently being used is predominantly covered with dielectrics and some exposed metallic pieces.

In an auroral event, the suit can become charged and discharges may occur. Such discharges in EVA suit materials have been observed in laboratory simulation tests. If discharges occur in space, then the transients induced could couple into the electronic system of the EVA equipment, the Shuttle, or a spacecraft being serviced. This transient coupling is more serious if fast, computer-level logic systems are incorporated into future equipment designs. Preliminary evaluations of the present EVA equipment are currently underway. All that can be stated at this time is that the charging mechanism exists in this orbit and that laboratory simulation tests indicate that the equipment surfaces can be charged to the point where discharges occur.

The second aspect of this interaction is multiple body charging (i.e. the differential charging between the astronaut and a near-by large structure, or between any two spacecraft). An evaluation of this differential charging phenomenon is currently underway and the initial results indicate that serious interactions could occur.

For this investigation an astronaut encounter with the Shuttle in polar orbits is used to model the interactions impact. Polar-auroral charging has been shown to be size dependent so that if each object were subjected to the same storm, the Shuttle would charge to a larger negative voltage than the astronaut. If the astronaut is in the Shuttle wake, then he, being in an ion depletion region, could charge more negatively than the Shuttle. This is illustrated in the POLAR code simulation of an astronaut in the Shuttle wake, shown in Figure 4.17.

Docking maneuvers under these conditions would be hazardous until the charging had been neutralized. All of these speculated effects are potentially serious, contain unknowns, and should be evaluated. The impact of polar-auroral charging is rated LARGE (4).

4.3.2.3 *High-Voltage System Interactions*

If the large structure has an operating high-voltage system, such as a solar array that the astronaut is expected to service or repair, then the interactions between the astronaut and the array must be evaluated. This would be analogous to a lineman on

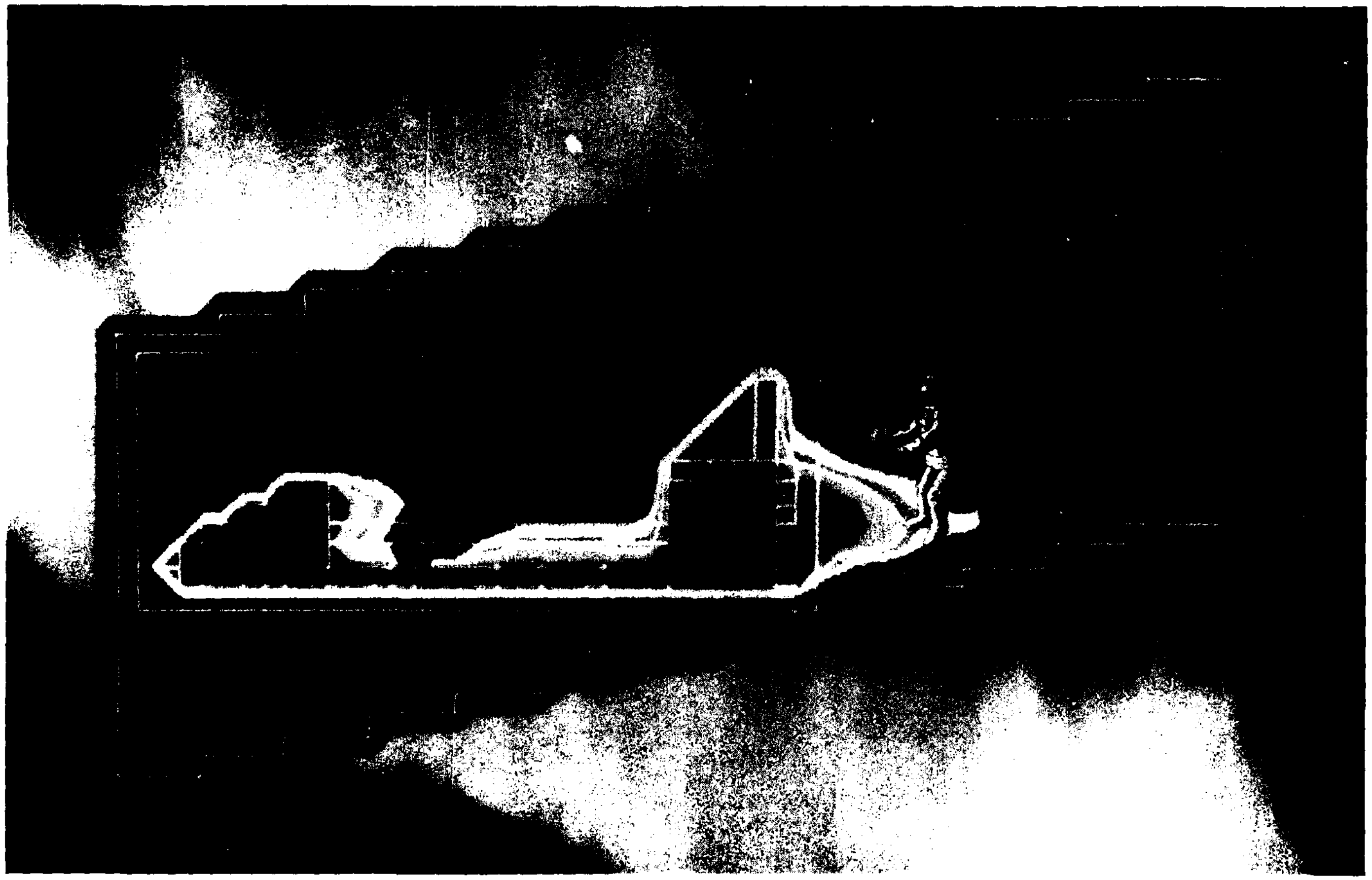


Figure 4.17 Polar Orbit EVA

Earth having to repair high-voltage transmission lines. In space, it is not easy to shut down a high-voltage array. The assumption that the array could be simply rotated out of the sunlight for repairs may not be adequate.

Care must be taken to insure that there is no source of incident energy onto the array. Earthshine and albedo combine to produce about one half the sun's intensity and, while the spectral match is not identical, the array can still generate a fraction of its power at design voltages. For large arrays envisioned in future applications, this voltage could be adequate to affect the EVA equipment. Thus, repair missions need to be conducted carefully. Therefore, the impact on the EVA equipment for this interaction is rated LARGE (4).

4.3.3 High Energy Radiation Environment Interactions

Discussion of these interactions, given in Section 4.2.3, is applicable to this concept. The only differences between EVA equipment and a space station is physical size and length of exposure to the environment. An additional factor to be considered here is the biological effects of radiation on the astronaut.

