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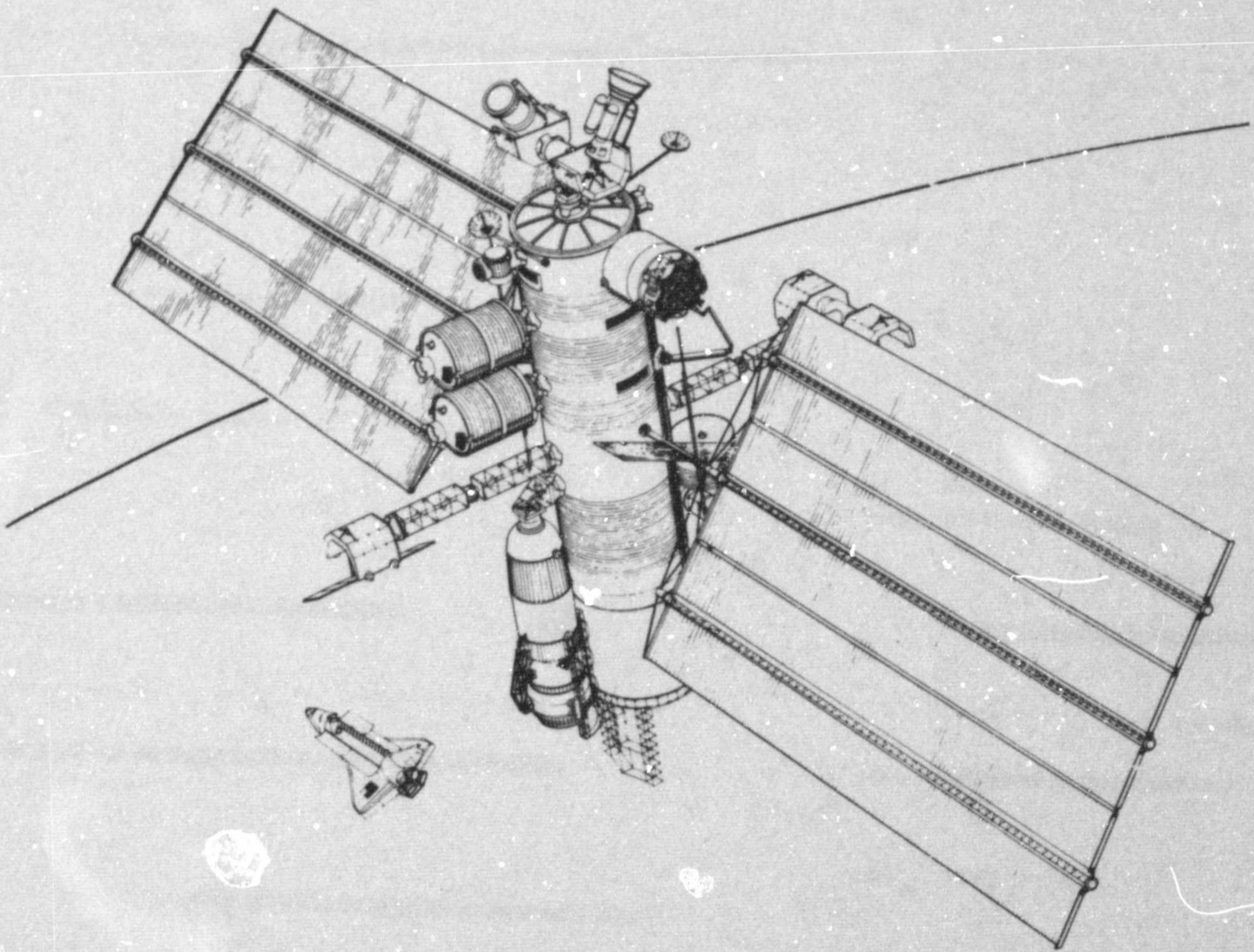
Final Report

November 1983

Power Subsystem Automation Study



MARTIN MARIETTA

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POWER SUBSYSTEM AUTOMATION STUDY

Prepared by:

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for:

National Aeronautics and Space Administration
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Contract NAS8-34938

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FOREWORD

This study was conducted by the Power Systems Section of Martin Marietta Denver Aerospace. The program manager was Mr. Matthew S. Imamura.

Study support personnel and their areas of contribution are:

Robert Moser	Power Processing, Subsystem, and Automation
Marty Veatch	Power Sources and Power Distribution Devices
Robert Richards	Energy Storage
Eric Dietrich	Artificial Intelligence and Expert Systems
Matthew Imamura	Subsystems, Systems, and Automation

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HQ

Mr. Simon Manson - Project and Technology Review

MSFC

Mr. Walter Frost - Program Directions and Artificial Intelligence Technology

Mr. Jimmy Miller - Power

Mr. Roy Lanier - Power

Mr. James Graves - Power

Mr. Audie Anderson - Software

ABSTRACT

Martin Marietta Denver Aerospace undertook a study to develop a method for analyzing, selecting, and implementing automation functions for multihundred-kW photovoltaic power systems intended for a manned space station. The study involved identification of generic power-system elements and their potential faults, definition of automation functions and their resulting benefits, and partitioning of automation functions between power subsystem, central spacecraft computer, and ground flight-support personnel. All automation activities were categorized as data handling, monitoring, routine control, fault handling, planning and operations, or anomaly handling. Incorporation of all these classes of tasks, except for anomaly handling, in power subsystem hardware and software was concluded to be mandatory to meet the design and operational requirements of the space station. The key drivers are long mission lifetime, modular growth, high-performance flexibility, a need to accommodate different electrical user-load equipment, onorbit assembly/maintenance/servicing, and potentially large number of power subsystem components. A significant effort in algorithm development and validation is essential in meeting the 1987 technology readiness date for the space station.

Artificial intelligence technology was briefly assessed, specifically with regard to the applicability of expert systems to the automation functions defined for the power subsystem. Expert-system software techniques have the potential of vast improvement over traditional approaches. Possible onboard applications are for electrical consumables management and battery-operations management, which are system-level tasks. Potential applications for ground use are in non-real-time fault diagnosis, anomaly assessment, and mission planning. An indepth research investigation is desirable to determine the range and domain of artificial-intelligence technology and the resulting hardware and software needs for onboard spacecraft use.

GLOSSARY

ADC	Analog-to-Digital Converter
AgZn	Silver-Zinc
AI	Artificial Intelligence
AMO	Air Mass Zero
APSM	Automated Power System Management
AU	Astronomical Unit
BOL	Beginning of Life
CDS	Control and Display Subsystem
CMD	Command
CPU	Central Processing Unit
CPV	Common Pressure Vessel
CTS	Communication and Tracking Subsystem
CV	Charge Voltage
dc-dc	Direct Current to Direct Current
dc-ac	Direct Current to Alternating Current
DD	Detailed Design
DDTE	Design, Development, Test, and Evaluation
DMS	Data Management Subsystem
DOD	Depth of Discharge
DV	Discharge Voltage
EC/LSS	Environmental Control/Life-Support Subsystem
EMS	Energy Management Subsystem
EOCV	End-of-Charge Voltage
EODV	End-of-Discharge Voltage
EODP	End-of-Discharge Pressure
EOL	End of Life
EPS	Electrical Power Subsystem
ESR	Equivalent Series Resistance
EVA	Extravehicular Activity
GaAs	Gallium Arsenide
GEO	Geosynchronous Equatorial Orbit
GNCS	Guidance, Navigation, and Control Subsystem

GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
H_2O_2	Hydrogen-Oxygen
I_m	Current at Maximum Power Point
IPV	Individual Pressure Vessel
I_{sc}	Short Circuit Current
IUS	Interim Upper Stage
JSC	Johnson Space Center
LEO	Low Earth Orbit
$LiSOCl_2$	Lithium Thionyl-Chloride
MSFC	Marshall Space Flight Center
NiCd	Nickel-Cadmium
NiH_2	Nickel-Hydrogen
P^3	Programmable Power Processor
PD	Preliminary Design
P_m	Maximum Power
PS	Propulsion Subsystem
PSAS	Power Subsystem Automation Study
PV	Photovoltaic
RF	Recharge Fraction
RFC	Regenerative Fuel Cell
RPC	Remote Power Controller
S/C	Spacecraft
SEP	Solar Electric Propulsion
SOC	State of Charge
SOH	State of Health
SCATHA	Spacecraft Charging at High Altitude
Si	Silicon
SR	Series Regulation
SW	Switch
TCS	Thermal Control Subsystem
TM	Telemetry
V_{oc}	Open Circuit Voltage
VO	Viking Orbiter

CONTENTS

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1.0 EXECUTIVE SUMMARY

This chapter presents an overall summary of the study results. Chapter 2.0 provides the objectives, guidelines, and background information for this study. Chapters 3.0 through 7.0 follow with detailed results of the study, arranged in order of the five study tasks. Chapter 8.0 summarizes the artificial intelligence technology and its status, and discusses the potential applicability of the expert system techniques among the power subsystem automation functions identified.

1.1 INTRODUCTION

A major purpose of the Space Station is to implement new designs, concepts, and methods that will reduce life-cycle costs, extend operational life, and yield improved system performance. The resulting power subsystems must therefore be flexible, reliable, efficient, controllable, and most of all, employ a high degree of automation in their operation. To this end, automation technologies are expected to make significant and important contributions to the development and affordable operation of these missions. Therefore, the electrical power subsystem (EPS) must ensure, in the event of a failure, that the onboard power capability will degrade gracefully while providing for some minimum set of useful services. The ultimate power-subsystem configuration would be one that protects against failures and reconfigures itself in the event of a failure so as to continue normal operations.

The primary objective of the NASA-MSFC study undertaken by Martin Marietta Denver Aerospace is to assess and trade off the automation technology required to support a multihundred-kW power subsystem in orbit. This study also is intended to identify the benefits that can be achieved by a logical and planned application of automated and autonomous functions. The basic study guidelines are:

1) Generic photovoltaic power system in the 100- to 250-kW range;

2) Manned and unmanned space station operation;

3) 10-year life.

It is intended that the automation concepts identified will significantly reduce the ground and onboard operational burden; accommodate near-term hardware-technology limitations; and reduce the development, operations, and resupply costs of the space station.

The following definitions of automation and autonomy apply to this study:

Automation - The performance of a function independently and in a manner invisible to the human user or operator;

Autonomy - The application of automated functions without external human intervention for a specified period of time.

There are two basic ways of implementing automation. One is to use hardwired analog circuits and discrete devices. The other is to use a programmable controller or computer. The automation of various monitoring and control tasks enables an autonomous operation. As the duration of autonomous period increases, so does the complexity of automation. Autonomy levels of a spacecraft developed by JPL for the Air Force (Ref 1)* were used in this study for the purpose of demonstrating a method for automation assessment and implementation. The duration of autonomy can be described as (1) operating for x days without ground intervention and no degradation, or (2) operating for y days without ground intervention and under a permissible degradation.