4.3.4 Neutral Environment Interactions

Effects due to the interactions associated with the neutral environment on astronaut EVA equipment are NEGLIGIBLE (1), since the equipment is small and the exposure time to the environment is short.

4.3.5 Particle Environment Interactions

Both micrometeoroid and debris impacts are considered together in this discussion. There is a small, but finite probability that an astronaut will be hit by a particle large enough to penetrate his suit. The expected number of impacts, of sufficiently large particles, is on the order of 0.1 per m^2 per year. Since the astronaut has a cross sectional area of about 1.5 m^2 and has a mission duration of hours before returning to the support station, he could expect about 1×10^{-4} hits from particles large enough to puncture his suit. Even if one punctured the suit, he should be able to return to safety. Smaller particles could pit his helmet, however this would be more of a nuisance than a hazard. Due to these small probabilities, the impact of these interactions on the astronaut EVA equipment is rated as SLIGHT (2).

4.3.6 Solar Radiation Environment Interactions

4.3.6.1 *Coating Degradation*

The effects of this type of interaction are NEGLIGIBLE (1), due to the short exposure times of EVA missions.

4.3.6.2 *Thermal Forces*

Effects due to this type of interaction are also NEGLIGIBLE (1), due to the small size of the astronaut EVA equipment.

4.3.6.3 *Biological Hazard*

The primary biological hazard to the astronaut is solar radiation absorbed through the helmet visor. This opening is protected by a solar shield that limits the entering radiation to a tolerable level. Other than accidental exposure from unexpected reflections, effects due to this interaction should be minimal. The potential impact is rated as SLIGHT (2).

4.3.7 Self-Generated Environment Interactions

Effects resulting from these types of interactions are NEGLIGIBLE (1) due to the relatively small size of the equipment and the short duration of EVA missions. It is assumed that potential damage during servicing missions can be avoided.

4.3.8 Electromagnetic Environment Interactions

Finally, electromagnetic interaction effects are also NEGLIGIBLE (1) due to the small size of the EVA equipment and the low power levels used.

4.4 SPACE PLATFORM IN GEOSYNCHRONOUS ORBIT

4.4.1 Concept Description

The concept used here is comprised of a fairly large spacecraft having two 12.5 kW solar arrays to power a variety of communications and sensor systems (Figure 4.18). Such a platform would be on the order of 80 meters across the array wings and about 10 meters wide. A sizable radiator would have to be included to dissipate the waste heat from the payload. This radiator would be on the order of 5 x 28 meters and probably contain a fluid loop.

It is assumed that this platform would be in geosynchronous orbit with a planned life of at least 15 years. The details of the payload are imprecise, but a prime function of this platform may be communications.

4.4.2 Plasma Environment Interactions

4.4.2.1 *High Altitude Spacecraft Charging*

This interaction occurs when the space platform encounters a geomagnetic substorm environment. The substorm deposits charges on and in the surfaces, producing negative voltages in the platform relative to the space plasma potential. These voltages can reach a threshold where discharges occur. This produces transients that couple into

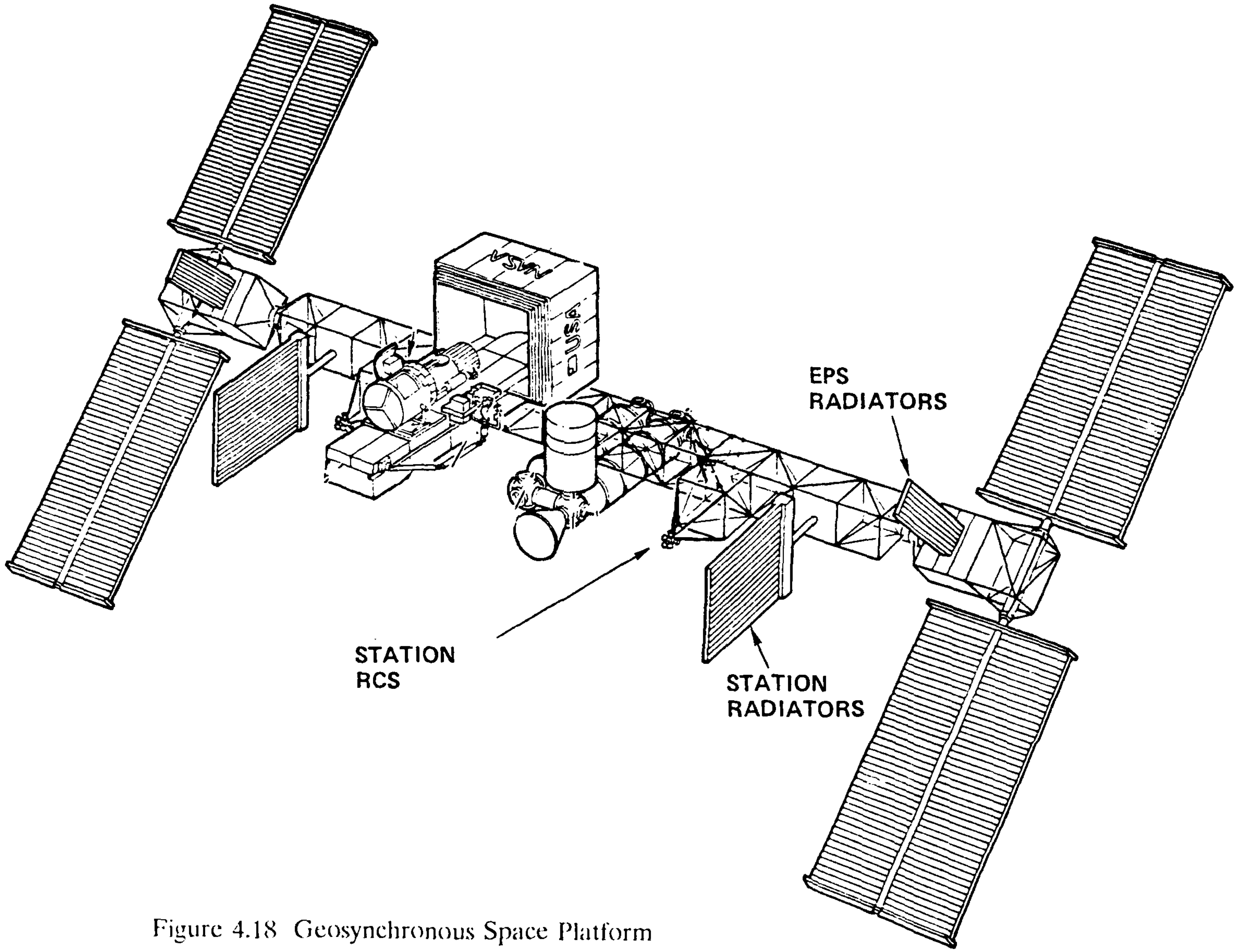


Figure 4.18 Geosynchronous Space Platform

spacecraft electronic systems, causing anomalous switching or component failures.

External surface charging on this type of spacecraft can be predicted by using the NASA Charging Analyzer Program (NASCAP) with the below environment parameters (Table 4.2). Figures 4.19, 4.20, and 4.21 show the NASCAP charging results of a three-axis stabilized platform in a severe substorm environment (see Section 3.2 of Scientific Report No. 1 for more information on this analysis).

Table 4.2
Recommended Geomagnetic Substorm Design Environment

Electron Density	$1.12\ cm^{-3}$
Proton Density	$0.236\ cm^{-3}$
Electron Temperature	12.0 keV
Proton Temperature	29.5 keV
Electron Current Density	$0.33\ nA/cm^2$
Proton Current Density	$2.5\ pA/cm^2$

The first figure, Figure 4.19, is the NASCAP computer model. Figure 4.20 illustrates the voltage distributions around the spacecraft. Likely regions for discharges occur where the voltage lines are concentrated. These suspect regions are the parts of the spacecraft body and solar arrays shaded from sunlight. Solar array discharges would result from interconnects being more negatively charged than the surrounding cover glass. Shaded area breakdowns would result from strong negative voltage gradients at dielectric edges. The final figure, Figure 4.21, indicates the expected structure potential for charging in sunlight conditions.

In addition to surface charging, charges deposited within dielectrics can also discharge. There is no criteria established for breakdowns, other than the existence of electric fields greater than 2×10^6 volts/cm. The amount of charge stored and released in a discharge has not been quantified.

When a discharge occurs, a transient is generated which apparently couples into the spacecraft electronics causing upsets. Some evidence suggests that components can fail under severe transients or after repeated events. The interaction uncertainty is how the transient couples into electronics. This appears to happen in space, but both analytical and ground simulation results have yet to demonstrate adequate understanding of the phenomenon.

The tendency to use faster integrated circuits and automated systems in future large platforms may increase these interactions effects. Discharge transients coupling into automated systems could trigger automatic responses which may result in attitude control changes, system failures, communication shutdowns, or loss of mission. Such