The study consisted of the following five tasks:

1) Characterization and classification of power subsystem;

2) Definition of faults and factors affecting electrical power subsystem (EPS) performance;

*The number in parentheses is the source reference listed in Chapter 10.

3) Definition of automation task candidates;

4) Partitioning of automation functions;

5) Development of automation assessment and implementation method.

The results of each of the above tasks are summarized in the following sections. Appendix A contains the contractual statement of work for these study tasks.

1.2 CHARACTERIZATION AND CLASSIFICATION OF POWER SUBSYSTEM

As shown in Figure 1.2-1, a generic photovoltaic power subsystem was defined by identifying the most promising components under each of the following major subsystem elements: (1) array, (2) power conditioning, (3) batteries, and (4) power distribution. Other elements such as gimbals, auxiliary power sources, and sensors/signal-conditioning circuits were also included. To provide the basis for definition of EPS faults and automation candidates, typical subsystem configuration arrangements were also identified. These arrangements fall into two basic categories, series regulation and direct-energy-transfer types (Fig. 1.2-2 and 1.2-3). The power-subsystem interfaces with all components that consume electrical power and with subsystems that are involved in monitoring and control functions. Figure 1.2-4 shows these interfaces, which are defined in terms of the major space-station disciplines.

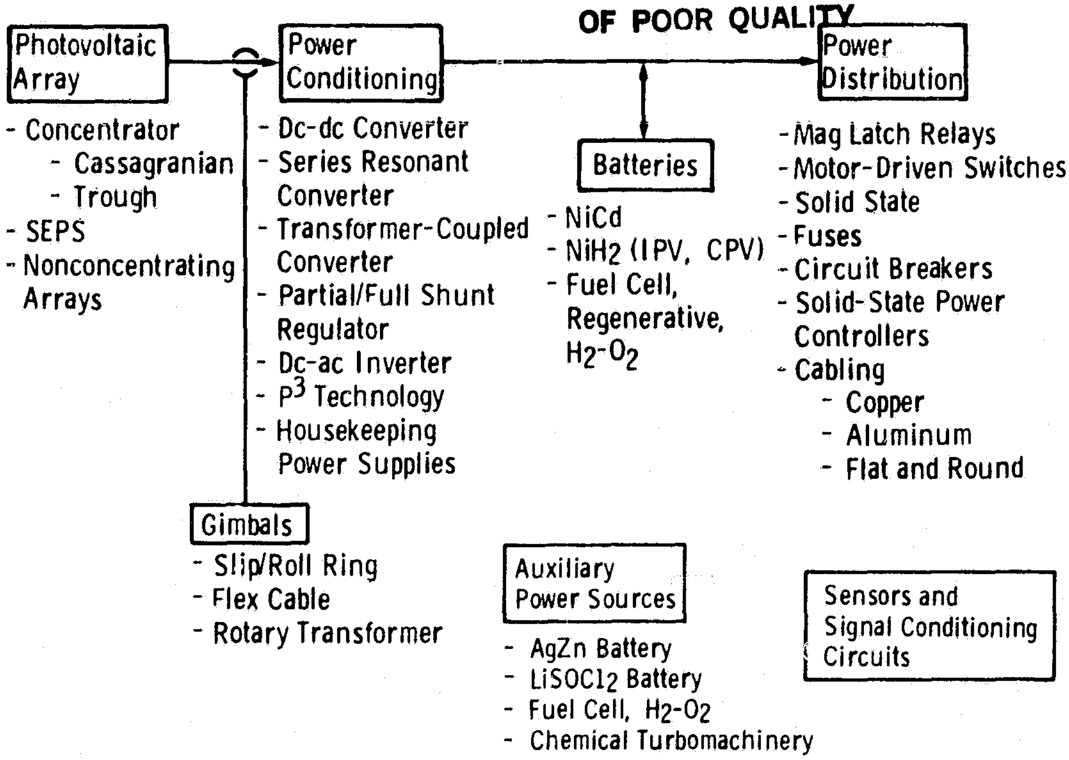

Figure 1.2-1
Generic Photovoltaic Power Subsystem for Space Station

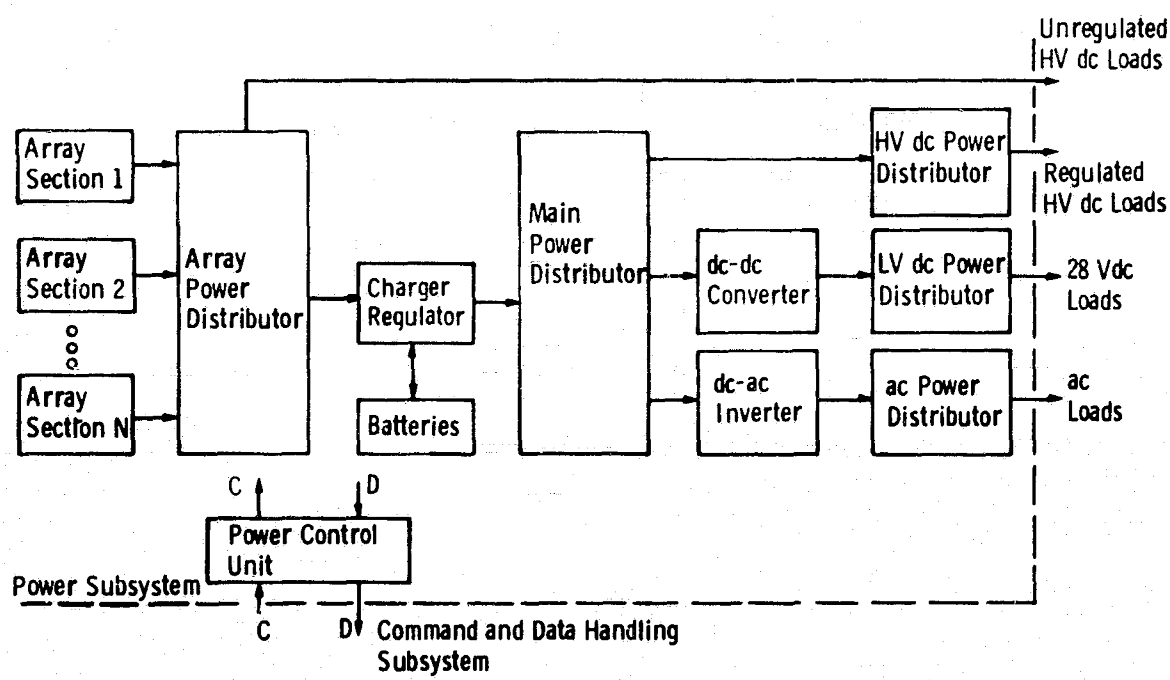

Figure 1.2-2
Series Regulation Configuration of a Space Station Power Module

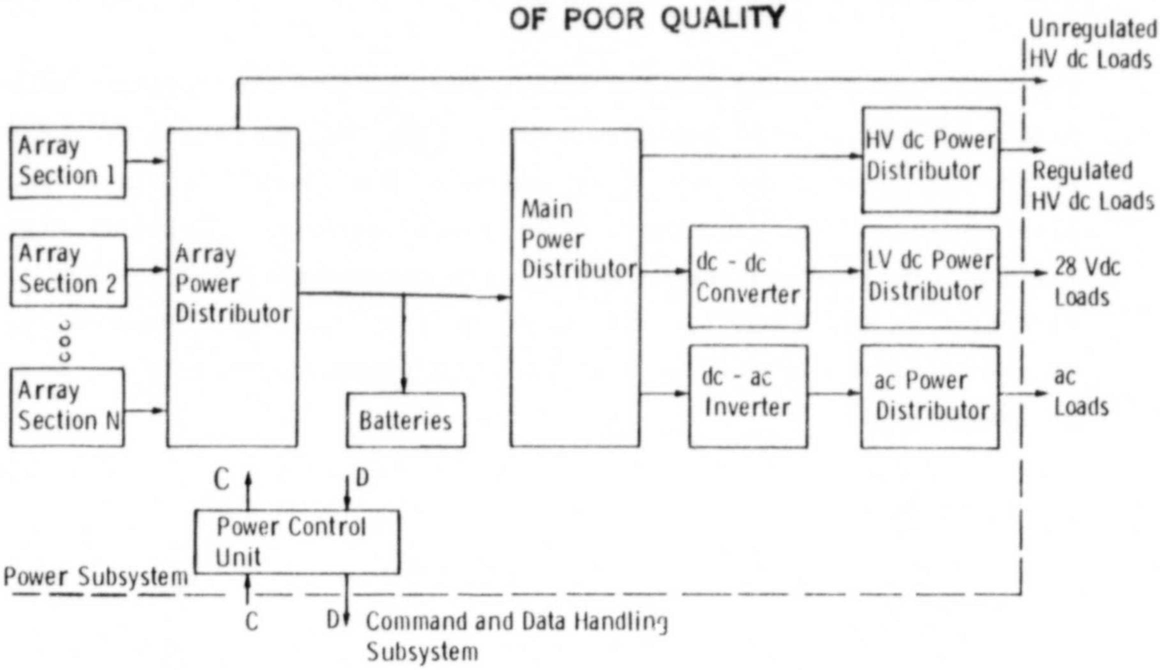

Figure 1.2-3
Direct Energy-Transfer Configuration of a High-Voltage, High-Power Space Station Power Module

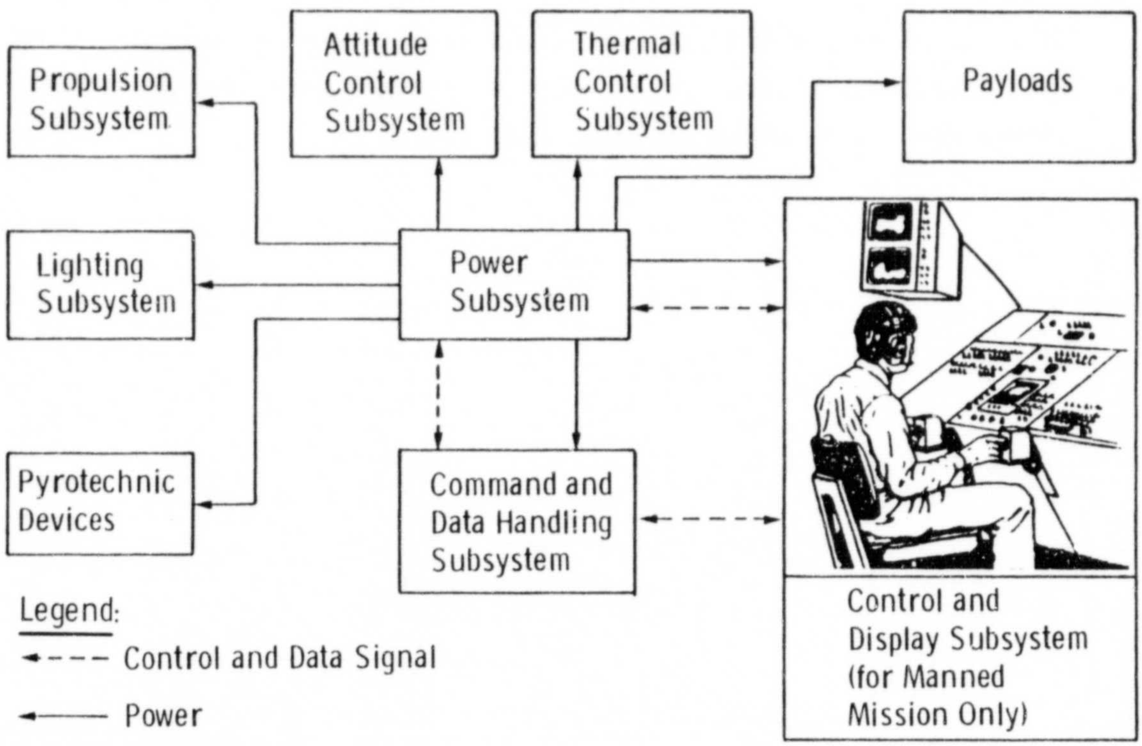

Figure 1.2-4 Power Subsystem Operation Interfaces

1.3 DEFINITION OF FAULTS AND FACTORS AFFECTING POWER-SUBSYSTEM PERFORMANCE

The basis for defining the automation function was the identification of all EPS and non-EPS faults and activities that could affect the EPS or prevent it from performing its intended functions. All major faults were identified for each generic subsystem components listed in Figure 1.2-1 except flywheel energy storage and computer-related devices and circuits. A fault may be defined as the interruption of service at one or more levels of the space station's functional architecture. Specific levels are:

- Piece Part

- Assembly

- EPS

- System

Table 1.3-1 is a summary of the major failure and degradation modes for each component. A summary of other subsystems and the failure that can affect the EPS is shown in Table 1.3-2.

Table 1.3-1 Summary of Major EPS Failure Degradation Modes

Component	Failure Modes	Degradation Modes
Photovoltaic Array	- Open - Short	- Filter, Antireflective Coating - Arcing - Power Loss to Plasma Interaction and Charged-Particle Radiation
Slip Rings	- Short	- Particle Generation from Brushes
Roll Rings	- Open	- Particle Generation from Rings
Twist Flex	- Open	
Regenerative Fuel Cell	- H_2 in O_2 Manifold - O_2 in H_2 Manifold - Voltage/Current Hi/Lo - Absolute Press. Hi/Lo - Excessive H_2 and O_2 P - Temperature Hi/Lo - Voltage Regulator Out of Spec	Seperator Electrode
Nickel-Cadmium Battery	- Shorted Cell - Open Cell - Overpressure Failure due to Cell Reversal	- Loss of Capacity - Low Voltage
Nickel-Hydrogen Battery	- Pressure-Vessel Leak Resulting in an Open Cell - Overpressure Failure due to Overcharge	- Loss of Capacity - Low Voltage
Lithium-Thionyl-Chloride Primary Battery	- Open Cell - Shorted Cell (That Can Cause Other Failures, Including Overpressure)	- Low Final Voltage - Loss of Capacity
P^3 (Dc/Dc Converter)	- Shorted Series-Pass Transistor - Output Overvoltage	- Efficiency - Ripple

Table 1.3-1 (concl)

Components	Failure Modes	Degradation Modes
Transformer-Coupled Converter (Dc/Dc Converter)	- Output Overvoltage	- Efficiency - Ripple
Series-Resonant Inverter (Dc/Dc Converter)	- Shorted Semiconductor Power Switch - Shorted Commutating Diode - Output Overvoltage - Input Cap. Destruction by Overvoltage	- Efficiency
Photovoltaic Array Voltage Controller	- Loss of All Output from an Array	- Partial Loss of Control and Regulation
Magnetic Latching Relays	- Fail To Operate - Transfer When Not Commanded	- Increased Contact Resistance
Remote Power Controllers	- Fail To Transfer - Spurious Transfer - Oscillation - Fail To Limit Rise-and-Fall Time of Current - Fail to Limit Fault Current	- Increased Contact Resistance - Loss of Status Indication
Fuses	- Opens at Current Less Than Specification - Does Not Open at Specification Current	
Cabling	- Open - Short	- Insulation Life Degraded due to Excessive Temperature or Voltage
Sensors	- No Output	- Accuracy Out of Specifications - Out of Calibration
Chemical Turbo Machinery	- Reactant Leakage - Turbine Mechanical Failures - Generator Electrical Failures	

Table 1.3-2 Other Subsystems and Activities that Affect EPS Operation

Subsystem	Failure/Activity	Effects
Structures	Modular Buildup	Modular EPS Required
Thermal Control	Impaired Capacity to Manage Waste Heat	Reduced Power
User Loads (All Subsystems and Payloads)	Shorts or Overloads	Bus Undervoltage
	Large Differences in Day and Night Power at Buses	May Reduce Bus Power; Excessive Battery DOD
Attitude Control	- Gravity Gradient Attitude Mode - Failure to Maintain Required Stable Attitude Because of Unknowns in Controlling Large, Flexible Structures	- Reduced Power - Reduced Power
Command	- Degraded TM Data Transmission - Loss of CPU Power	- Reduced Information - Reduced Automation Capability
Data	Software Maintenance	Reduced Power
EPS/Crew Interface	Crew Commands, Displays, New Crew, Interface Ambiguity, Mistakes	Reduced Power; Unintended Shutdown
EPS Ground Operations	Power Management Configuration History; Audit Trail or Automated Activities; Training; Commands/Displays	Reduced Power

1.4 DEFINITION OF AUTOMATION FUNCTIONS

The ultimate objective is to produce a spacecraft that is fault tolerant and able to perform routine health and maintenance functions without ground intervention. To this end, faults and activities identified for the generic power subsystem were used as a starting point. Specific fault correction and routine health and maintenance functions were then identified. All specific automations were categorized under following classes: data handling, monitoring, routine control, planning and operations, and anomaly handling. A generalized list of benefits was developed (Tables 1.4-1 and 1.4-2). An example of the analysis applied to faults for a dc/dc converter is shown in Table 1.4-3. Table 1.4-4 lists specific examples of automation tasks for monitoring, routine control, and mission operations and planning.

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Table 1.4-1
A List of Generic Automation Functions

Data Handling - Acquisition - Processing - Storage
Monitoring - Operational State - State of Health - Performance Analysis - Trend Analysis
Fault Handling - Fault Detection (Caution/Warning/Alarm Limit Check) - Fault Isolation - Fault Correction
Control
Planning and Operations
Anomaly Handling

Table 1.4-2
A Generalized List of Potential Benefits from EPS Automation

- Increased Life
- Increased Reliability, Maintainability, and Safety
- Improved Performance
- Reduced Cost
- Subassembly
- Subsystem
- Spacecraft
- Launch Operations
- Flight Operations
- Inflight Fault Detection, Maintenance, and Servicing
- DDTE (Design, Development, Test, and Evaluation)
- Ground Support Personnel Labor
- Ground Support Equipment (Prelaunch and Flight Operations)
- C&DH Subsystem
- Thermal-Control Subsystem
- Life-Support Subsystem
- Crew Training Simulator/C&D Subsystem
- Reduced Maintenance
- Able to Overcome Technology Limitations
- Reduced Astronaut/Power Subsystem Interaction
- Reduced Number of Ground-Support Personnel
- Reduced New Subsystem Familiarization/Training Time
- Reduced PV Array Size and Weight
- Reduced Battery Size and Weight
- Reduced Power Conditioning Size and Weight
- Minimized Human Error
- Allows Space Operation without Crew
- Provides Real-Time Short-Response Control
- Reduced Software/Hardware Interfaces to C&DH Subsystem
- Improved Security and Survivability

Table 1.4-3 Dc/Dc Converter Failure Modes, Automation Candidates, and Benefits

Failure Mode	Automation Candidate*	Method	Benefits**
Shorted Series-Pass transistor		Detect overvoltage and close shunt switch.	
Low V (OUT)		Sense V(OUT). When valid overvoltage, priority and load shed, and bus test. Determine P^3 good/bad. Determine V(IN) good/bad. If P^3 bad, switch in backup, priority load, connect. If P^3 good, source overloaded, limit loads reconnected.	
V(IN) High	1, 2, 3	Monitor V(IN). P^3 shutdown on V(IN) H1. Shift loads to another P^3, or add loads to one with H2 V(IN).	1, 2, 4,
I(IN) High		Priority load shed, then if still failed, switch off and bring on backup.	
High Internal Temperature		Monitor temperatures, shut down on overtemperature. Bring back up online. Priority load add.	
I(OUT) Overload		Monitor I(OUT), compare to limit, support for programmed time, turn off pause, restart.	

*See Table 1-3
**See Table 1-4

Table 1.4-4 Examples of Monitoring, Control, and Mission-Operation Automation Tasks

Monitoring Tasks

Operational State Determination
- Number and Identity of Components Online, Offline, or Failed
- Relay Position and Command State

State of Health
- Solar Array, Batteries, Power Conditioning, Bias (Housekeeping) Power Supplies
- Built-in Test and Checkout

Performance and Trend Analyses

Solar Array
- Normalized Peak Power (NPP); Available Average Power/Daytime versus Orbit Number
- NPP and I_{sc} Degradation
- Minimum, Average, and Maximum Temperature

Batteries
- SOC, DOD, EODV, and EOCV Limit versus Orbit Number
- Average Temperature during Charge and Discharge versus Orbit Number
- Total Number of Cycles above X%, DOD, T% DOD
- Number of Cycles Since Last Reconditioning
- Battery Recharge Fraction versus Orbit Number

Bus Power Capability (Orbital Average, Average Power Margin)

Bus Load (Day, Night, and Orbit Average)

Converters and Inverters
- Normalized Efficiency
- Output Impedance

Load Equipment
- Input Impedance

Control Task

Solar Array
- Orientation Control
- Voltage Regulation

Batteries
- Charge and Discharge Control
- Spare Module or Cell Management
- Reconditioning
- Redundancy Management

Converters
- Load-Sharing Control
- Redundancy Management

Imbedded Controller (e.g., P^3 Converter)
- Mode Control (Voltage Regulator or Battery Charger)
- Internal Fault Detection and Isolation
- Overload Handling
- Output Voltage Programming

Planning and Operations Task

Electrical Consumables Management
- State-of-Health Prediction
- Operational State Determination
- Energy-Balance Calculation
- Bus Power Capability and Power-Margin Prediction
- Load Profile Determination
  - Power versus Time
  - Day, Night, Orbital Average
- Load Equipment Timeline versus Power Capability Analysis
- Mission Timeline Compatibility Analysis
- Load Equipment Sequence-and-Command Generation

1.5 PARTITIONING OF AUTOMATION FUNCTIONS

The basic purpose of this task is to develop a method for partitioning the automation candidate between the system, power subsystem, and ground. The partitioning method used is as follows. First, the time criticality of the function is determined. From this analysis, functions can be separated into time-critical functions that require dedicated hardware, such as bus overvoltage; and functions that do not require the fast response time and are candidates to be performed by a computer. Next, the location where the task is to be performed and the resources to do the task are identified. A determination is then made of the external interface impacts--Are the impacts totally within the EPS? Or are these impacts outside the EPS? General criteria established for partitioning the automation functions are as follows:

- Dedicated hardware is to be located in the EPS component;

- Fault detection, isolation, and correction can be partitioned to different levels;

- To be partitioned to the EPS, the fault must originate in the EPS; the correction resources should be in the EPS; and there should be no impacts outside the EPS.