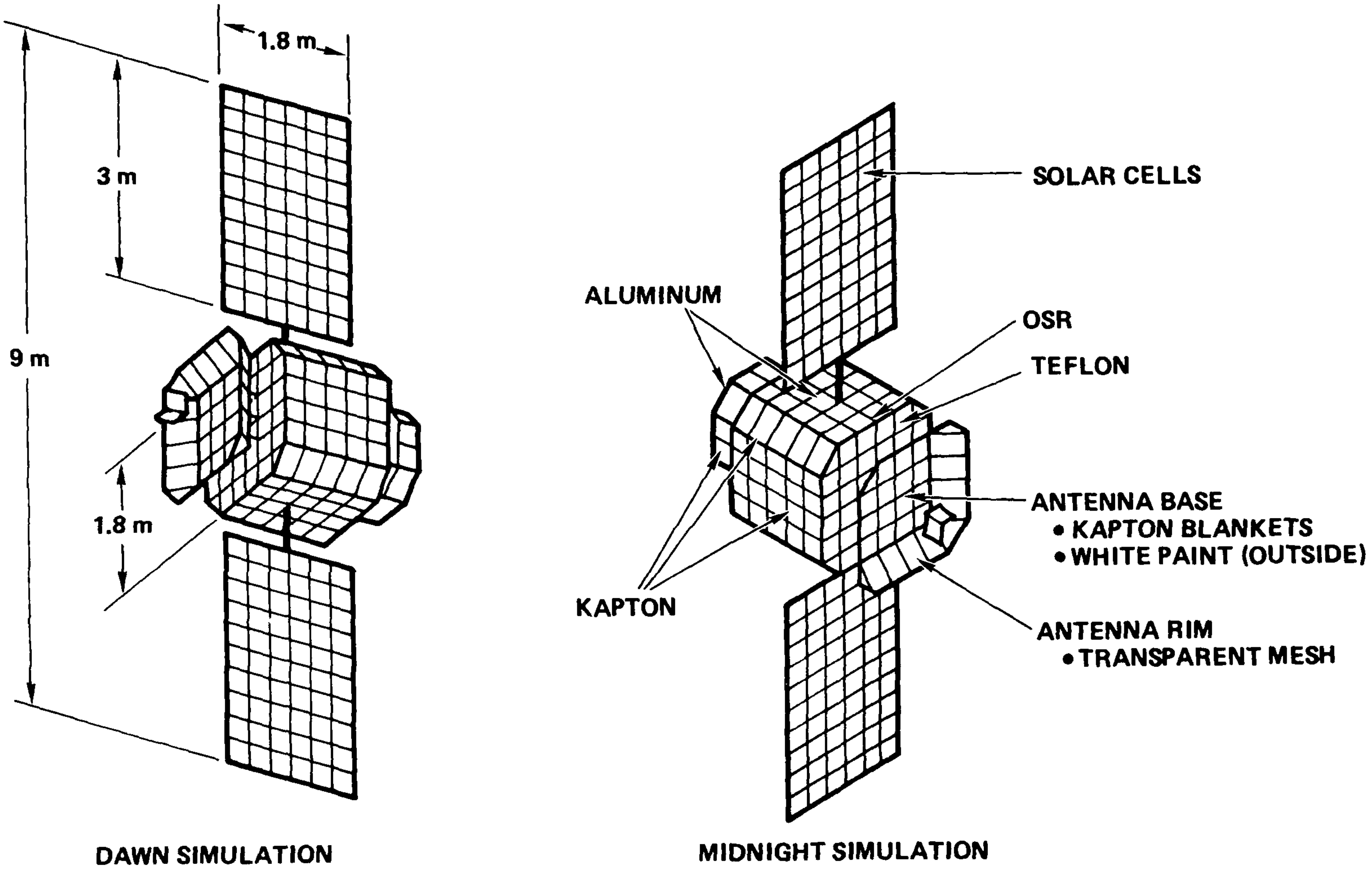


Figure 4.19 NASCAP Model of a Three-Axis Stabilized Spacecraft

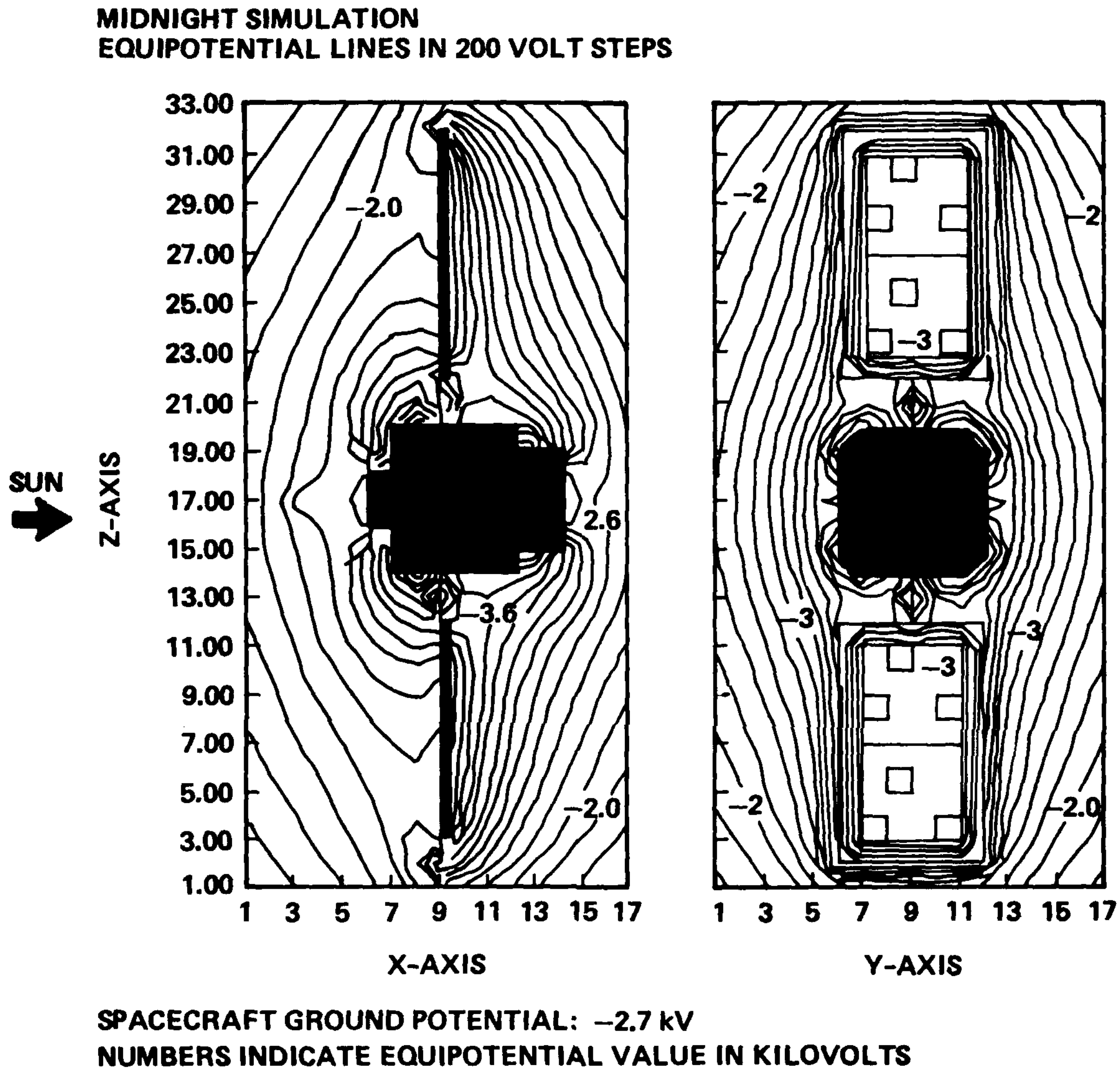


Figure 4.20 Predicted Voltage Distributions

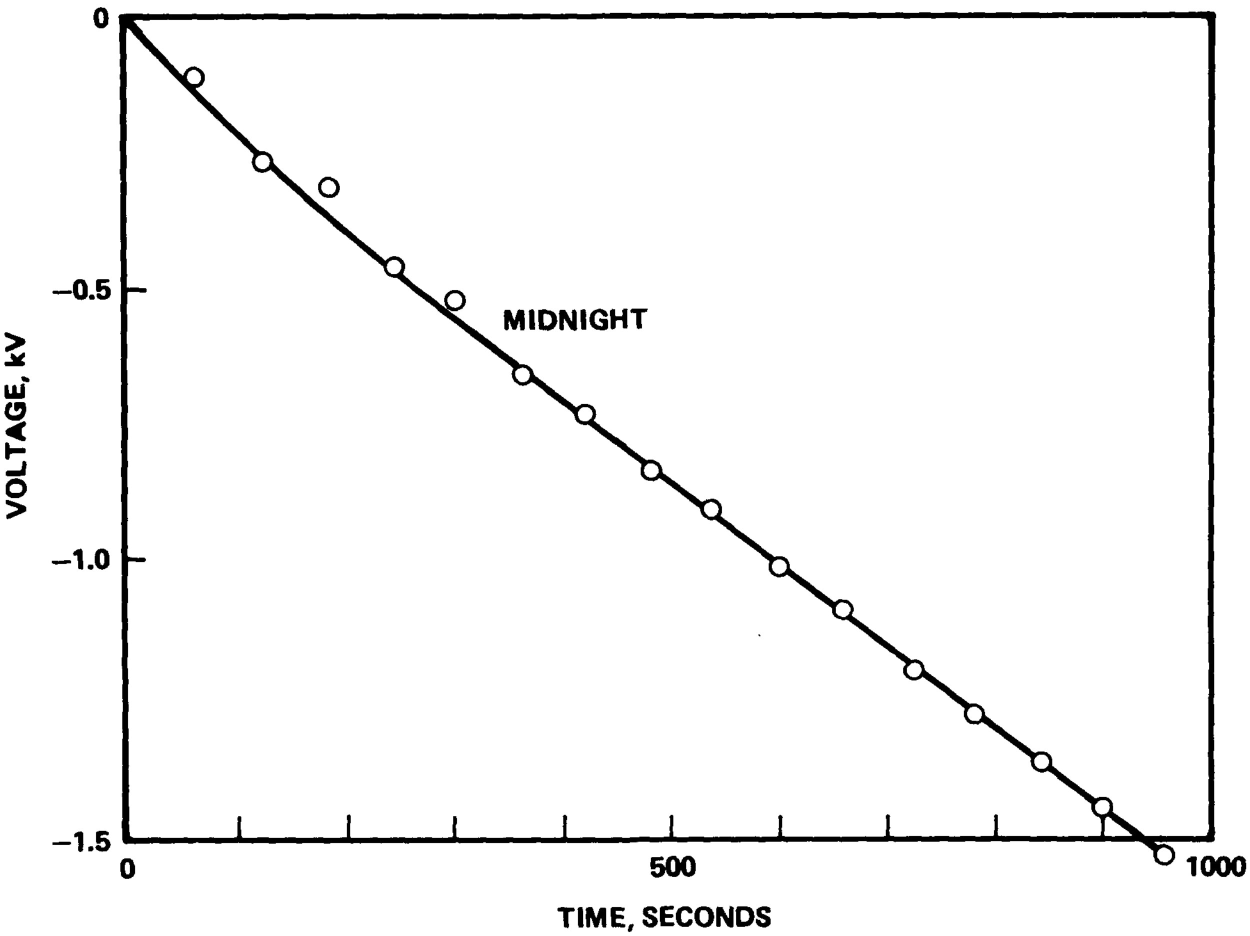


Figure 4.21 Charging History of Spacecraft Ground

occurrences have already been encountered. Based on these concerns, the impact for this interaction is rated as LARGE (4).

4.4.2.2 *Polar-Auroral Spacecraft Charging*

This interaction has no effect on this concept, since the platform will operate in geosynchronous orbit.

4.4.2.3 *High-Voltage System Interactions*

This interaction depends upon the low energy, thermal plasma density. Since the density in geosynchronous orbit is on the order of one particle per cubic centimeter, the consequences of the interaction are minimal. Breakdown thresholds, under negative voltage conditions for this environment, are estimated to be on the order of -10 kV. Presently there is no need for this level of operational voltages, thus the impact must be rated as SLIGHT (2).

4.4.3 High Energy Radiation Environment Interactions

The interactions included in this environment category were discussed in Section 4.2.3. This discussion is equally applicable for this concept with the exception that, in this orbit, the Earth's magnetic field will not shield the spacecraft from the solar flare and cosmic ray environments. Solar flare and cosmic ray environments were defined by the free-field fluxes given in Scientific Report No. 1 (Section 4.1). Hence, the radiation environment in this orbit will have greater effects than in low-altitude polar orbits. The potential impact for all of these interactions is rated as LARGE (4).

4.4.4 Neutral Environment Interactions

The interactions in this environment category are not applicable because they are not significant at geosynchronous altitudes.

4.4.5 Particle Environment Interactions

Only micrometeoroid impact interactions are considered in this category. While there may be debris at geosynchronous altitudes, the particle densities are unknown. Hence, the impact of debris bombardment cannot be evaluated.

Each solar array wing can expect to be hit by approximately 65,000 times in its 15 year life by micrometeoroids with sufficient energy to penetrate a 6 mil cover glass. The damage to the array should amount to about 0.01% of the total area. Thus, the probability of an impact to a sensitive component in this concept is small. The potential impact for this interaction is rated as SLIGHT (2) and should be reviewed when geosynchronous debris models are developed. For more information, see Scientific Report No. 1 (Section 6.0).

4.4.6 Solar Radiation Environment Interactions

The effects of the interactions defined under this environment category have been discussed in Section 4.2.6 and are applicable to this concept.

4.4.7 Self-Generated Environment Interactions

The effects of the interactions defined under this environment category have been discussed in Section 4.2.7 and are also applicable to this concept.

4.4.8 Electromagnetic Environment Interactions

Electromagnetic interaction effects are not applicable to this concept, since the Earth's magnetic field is negligible at geosynchronous altitude.

4.5 NUCLEAR SPACE POWER SYSTEM

4.5.1 Concept Definition

The proposed SP-100 power system design (Figure 4.22) was used for this concept. This system is designed to generate 100 kW of power with a reactor system located about 25 meters from the payload. The reactor is assumed to have a diameter of one meter. The heat rejection radiators for the reactor are on the order of 100 m^2. The shielding for the payload is proposed to be 50 cm thick for unmanned missions and 120 cm thick for manned applications.

Since the payload for this concept is not defined, it was assumed to consist of communications and sensor electronic packages. In addition, it was assumed to operate in a low altitude polar orbit. The expected radiation levels for a payload designed for a 7 year mission are 5×10^5 rad total dose and 10^{13} cm^{-2} neutron flux. The reactor will account for both of these levels, however, the natural space environment will contribute to total dose.

4.5.2 Plasma Environment Interactions

4.5.2.1 *High Altitude Spacecraft Charging*

This interaction is not applicable in polar orbits.

4.5.2.2 *Polar-Auroral Spacecraft Charging*

The effects of this interaction were discussed in Section 4.2.2.2 and those comments are applicable here. This interaction impact was rated LARGE (4).