Finally, the last step consists of considering each function partitioned to the EPS, the space station system, and the ground, and providing rationale for or against each subsystem's partitioning. Examples of partitioning of automation functions between the onboard and ground are shown in Table 1.5-1. Note that partitioning can be facilitated in terms of where sensing, analyzing, and acting should best be performed.

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Table 1.5-1 Partitioning of Automation Functions

Automation Function		Partitioning Sense	Analyze	Act	Rationale/Comments
Monitoring					
- Operational State		EPS	EPS	EPS	
- Performance and Trend					
- Solar Array and Batteries		EPS	EPS	EPS	
- Power Conditioning		EPS	EPS	EPS	Other Subsystems Involved;
- Load Equipment	(A)	EPS	EPS	EPS	Data Available to SYS
	(B)	EPS	SYS	EPS	Simplest to Implement
- Bus Power Capability	(A)	EPS	EPS	EPS	
	(B)	EPS	SYS	EPS	
	(C)	EPS	Ground	EPS	
Control					
- Solar-Array Orientation	(A)	ACS	ACS	EPS	Past-Practice Data
	(B)	SYS	SYS	EPS	Available to SYS;
	(C)	EPS	EPS	EPS	Requires SYS Concurrence
- Solar-Array Voltage Regulation		EPS	EPS	EPS	
- Battery Charge and Discharge Control		EPS	EPS	EPS	
- Battery Reconditioning	(A)	EPS	EPS	EPS	Requires SYS Concurrence;
	(B)	EPS	SYS	EPS	May Require Load Management
	(C)	EPS	Ground	EPS	Past Practice
- Battery Spare-Cell/Module Management		EPS	EPS	EPS	
- Redundancy Management	(A)	EPS	EPS	EPS	
	(B)	EPS	SYS	EPS	Whenever Other Subsystems Are Affected
- Converter Loadsharing Control		EPS	EPS	EPS	
Planning and Operations					
- Electrical Consumables Management	(A)	EPS	Ground	EPS	Past Practice (Skylab);
	(B)	EPS	SYS	EPS	Other Subsystems Involved
	(C)	EPS	EPS	EPS	

Legend:

SYS System
ACS Attitude-Control Subsystem
EPS Electrical Power Subsystem

Note:

(A), (B), (C) Are Options

1.6 METHOD FOR AUTOMATION ASSESSMENT AND IMPLEMENTATION

The first step is to define a specific study area such as how to automate the correction of overtemperature faults in batteries. Three basic inputs required for the study are:

1) System-Level Criteria

   a) Space station autonomy/automation requirements, including autonomy level,

   b) Reliability, maintenance and safety requirements;

2) Subsystem-Level Criteria

   a) Functional requirements and description,

   b) Subsystem interfaces,

   c) Component functional requirements;

3) Mission Operations

   a) Man-machine interface,

   b) Flight-controller functions (i.e., ground crew),

   c) Astronaut/subsystem operational criteria and constraints.

The autonomy level is used to prioritize automation candidates and aid in partitioning automation functions between the ground and the Space Station. Reliability requirements are used to categorize faults and to aid in selecting a fault-correction option. Mission-operations criteria are used to define specific automation functions needed for orbital operations.

Factors to be analyzed and defined in a detailed assessment of the automation function are:

1) Impact;

2) Fault category;

3) Fault correction options;

4) Benefits;

5) Time-criticality;

6) Basic implementation, hardware or software.

Basic technical elements in NASA's program development usually consist of Phase A (planning, conceptual requirements definition, and design), Phase B (preliminary requirements definition and design), and Phases C & D (detailed design, fabrication, and integration; launch operations; mission operations). It is assumed that Space Station-level autonomy/automation and reliability requirements will be addressed in each of these program phases, and their details will increase the program phases' progress. The method outlined here depends to a large extent on the system-level requirements available. Therefore, the extent to which automation assessment can be done at the subsystem level would be a function of level of details available at the station level. It is logical then to assume that the designers, especially during Phases B, C, and D, would have access to top-level specifications and design-criteria documents covering not only autonomy/automation requirements, but also other high-level functional criteria.

1.7 ARTIFICIAL INTELLIGENCE (AI) AND EXPERT SYSTEMS

1.7.1 AI Technology

Artificial intelligence is that branch of computer science concerned with the design and implementation of programs that make complicated decisions, learn, or become more adept at making decisions, interact with a man in a natural way, and, in general, behave in a manner typically considered the mark of intelligence.

Intelligence is to be understood not as a property that, for example, gifted mathematicians possess, but rather as a property all men and some animals possess. Intelligence, in this sense, is the ability to understand and process large amounts of information. It is the ability to meet and cope with novel situations, to comprehend the interrelationships between facts and concepts, and to generate new concepts and relationships from those already known (i.e., already in the data base). The artificiality of the intelligence means merely that the intelligence is achieved by means of technology.

Scientific research done in AI covers a large area of theoretical topics such as knowledge representation, knowledge acquisition, problem solving and search, vision, theorem proving, and natural language. Though each one of these topics can be researched from the human-ability perspective, i.e., by asking how a man represents knowledge, acquires knowledge, solves problems, sees objects, communicates, etc, researchers in AI are concerned with implementing the given ability in computers. AI is not only a theoretical enterprise, it has definite and robust applications. The primary concern in the applications arena is the design and implementation of expert systems and natural language interfaces.

1.7.2 What Is An Expert System?

An expert system is an intelligent computer program that embodies the knowledge of human experts in a particular domain of expertise. Expert systems recognize situations, derive conclusions, make decisions based on what they recognize, and recommend corrective and directive actions. All of this is done with a competence comparable to that of human experts. Figure 1.7.2-1 illustrates the basic components of an expert system. It contains a knowledge base, a rule base, and an inference engine. The knowledge base (sometimes called working memory) stores the information (data) on which the expert system operates. The knowledge base is constantly updated as data are added or deleted. The rule base is the component that gives the expert system its expert competence--that is, the ability to make decisions, recommend actions, etc.

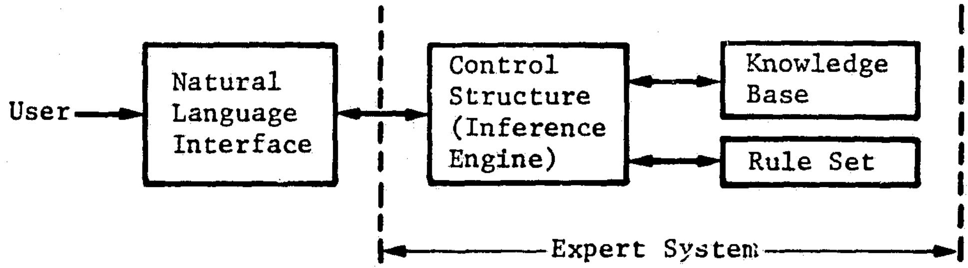

Figure 1.7.2-1 Basic Components of an Expert System

The inference engine's job is to execute various rules depending on the contents (data elements) of the knowledge base. Conceptually, the inference engine's algorithm is a search and pattern match. It scans the rules, efficiently searching for a rule whose antecedent (the IF part) matches the present state of the world, i.e., the facts in the present knowledge base. If a match is found, the consequent of the rule (the THEN part) is executed. The actions can be anything from querying or advising a human user to performing a real-world action, such as uplinking commands to a satellite or moving a robot arm, to manipulating its knowledge base or rule set and modifying the behavior of the expert system itself.

1.7.3 Natural Language Interface

It is usual to have a natural language interface to facilitate the use of the expert system. A natural language interface is a computer program that allows an end user to interact with an applications program using a "natural" language such as English rather than special menus or special-purpose languages such as FORTRAN for programming, RAMIS for data-base queries, or JOVIAL for command and control. A key advantage to using a natural language interface rather than a more conventional interface is ease of learning and use. Because English is used, no special languages must be learned. Because its use is an extension of a person's normal communication skills, a natural language interface can often be a highly effective way to interact with a computer program.

1.7.4 Expert System Applicability

Four considerations must be taken into account when deciding whether an activity warrants using an expert system. These four are applicable to a wide variety of domains and find ready application in the area of automated power subsystems. The reader is referred to other publications (Ref 2 and 3) for a discussion of expert systems.

A given candidate for automation warrants considering the use of an expert system if it:

1) Is to be used for possible control applications, for non-real-time processing, or where very slow response is required;

2) Must process large amounts of information;

3) Requires nonalgorithmic, heuristic problem solving;

4) Requires a high-level, human-like decision;

5) Is such that the software requires frequent modification as a result of changing performance characteristics, and operating criteria and constraints.

Another discriminator is complexity and how the tasks were performed in the past. Simple tasks that are well understood and have algorithmic solutions are not good candidates for expert-system solution. If the task is complex enough that in the past it could only be performed by a recognized expert, or group of experts, then the task is a good candidate for automation by expert-system software.

The following functions were identified as good candidates for automation by expert-systems software:

1) Battery operations management (as contrasted with routine charge/discharge control and protection);

2) Electrical consumables management;

3) Trend analysis;

4) Fault analysis (fault detection and diagnosis only and not corrective actions);

5) Anomaly handling.

In the past, the computer has been used to maintain a data base and to plot data on request, but a man was required to interpret the data and initiate corrective action. This is an area where expert-system software could be used to replace some of the human experts. Complex faults that would require tree searching using algorithmic software could be replaced by the heuristic approach. Consumables management could be done with algorithmic software, but there may be benefits in development time and ease of modifications if expert system software were used because of the dynamic natures of power management and load management. In the past, an anomaly has occurred when there was no preprogrammed, algorithmic response to a situation. A group of experts would be assembled to analyze the data, propose experiments, and deduce a response. Many types of faults have similar traits. Anomaly handling and some types of faults therefore appear to be a fertile area for an indepth assessment of expert-system applicability.

1.8 CONCLUSIONS AND RECOMMENDATIONS

The significant conclusions and recommendations of the study are as follows:

1) To meet basic station objectives and goals presently defined in the NASA Space Station Definition Book, all power subsystem automation candidates defined in this study, except for anomaly handling, must be implemented to a varying degree of automation.

2) Specific functions that have immediate high payoffs for onboard applications are:

   a) Data Acquisition, Processing, and Storage,

   b) State of Health Monitoring,

   c) Built-in Test and Checkout,

   d) Fault Detection, Isolation, and Correction,

   e) Performance and Trend Analysis,

   f) Integrated Array/Battery Controller and Load Management (Space Station Level),

   g) Electrical Consumables Management (Space Station Level).