4.5.2.3 *High-Voltage System Interactions*

Operation of the power system at voltages up to 400 V raises concerns about

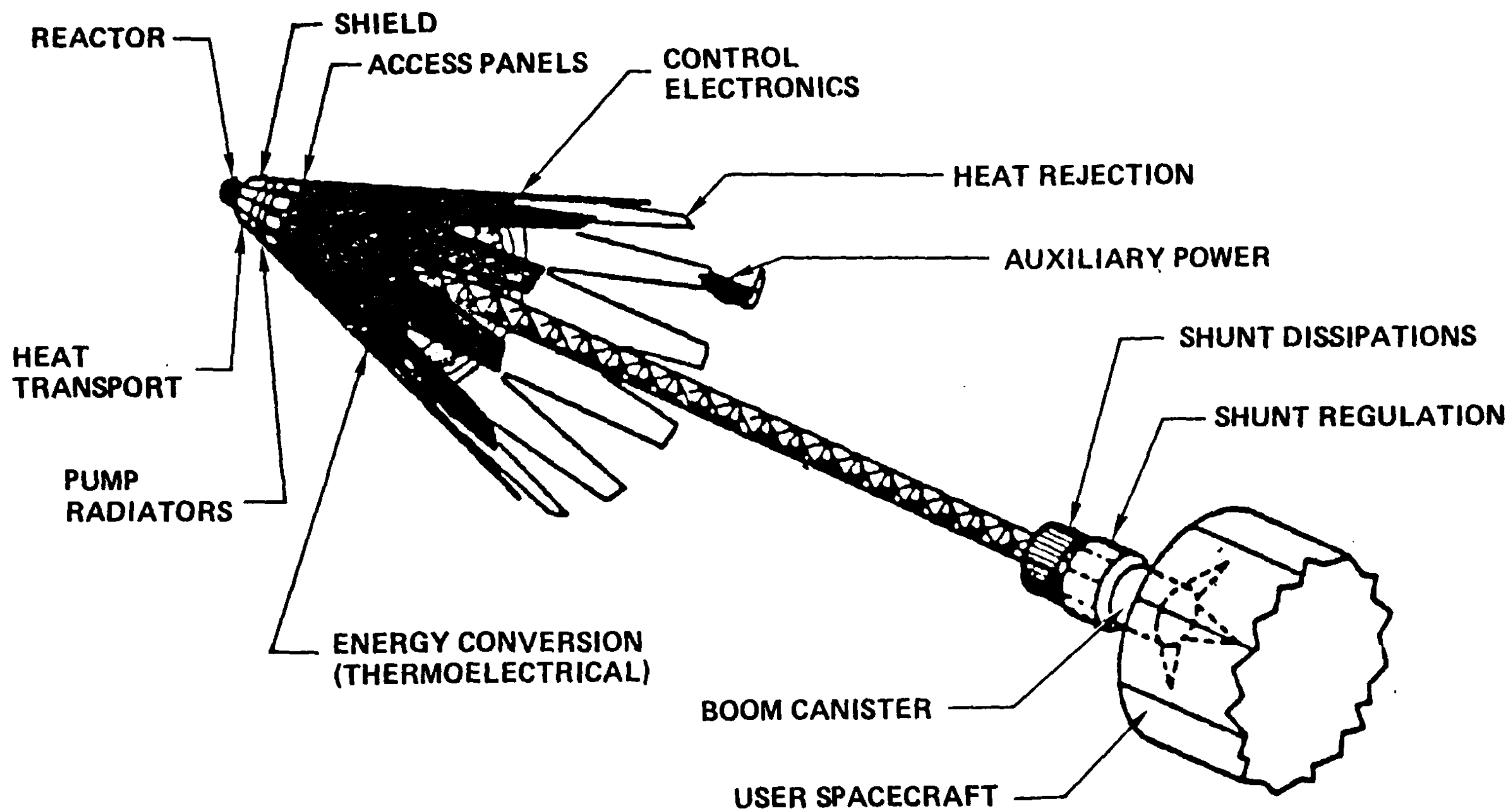


Figure 4.22 Nuclear Power System

possible interactions with the plasma environment. These interactions would be similar to those described in Section 4.2.2.3 for the solar thermal dynamic power option of the space station.

Power leads and connections can be insulated from the plasma environment, however, defects are possible. The insulation integrity could be compromised by high energy radiation, neutral particle erosion, or micrometeoroid/debris particle impacts. Whatever the cause, once the insulation is breached, interactions can occur. The effect of discharge transients on these power system components has not been evaluated. Thus, the impact of this interaction is rated a LARGE (4).

4.5.3 High Energy Radiation Environment Interactions

The effects of these interactions were discussed in Section 4.2.3 of this report and the discussion is applicable to this concept. However, the total dose accumulation must include the dose from the reactor (Section 4.5.7.3). The impact rating for these interactions are LARGE (4).

4.5.4 Neutral Environment Interactions

These interactions were discussed in Section 4.2.4 and are applicable to the nuclear power system concept. The impact ratings are the same as those for neutral environment interactions in Section 4.2.4.

4.5.5 Particle Environment Interactions

The effects of these interactions were discussed in Section 4.2.5. While that discussion is applicably here, thicker materials must be used to shield the payload and contain high operational temperatures. The use of thicker payload shielding will significantly reduce the number of penetrating particles. Hence, this interaction impact is rated SLIGHT (2).

4.5.6 Solar Radiation Environment Interactions

Solar radiation effects were discussed in Section 4.2.6 of this report and are applicable to this concept. The high temperature required for reactor operation increases the concern for induced thermal stresses. However, these forces should be amenable to standard design practices. Thus, the impact rating is MODERATE (3).

4.5.7 Self-Generated Environment Interactions

4.5.7.1 *Material Contamination*

The discussion of these interaction effects in Section 4.2.7.1 of this report are applicable for this concept.

4.5.7.2 *Thruster Effluent Contamination*

The effects of these interactions were discussed in Section 4.2.7.2 of this report and are also equally applicable to this concept.

4.5.7.3 *Nuclear Systems Interactions*

Shielding from reactor fission products is needed, however, it may not provide complete protection for all components. A summary of radiation tolerances for various parts is shown in Figure 4.23. As can be seen in this figure, CMOS, NMOS, PMOS and SOS/SOI devices would fail under the design total dose environment, while BIPOLAR, CMOS (RAD HARD), I^2L and SOS/SOI (RAD HARD) devices would be questionable. Thus, if these sensitive devices were required for use in a payload, additional radiation shielding would be necessary.

For the neutron flux specification, all devices are safe with only I^2L devices in the questionable range. This is fortunate since it is more difficult to protect these devices from neutrons. Based upon these concerns, the potential system impact for this interaction has been rated as LARGE (4).

4.5.8 Electromagnetic Environment Interactions

The effects of these interactions were discussed in Section 4.2.8 and are applicable to this concept.

4.6 ORBITAL MANEUVERING/TRANSFER VEHICLE (OMV/OTV)

4.6.1 Concept Description

The OMV/OTV spacecraft concepts are essentially space tugs to be used to move other spacecraft to various orbits, retrieve spacecraft, or service them. As such, they are required to operate in orbit for a minimum of 10 years. These vehicles, really two different spacecraft, are relatively small (4.5 meters in diameter by about 1 meter high). Figure 4.24 is an artist's concept of the OMV. The OTV is assumed to be similar.

The OMV was originally considered as a vehicle that would be limited to altitudes below 1000 km. Its role was to ferry objects from a Space Station to higher or polar orbits and return. The OTV was originally conceived as the vehicle that transferred objects from low to geosynchronous orbits and return. Presently, it is believed that these two concepts will merge. These tugs will operate primarily in automatic modes, moving from one location to another using their own guidance, sensor, and propulsion systems. The final delivery or retrieval phases would be man controlled.