   Automation of any combination of the above functions (a through g) will have a significant beneficial effect on mission-operations efforts on the ground. A detailed study is recommended to determine the effects of onboard automation of monitoring functions on ground activities such as failure detection, consumables management, and crew and flight-controller training.

3) A key driver in when and what to automate in the subsystem is spacecraft autonomy level, which must be defined at the program level.

4) The best way to partition an automated activity between the EPS, spacecraft system, and ground is to first define each subtask required to be performed, and then assign each subtask to EPS, system, and ground, in terms of:

   a) Sensing,

b) Analyzing,

c) Acting;

5) For real-time control consideration, the principal driver in hard-wired-versus-software (i.e., using digital computer) trade is the speed requirement for implementing that control function. Therefore, in general, all offline or non-real-time tasks such as monitoring, performance analysis, and fault diagnosis that require slow response and are not in the control loop, can be done with a digital computer.

6) The best onboard-application candidates for expert systems for any of the power automation functions appear to be for electrical-consumables management and battery-operations management. Potential ground applications are in non-real-time fault assessment and mission planning. An indepth research investigation is desirable and highly recommended to determine:

   a) The range and domain of its applicability to power-system control functions;

   b) Adequacy of AI language for onboard use;

   c) Computer hardware (speed, memory) required to support expert-system software.

7) A significant effort in engineering-algorithm development and validation is essential in meeting the 1987 technology-readiness date. There are many implementation approaches to each automation function because they are done by software. Thus, future efforts in algorithm development must include optimization processes with simplicity and reliability in mind. It should be emphasized that algorithm development also is necessary to permit a detailed design of any expert-system software such as that for electrical consumables and battery management.

2.0 INTRODUCTION

2.1 OBJECTIVES AND SCOPE

The primary objective of the study was to assess automation technology required to support a multihundred-kW photovoltaic power subsystem for space station and platforms. To do this, the following five subtasks were identified in the statement of work (see Appendix A):

- Task 1 - Characterize and Classify a Generic Power Subsystem
- Task 2 - Define Faults and Activities That Could Affect Power Subsystem Operation
- Task 3 - Define Candidate Automation Tasks
- Task 4 - Partition Automation Tasks between the EPS, Space Station (Central Computer), and Ground
- Task 5 - Develop Method for Assessing and Implementing Automation Tasks

A secondary objective of this study was to evaluate artificial intelligence technology and identify its potential role in power subsystem automation.

2.2 STUDY GUIDELINES

The following study guidelines were used:

- Power Subsystem Type: Photovoltaic/Battery
- Power Level: Multihundred-kW Range
- Modular Design

- Lifetime of At Least 10 Years

- Use of Space Station and Autonomy/Automation Study Documentation:

  - Space Station Systems Definition, Book 5, Nov 82 (Ref 9)

  - Autonomous Spacecraft Program Study for the Air Force by Jet Propulsion Laboratory (JPL) (Ref 10-12)

2.3 BACKGROUND INFORMATION

A major goal of the present Space Station is to implement new designs, concepts and methods to reduce life-cycle costs, extend operational life, and yield improved system performance. The resulting power subsystems must be flexible, reliable, efficient, controllable, and most of all, employ a high degree of automation. To this end, automation technologies are expected to make significant and important contributions to the development and affordable operation of these missions. Therefore, the electrical power subsystems must ensure, in the event of a failure, that the onboard power capability will degrade gracefully and provide a minimum set of useful services. The ultimate power-subsystem configuration would be the one that protects against failures and reconfigures itself in the event of a failure so as to continue normal operations.

This study is concerned with automation of functions within the power-subsystem and also space-station level tasks related to it. The term "automation" has diverse interpretation. It can describe a simple control of a process by an on-off device as in a thermostatic control. It is used to describe a complete feedback-control process that includes sensing, analyzing, and doing a required operation like voltage regulation. Automation has also been used to describe more complex processes in which the automated system replaces some of the human activities.

All automation functions fall basically into two categories: monitoring and control. The monitoring function involves sensing, analyzing, and displaying solution approaches and simple decisionmaking information for user (i.e., human) disposition. It is not in the control loop, so monitoring _per se_ does not affect the reliability of that control circuit. A control function consists of all the elements of an operation--sensing, analyzing, and effecting. The fundamental problem of automation, given that the function should be automated, is that of strengthening the designer's and user's confidence that automated functions will be accomplished effectively and reliably. This requires confidence in hardware and software reliability, adequate optimization and validation, and flight experience. Questions such as the following are of concern to this and future studies involving automation:

- What is automation all about? What is the minimum level of automation? What can be automated?

- Why should automation be undertaken? Can it significantly improve the life and performance of some components? Can it increase the specific power of the power subsystem? Can it reduce the cost of the power and other spacecraft subsystems?

- What system-level studies are needed to evaluate the desirability and identify guidelines for subsystem automation development? What are the appropriate jobs for the flight crew?

- What effect might automation have on the next version of the subsystem design? How can subsystems be designed or modularized to minimize the consequences of changes? Can software minimize changes? Is standardization an issue?

To address the question of which activities to automate, it is necessary to examine (1) basic criteria that direct space station (and other spacecraft) toward automation, (2) how automation tasks work at the component, subsystem, and system levels to meet their objectives, (3) problems encountered in past spacecraft, and (4) what has been done in past automation efforts.

Table 2.3-1 lists the basic reasons from the system and subsystem points of view as to why automation is often mandatory in many cases. The basic approach necessary in achieving an autonomous operation is to provide adequate sensors, redundant hardware, switching capabilities, and software. The principal goal of this approach is to prevent loss of any critical function via timely reconfiguration and graceful degradation.

Table 2.3-1 Why Autonomy and Automation?

From Mission and Spacecraft Viewpoint: - Enable Autonomous Spacecraft Operation, Especially during Degraded Modes - Enable Rapid Changes in Mission Sequence - Enable Onorbit Subsystem Checkout, Verification, and Maintenance Quickly and Precisely - Decrease Reliance on Ground Stations and Reduce Long-Term Flight Operations Cost - Decrease Cost of Other Housekeeping Subsystems
From Subsystem Viewpoint: - Reduce Subsystem Size and Weight - Increase Operational Life and Performance Reliability - Decrease Subsystem Cost - Respond Rapidly to Malfunctions - Permit Maximum Use of Capability - Permit Graceful Degradation - Overcome Technology Limitations - Accommodate New Technologies

Table 2.3-2 shows the key projects collectively representing the state of development in spacecraft power subsystem automation. Note that the more recent efforts by the Air Force are being performed at the spacecraft level. The principal features and results of these major programs (Ref 4 through 8) are summarized in Table 2.3-3. It should be emphasized that the microprocessor is the key technology that enabled these development projects to be carried out effectively. However, several key issues have yet to be addressed and validated. Among these are processor redundancy configuration and management strategy, processor fault-tolerant criteria and implementation approach, and optimization of application software and long-term validation.

Table 2.3-2
Major Projects Involving Spacecraft Power Subsystem Automation

Project	Dates	Funding Source	Contractor
ARMMS (Autonomous Redundancy and Maintenance Management Subsystem)*	1982-1986	AF-STC	JPL
Autonomous Spacecraft*	1981-1986	AF-STC	JPL
Power Subsystem Automation Study	1982-1983	NASA-MSFC	Martin Marietta
Energy Management System Software Development (Expert System Demonstration)	1983-1984	NASA-MSFC	Martin Marietta
MAPS (Miniaturized Autonomous Power System)	1980-1982	Classified	Martin Marietta
AMPS (Autonomously Managed Power System)	1978-1982	NASA-MSFC	TRW
P^3 (Programmable Power Processor)	1979-1981	NASA-MSFC	Martin Marietta
APSM (Automated Power Subsystem Management)	1978-1979	NASA HQ-JPL	Martin Marietta
SBPS (Single-Cell Battery Protection System)	1975-1977	NASA-LeRC	Martin Marietta

*Spacecraft level, including power subsystem.

Table 2.3-3 Principal Features and Results of Major Projects

Project	Features	Key Results
ARMMS (Continuing) (Ref 4)	- Add-On Computer-Based Subsystem - Interfaces Only with Satellite TT&C - Receives TM Data, Determines Maintenance, and Implements Contingency Plans - Allows for Evolutionary Development - Test Bed for Ground Validation - CMOS Processor (16-bit) - Emphasis: Algorithms, Architecture, and Proof of Concept	- Engineering Algorithms Defined for DSCS III Satellite - Communication Interfaces and Computer Architecture Defined - Functional Requirements Identified - Spacecraft Telemetry Simulator Designed
AMPS (Continuing) (Ref 5)	- 250-kW Design (17 Channels, 16.7 kW Each); Channels Isolated - 220-Vdc Nominal - 150-A-h, 160-Cell, Ni-H2 Battery per Channel - Array Series-String Switching for Voltage Control - Algorithms: Power Source, Load Center, and EPS Management	- Detailed System Design Completed - Algorithms Designed - Computer Architecture and Hardware Defined

P^3
(Completed)
(Ref 6)
	- Charger or Regulator Function via Software Change
	- Single Imbedded Computer (TI9900)
	- Input/Output:
	  - Input: 26 to 375 Vdc
	  - Output: 24 to 180 Vdc
	- Algorithms: Array Peak-Power Tracking, Caution and Warning, Current Limit
		- Engineering Prototype Designed
		- Algorithms Demonstrated and Validated

APSM (Completed) (Ref 7)	- Test Bed Using V075 Power Subsystem Components - Distributed Processors with Central (TI9900) and Local (RCA 1802) - Fault Simulators - Cell-Level Battery Protection (One Battery) - Algorithms: Data Handling, Monitoring, Control, Resource Management, Fault Handling	- Test Bed Operational - Algorithms Functional - Distributed-Microprocessor Concept Demonstrated
SBPS (Ref 8)	- Cell-Level Protection, Both Analog and Digital Configurations - Intel 8008 & 8080 Processors - 18-Cell Secondary AgZn Battery	- First Use of Microprocessor Verified on Secondary AgZn Battery Protection - Hardware and Software Demonstrated - Battery Cycle-Life Improvement (AgZn)

Effective use of automation often implies performance of several tasks concurrently. This means both subsystem- and system-level tasks should be identified and evaluated. Successful automation of the space station may, therefore, transcend boundaries created in the past between disciplines. The classical parochial and dissected view of a spacecraft is likely to be changed. The interaction between the EPS, life-support subsystem (LSS), and thermal-control subsystem (TCS), for example, can be so involved that functions like load sequencing and overall power management can be viewed only at the system level. One attractive system-level automation task is spacecraft energy management. This involves a carefully coordinated electrical-load management that satisfies both experimental needs and the functional requirements of critical subsystems such as LSS and TCS. This activity can significantly reduce the battery mass, which is a substantial fraction of the overall space-station weight if a conventional approach is used.

3.0 TASK 1 - CHARACTERIZATION AND CLASSIFICATION OF POWER SUBSYSTEMS

OBJECTIVE

The objective of this task was to classify and characterize the photovoltaic power subsystem and its major elements. This task was intended to provide the basis for subsequent study tasks.

SUMMARY

A generic photovoltaic power subsystem was defined by identifying the most promising components under each of the following five major categories:

1) Photovoltaic array,

2) Power conditioning,

3) Batteries,

4) Power distribution,

5) Power control.

Thermal control hardware was not considered in this study. However, it must be recognized that heat dissipation management presents a significant problem for high-power systems. Other elements such as gimbals, sensors, signal conditioning circuits, and auxiliary power sources were included. Typical subsystem arrangements were also identified. These arrangements fall into two basic classes by the power conditioning strategy used, the series regulation, and direct energy transfer.

The power subsystem interfaces with all electrical components that use power and with the spacecraft subsystem involved in data acquisition and command functions (C&DH and control and display subsystems).

The photovoltaic power systems can be classified roughly by:

1) Application or mission type: LEO, medium altitude, GEO, planetary;

2) System arrangement: series regulation (SR) or direct energy transfer (DET);

3) Bus voltage level and type: ac, dc, or combination.

A key system performance parameter is the overall specific power (W/lb) which is basically a function of the type of solar cell and battery cell used and the orbit altitude. Typical values estimated by the Air Force (Ref 16) are depicted in Figure 3.0-1 for several combinations of these hardware. The specific power for a system is highly dependant on the battery energy density used.

3.1 SYSTEM CONFIGURATION

3.1.1 General Classification

A power subsystem for any spacecraft comprises the following generic elements:

1) Energy source,

2) Energy storage,

3) Power conversion,

4) Power processing (conditioning),

5) Power distribution,

6) Power control.

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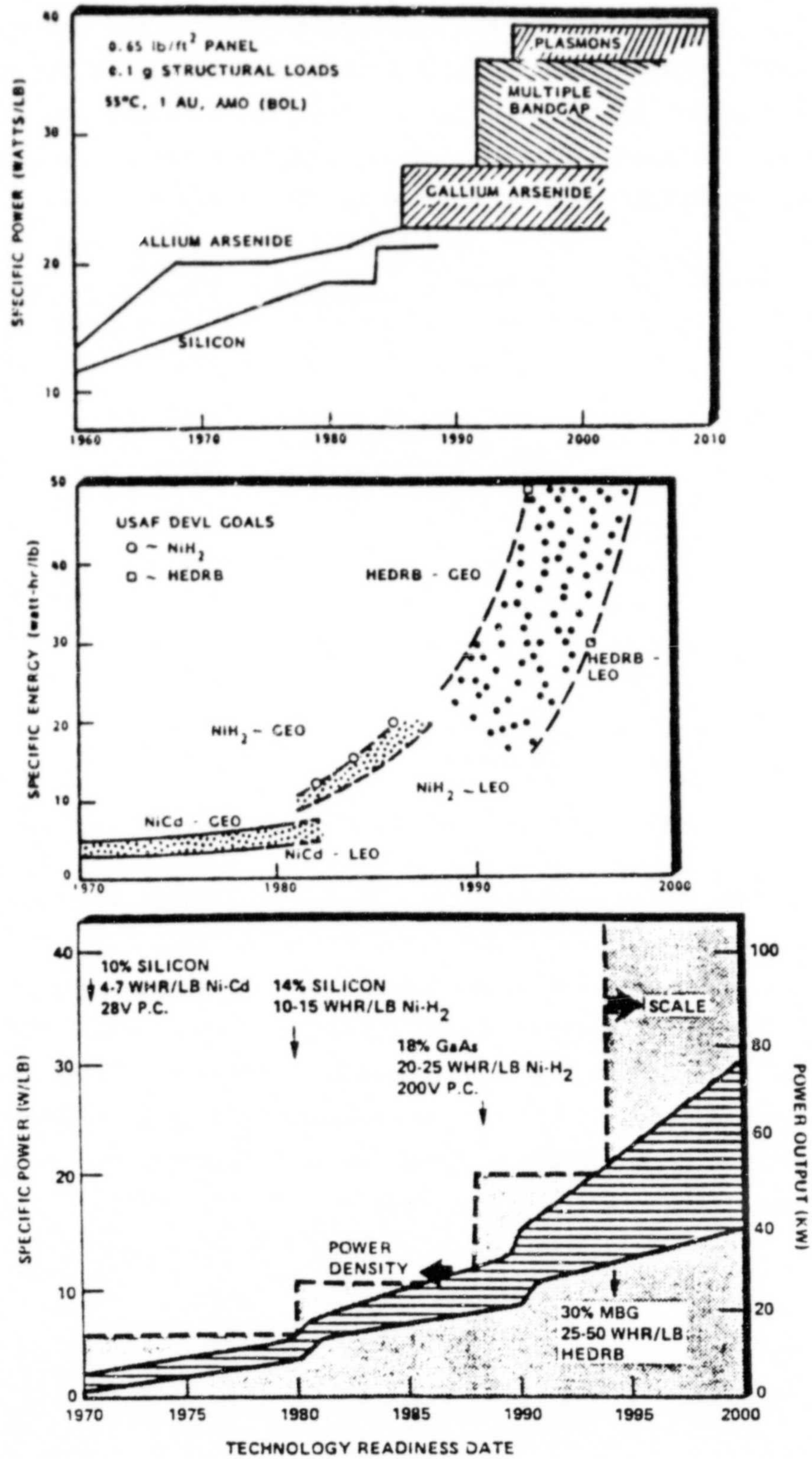


Figure 3.0-1
Specific Power Projection for Photovoltaic Power System

Figure 3.1.1-1 shows the relative arrangement of these subsystem elements along with their principal interfaces. In past spacecraft, control and data interfaces from the spacecraft C&DHS to the power subsystem components were distributed rather than centralized in the power control as depicted in Figure 3.1.1-1. That is, data and control signals were usually routed directly to the power subsystem assembly, such as the power distribution unit and the battery charger.

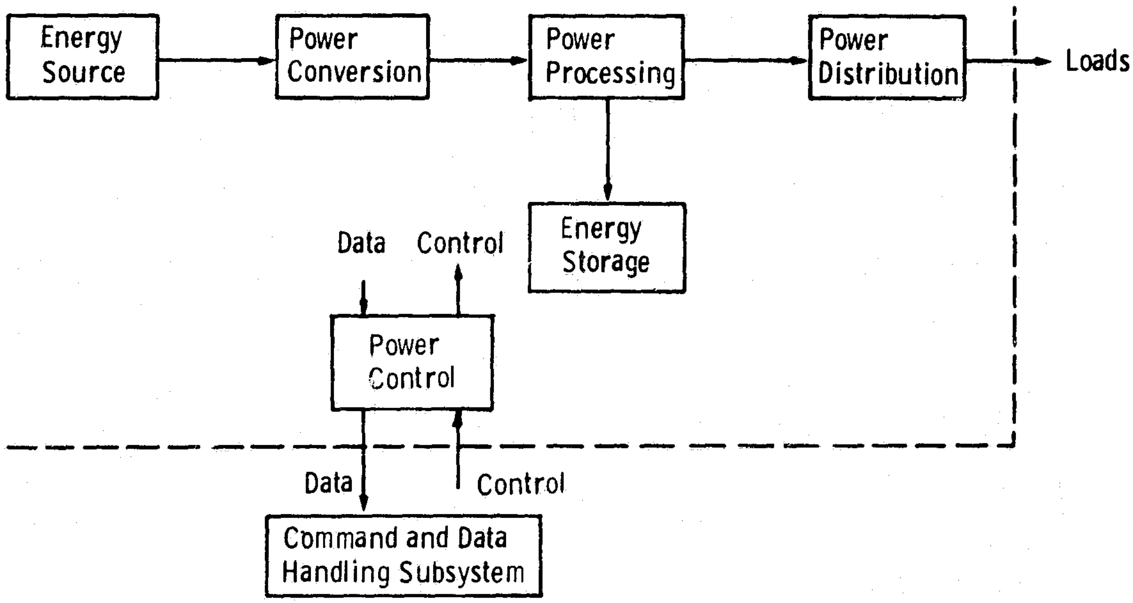


Figure 3.1.1-1 Generic Power Subsystem Elements and Interfaces

The photovoltaic power subsystem was defined to include various components listed under each major subsystem category (Fig. 3.1.1-2). Each component was characterized by key design features, operating characteristics, state of the art, flight history, and types available. Flywheel energy storage was the only component in Figure 3.1.1-2 that was not characterized because of its low development state. System-level options, such as dc bus voltage level, ac vs dc, and number of power channels, are listed in Figure 3.1.1-3.

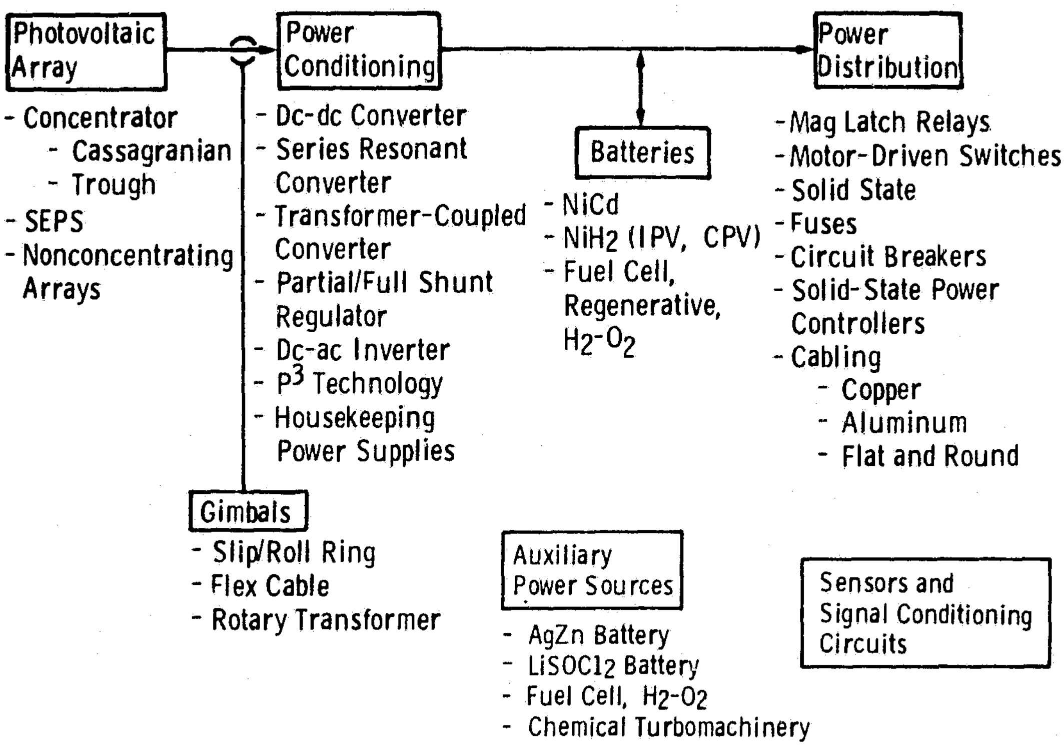


Figure 3.1.1-2 Photovoltaic Power Subsystem Options

Arrangement: Direct Energy Transfer (DET) vs Series Regulation (SR)