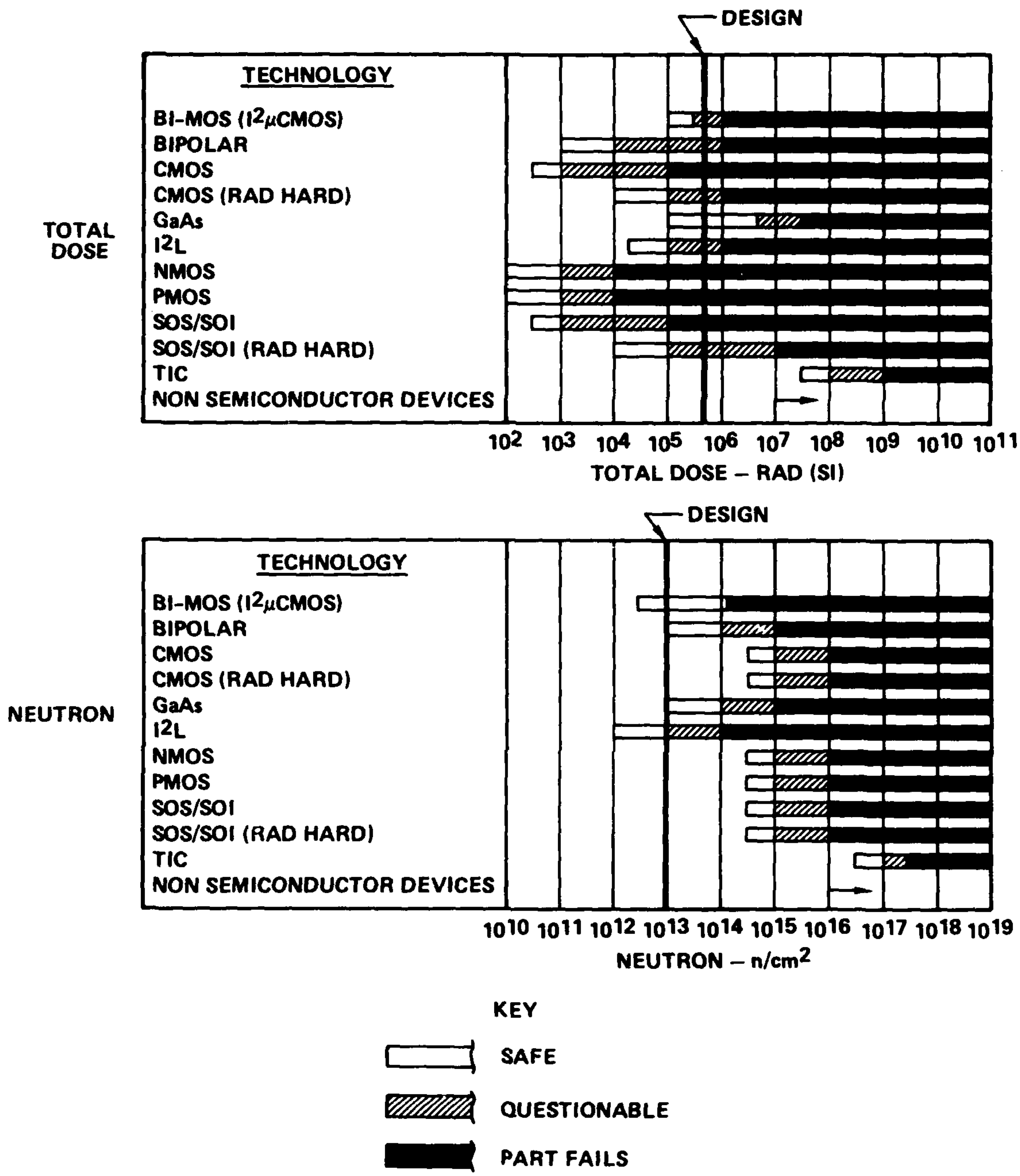


Figure 4.23 Radiation Analysis of Semiconductor Technology

Figure 4.24 OMV Concept

4.6.2 Plasma Environment Interactions

4.6.2.1 *High Altitude Spacecraft Charging*

The effects of this interaction were discussed in Section 4.4.2.1 and are applicable here. Although the OTV/OMV units can be stationed in geosynchronous orbit, they are presently planned to only deliver and retrieve objects to these orbits. During these short duration trips, they may encounter substorms which could result in differential charging. Discharges resulting from substorm encounters could upset the automated systems and seriously disrupt a mission.

In this concept, there is also the possibility of multiple body interactions. The tug and the spacecraft-in-tow can be differentially charged by a substorm. A discharge is possible during separation if they were charged as a unit. Also, there could be discharges when the two mate during retrieval, if one vehicle was charged by a substorm. The mate/demate discharge possibilities depend upon contact taking place before the charged vehicle has time to dissipate its charge into the surrounding plasma.

Since this orbital environment may charge spacecraft and produce multiple body interactions, the system impact for this concept is rated as LARGE (4).

4.6.2.2 *Polar-Auroral Spacecraft Charging*

The effects of this interaction were discussed in Section 4.2.2.2. That discussion is applicable to this concept. In addition to the possibility of OMV/OTV charging, there are concerns of multiple body interactions as expressed in Section 4.6.2.1. These mate/demate interactions are equally applicable to this environment should the rendezvous take place over the poles. The system impact for this concept was also rated LARGE (4).

4.6.2.3 *High-Voltage System Interactions*

The effects of this interaction are not applicable to this concept, since high-voltage systems are not part of the OMV/OTV concept.

4.6.3 High Energy Radiation Environment Interactions

4.6.3.1 *Radiation Damage*

This interaction was discussed in Section 4.2.3.1 of this report and is applicable here. For this concept, the possibility of radiation damage to electronics is higher since this vehicle is expected to traverse the radiation belts many times in its 10 year life. Traveling between low and high orbits would increase the total radiation dose to the electronics, thus increasing the susceptibility of degradation and failure. Increased shielding would be necessary to protect electronics, especially those involved in the automatic flight control system, since a failure there could cause loss of mission.

In addition, increased radiation levels can accelerate the deterioration of composites and other materials. The system impact for these interactions is rated as LARGE (4).

4.6.3.2 *Single Event Upsets*

The effect of this interaction, discussed in Sections 4.2.3.2 and 4.4.3.2, are applicable to this concept. It is important that these upsets do not take place in the automatic flight control system, as mentioned above.

4.6.3.3 *Radiation Hazard To Man-In-Space*

This interaction is not applicable to the OMV/OTV concept since these vehicles will be remotely operated.