Number of Power Channels

dc Bus Voltage: Nominal 28 Vdc and 120 Vdc or 240 Vdc

ac Voltage, Frequency, and Phases

Power Conditioning & Distribution:

- Centralized vs Distributed Voltage Regulation and Power Distributors

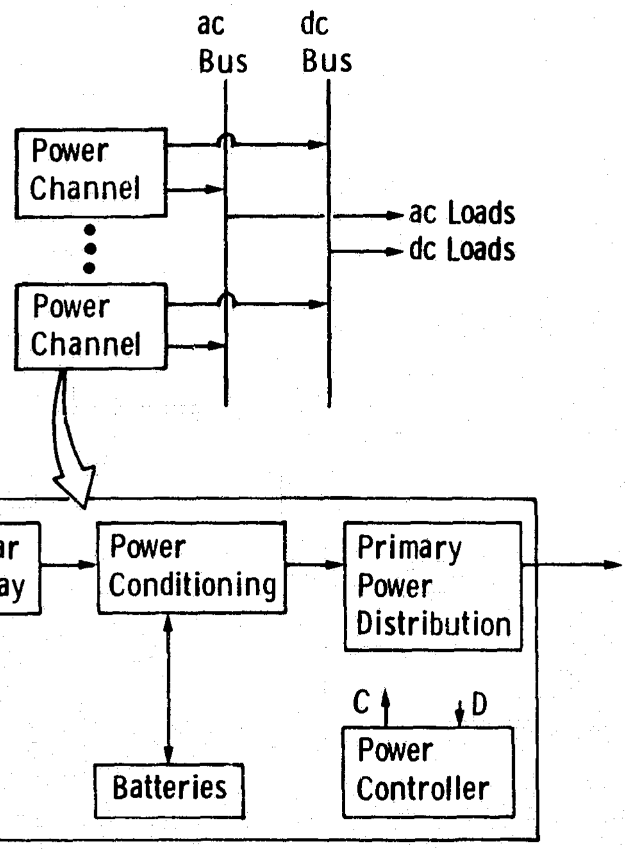


Figure 3.1.1-3 System-Level Options for a Multihundred kW Power Subsystem

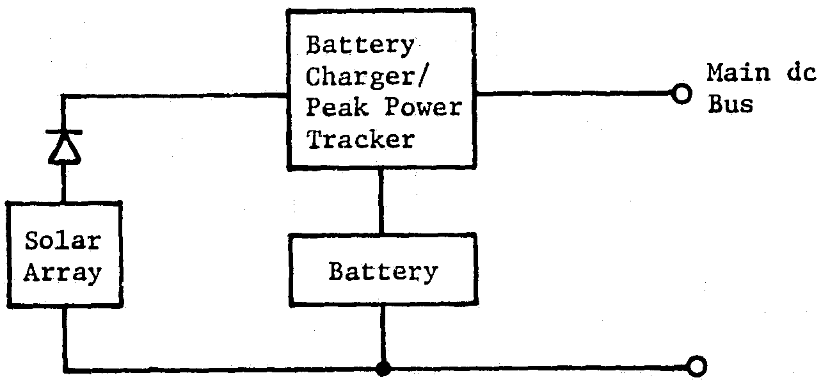
The arrangement of the electrical power subsystem connecting the solar arrays, batteries, power conditioning, and power distribution network to the user loads is critical to reducing the specific weight and cost of the subsystem and improving its efficiency. Figure 3.1.1-4 shows the two basic arrangements that have been used predominantly in spacecraft: one is a direct energy transfer (DET) and the other is a series regulation (SR) type. These configurations differ basically in their methods of controlling the solar array output voltage and providing battery charge/discharge protection.

Configuration I features a dc battery charger and peak-power tracker combination. The peak-power tracker integral to the battery charger provides maximum solar array energy collection whenever the battery is not fully charged and can accept the available power.

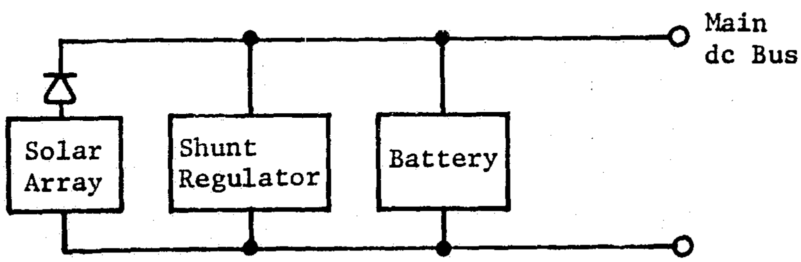
Configuration II requires no dc battery charger but relies on full-shunt regulation to limit battery charge voltage. This arrangement eliminates the cost of the dc charger and the efficiency loss caused by charger operation. The increase in total system efficiency gained by deleting the series charger more than offsets nonoptimum solar array operation off the peak-power point. The main penalty of this full-shunt regulator approach is the need to dissipate a large amount of unusable array power in the regulator.

Configuration III controls the dc bus voltage in a manner similar to II, and is known as a partial shunt regulation system. Its advantages over II are basically a much lower level of power dissipation (in the bypass switches) and elimination of the full-shunt regulator hardware. Its principal drawback is control complexity and related electronics.

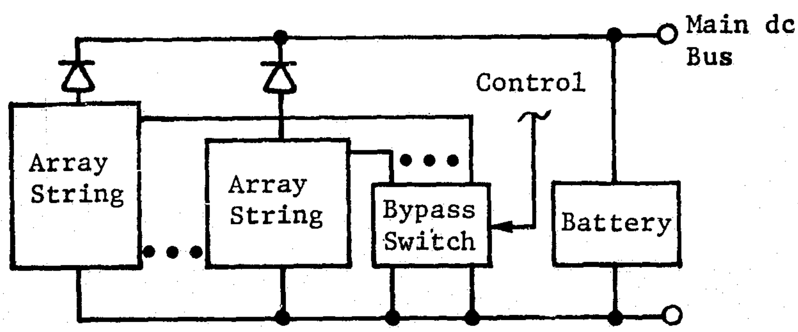
Configuration I (SR)



Configuration II (DET)



Configuration III (DET)



Note:

1. Main dc bus is connected to a load regulator, inverter, and/or power distributor.
2. The bypass switch in Configuration III can be linear partial shunt or digital switch

Figure 3.1.1-4
Basic Photovoltaic Battery. Power Subsystem Arrangements

A combination of II and III, controlled by a miroprocessor, has been used very effectively in a large terrestrial system (Ref 14, 15). Its advantages are:

1) The overall system cost is lowest (comparted to I and II) because the intermediate power processor is eliminated and the partial sub-array on/off switching approach permits the full-shunt regulator to be sized to only handle a fraction of the total available power (partial shunt regulator), and thereby minimize thermal dissipation management.

2) The partial shunting approach provided a very flexible and effective battery control for four 240-Vdc batteries in parallel.

3.1.2 Specific System Arrangements

The modular nature of a PV/battery system allows this power source to be used in applications ranging from a few watts to megawatts. For a multihundred kW system, the key tradeoff issues are the (1) main dc bus voltage level (120 vs 240 Vdc), (2) ac vs dc for main power distribution, and (3) the power distribution scheme to meet the redundancy criteria. An example of an arrangement that can provide a combination of unregulated (150 to 300 Vdc) and regulated (200 to 300 Vdc) HV, low voltage (28 Vdc), and ac power in a DET configuration is shown in Figure 3.1.2-1. This arrangement can serve as a building block to scale up to the required Space Station power levels while providing redundancies in power channels. The power distribution configuration and load control strategy must be carefully designed at the system level to provide the flexibility required for load management during various phases of station growth. Several examples of photovoltaic power system configurations are presented in simplified forms in Appendix B.

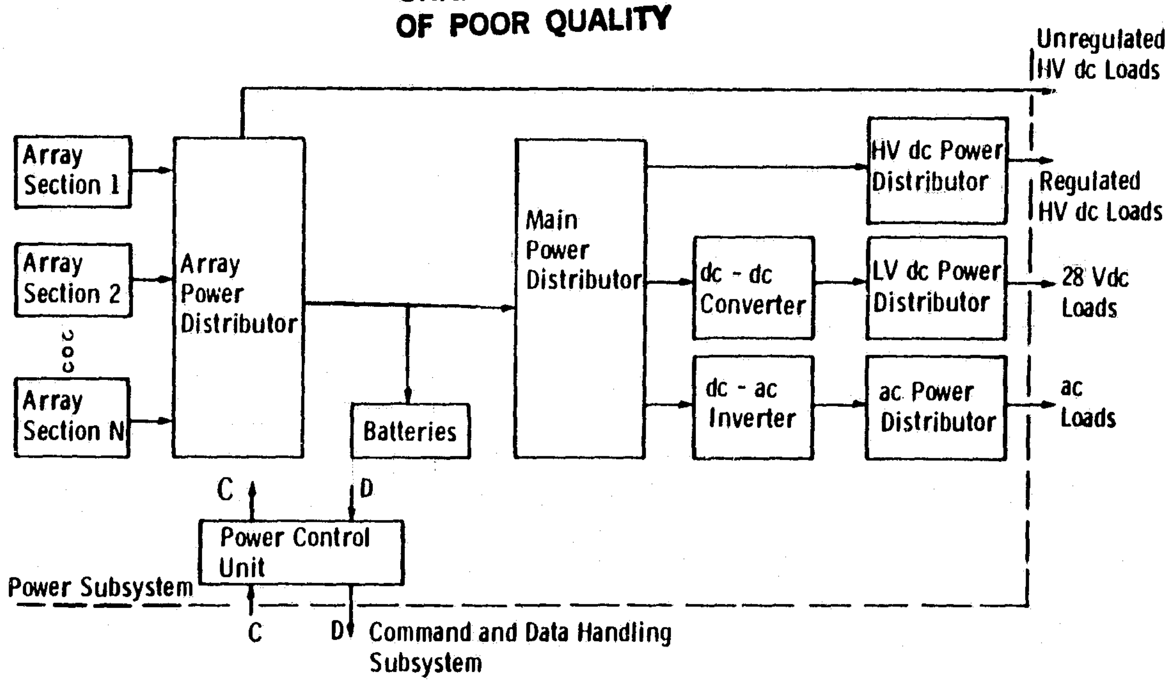


Figure 3.1.2-1
An Approach to a High-Power System to Provide HV and LV dc and ac Power

3.2 PHOTOVOLTAIC ARRAY

An array consists of a number of solar cell module strings or branches connected at the dc bus. The number of modules in series is determined by the desired dc bus voltage level, and the number of strings by the total array power required. Key factors affecting the electrical performance of the PV array are: (1) solar irradiance; (2) solar cell temperature; (3) solar incidence angle; (4) charged particle radiation; (5) reverse voltage breakdown; (6) plasma arcing; and (7) electrical wiring configuration including line resistances and bypass diodes.