4.6.4 Neutral Environment

4.6.4.1 *Atmospheric Drag*

This applicable interaction was discussed in Section 4.2.4.1. In addition to those concerns, one proposed operating technique for the OMV/OTV is to use aerobraking for the return to LEO (Figure 4.25). This would reduce the amount of fuel necessary to decelerate the vehicle. Aerobraking trajectories call for the vehicle to pass within 70 km of the Earths' surface to achieve sufficient braking. Atmospheric models from 60 to 90 km are not adequate to predict if braking in this regime is potentially hazardous. Any uncertainty could cause vehicle loss.

The major concern about atmospheric braking is the reentry heating associated with the maneuver. Another possible interaction, which could occur during the passage through the atmosphere, is vehicle charging by triboelectric processes. As the vehicle passes through the rarified atmosphere, it could produce sufficient friction to cause surface charging. If this charging occurs, then discharge transients are possible that could upset vehicle electronics. Due to the uncertainties associated with aerobraking, the system impact of this interaction is rated as CATASTROPHIC (5).

4.6.4.2 *Atomic Oxygen Surface Erosion*

The discussion in Section 4.2.4.2 is applicable to the LEO portions of OMV/OTV missions.

4.6.4.3 *Surface Glow*

The effect of this interaction was discussed in Section 4.2.4.3 of this report. That discussion is applicable to this concept.

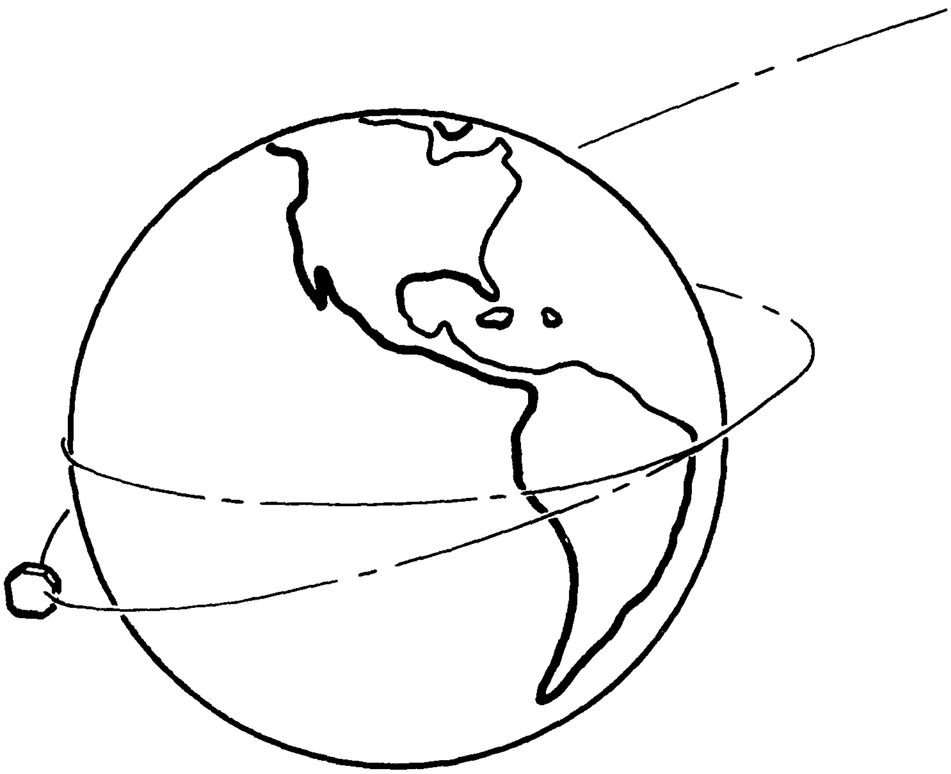


Figure 4.25 Aerobraking - OTV Missions

4.6.4.4 *Sputtering*

The effect of this interaction was discussed in Section 4.2.4.4, and it applies to this concept.

4.6.5 Particle Environment Interactions

The micrometeoroid and debris impact discussions in Section 4.2.5 apply to the OMV/OTV concept. However, the shorter lifetime and smaller size of the OMV/OTV reduces the system impact rating to SLIGHT (2).

4.6.6 Solar Radiation Environment Interactions

Coating degradation and induced thermal forces, due to solar radiation, should not impact the OMV/OTV because of their small size and short lifetimes.

4.6.7 Self-Generated Environment Interactions

4.6.7.1 *Material Contamination*

The effects of this interaction were discussed in Section 4.2.7.1 of this report. That discussion applies to this concept. The primary concern, however, is possible deterioration of sensor performance due to material contamination. Contamination can originate from either the OMV/OTV or the satellite being serviced. Assuming there are no cryogenically cooled OMV/OTV sensors, the system impact rating is LARGE (4).

4.6.7.2 *Thruster Effluent Contamination*

This interaction, discussed in Section 4.2.7.2, applies to the OMV/OTV. The main concern is that the OMV/OTV must be maneuvered to achieve docking with the spacecraft being serviced. Multiple firings of the attitude control jets will be necessary, thus enhancing the possibility of thruster effluent contamination. In addition, the OMV/OTV must regularly use larger thrusters to change orbits. Since this form of contamination must be evaluated, the system impact for this interaction was rated LARGE (4).

4.6.7.3 *Nuclear System Interactions*

This interaction is not applicable to this concept.

4.6.8 Electromagnetic Environment Interactions

The effects of these interactions are not applicable to this concept. The vehicle is too small and the power levels are too low for electromagnetic interactions to occur.

4.7 SUMMARY

The MSSTP was used to establish a set of concepts for an engineering level review of spacecraft environmental interactions. Five concepts were chosen to provide a cross section of operational conditions, spacecraft sizes, and environments. These concepts were: Space Station In Polar Orbit, Astronaut Life Support/Maneuvering System, Space Platform in Geosynchronous Orbit, Nuclear Space Power Systems, and Orbital Maneuvering/Transfer Vehicle.

While these concepts were generally ill-defined, current conceptual designs were used in order to discuss the relevant interactions. The environmental interactions used in this applications study were described in detail in Scientific Report No. 1 of the Spacecraft Environment Interactions Investigation. The interactions were rated for system impact on a scale of 1 to 5, where 5 corresponded to a CATASTROPHIC impact. The results of this applications evaluation were summarized in Table 1.2.

It was not intended for these application considerations to be viewed as a detailed study of the environmental interactions for these concepts, since most are still in the conceptual design stages. There was also no attempt in this report to indicate mitigation techniques for these interactions. Present mitigation techniques for each of the spacecraft environmental interactions can be found in Scientific Report No. 1. However, more detailed trade studies will be necessary before optimum mitigation techniques can be determined.