The solar arrays can be classified by how they are mounted to the spacecraft and oriented to the Sun. The three basic array types are body mounted, paddle mounted, and panels mounted and Sun-oriented as shown in Figure 3.2-1. To reduce the array area, high-power multi-kW spacecraft would require array articulation capability for Sun orientation.

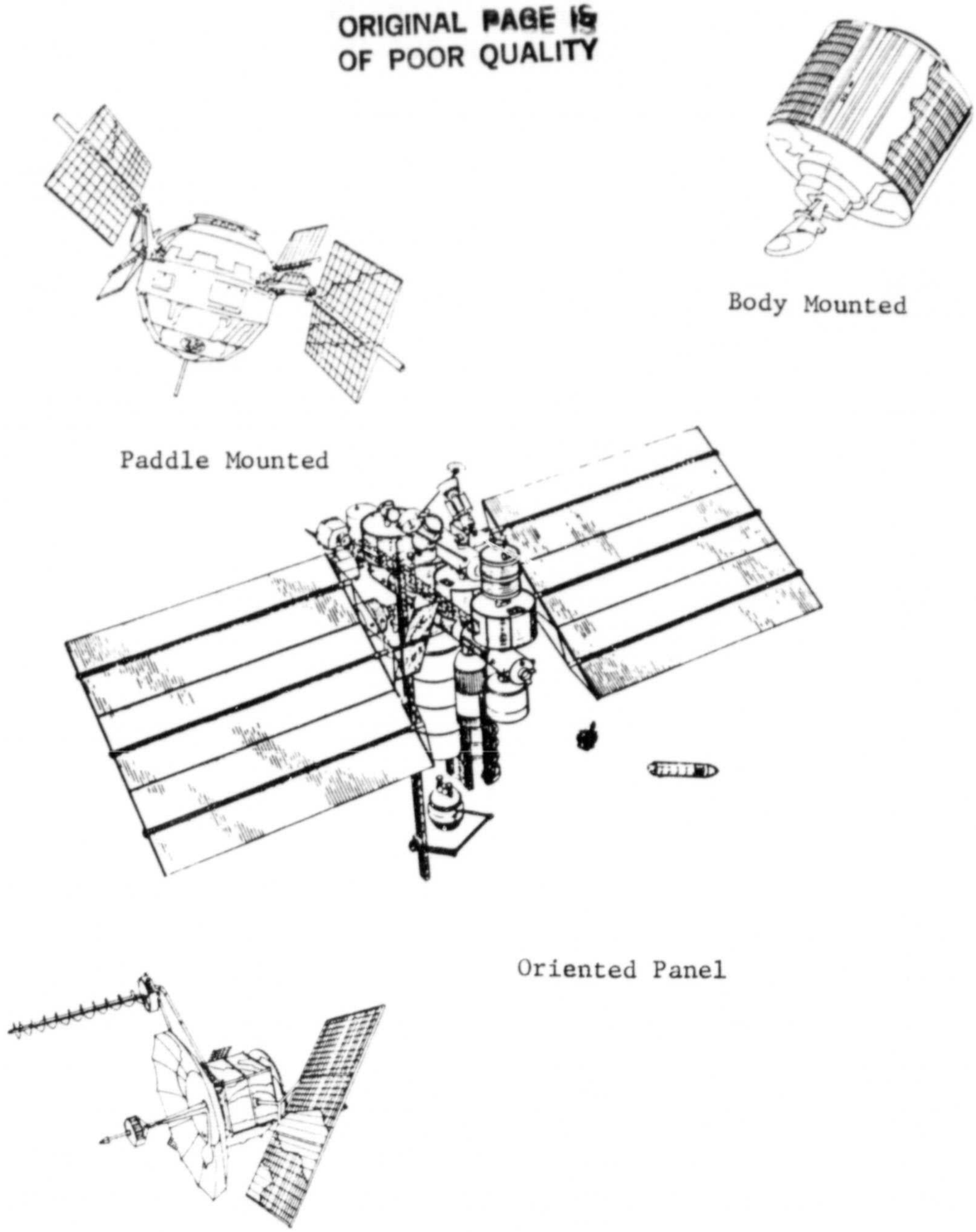


Figure 3.2-1 Basic Solar Array Configurations

The types of photovoltaic systems applicable to the space station are as follows:

1) Planar, nonconcentrating array (SEP and ultralightweight arrays),

2) Concentrating array (cassegranian and trough).

The basic features of specific candidate designs of each array type are summarized in the following subsections.

3.2.1 SEP Solar Array (Ref 16)

Description - The SEP solar array consists of five major components: array blanket, mast, tensioning mechanisms, containment box, and box cover (Fig. 3.2.1-1). The solar array wing can extend or retract fully or partially to a predetermined point. Table 3.2.1-1 lists SEP blanket physical characteristics.

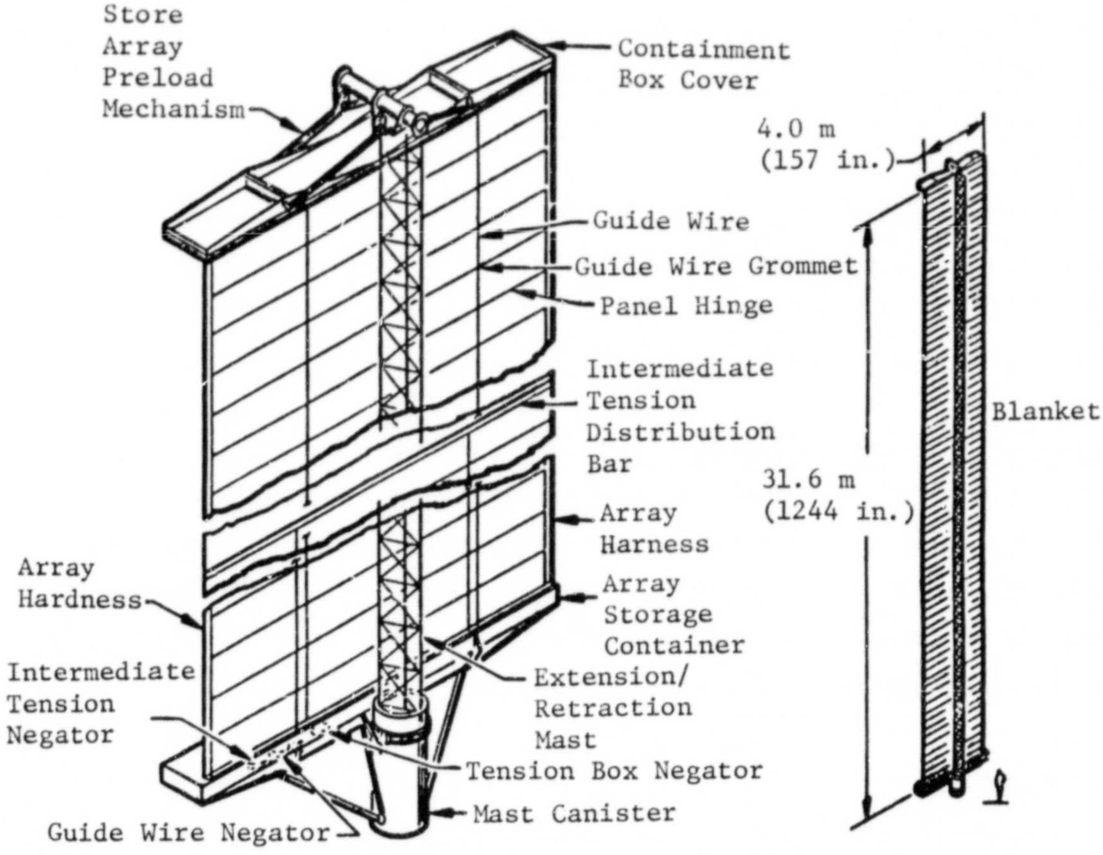


Figure 3.2.1-1 SEP Solar Array Wing

Table 3.2.1-1 SEF Array Blanket Characteristics (One Wing)

No. of Cell Assemblies/Electrical Module	1530
No. of Electrical Modules/Wing	82
No. of Cell Assemblies/Wing	125,460
Single Cell Area	8068 cm^2
Total Cell Area	101.47 cm^2
Nominal Cell Spacing (On-Array Padding)	1.09 mm (0.043 in.)
Overall Blanket Area 41x158x29.9 in.*	125 m^2 (1345 ft^2)
Cell Area Packing Factor (1.19 mm Cell Spacing)	0.887
Overall Blanket Area Cell Packing Factor	0.812
Printed Circuit Substrate Area Density (No Cells)	0.1358 kg/m^2 (0.02776 lb/ft^2)
Substrate Plus Cell Assemblies Area Density	1.0132 kg/m^2 (0.2072 lb/ft^2)
Total Blanket Plus Harness Area Density+	0.9785 kg/m^2 (0.2001 lb/ft^2)

*Includes area for array harness, panel stiffening, and panel-to-panel hinges.
+Includes hinges, panel stiffening, on-array padding, and tension-distribution bars.

The mast is a continuous Longeron lattice structure made from high temperature polyimide resin (See Table 3.2.1-2). The deployment canister used to extend and retract the mast uses two 27-Vdc motors, is 58-in. high, 16.24-in. diameter, and weighs 17.35 kg (38.17 lb).

Principal Operating Characteristics - Present-technology 25-kW SEP array uses a 12.3% efficiency solar-cell having a back-surface reflector. The solar cell also employs a dielectric wraparound contact. Table 3.2.1-3 lists solar-cell characteristics. The array system is composed of two wings, each providing 12.5-kW BOL power at 1 AU. The array sizing assumes the following losses:

- Assembly 3%
- Bussing 4.4%
- Diode 0.4%
- Present-Technology Array Design Provides 66 W/kg Using the Minimum Cell Efficiency

Table 3.2.1-2 Extension Mast Design

Mast Diameter:	37.3 cm (14.7 in.)
Mast Mass:	16.74 kg (36.8 lb)
Longerons:	
- Cross-Section:	0.553x0.572 cm (0.218x0.225 in.), Rectangular, with Corners Rounded to 0.030-in. Radius
- Material-S-Glass/Polyimide Composite Using 20-End-Glass Roving/ PMR15 Polyimide Resin	
Battens:	
- Cross section:	0.457x0.457 cm (0.18x0.18 in.), Square, with Corners Rounded to 0.030 in. Radius
- Material: Same as Longerons	
Diagonals:	3/64-in. Diameter, 3x7-Strand, Stainless-Steel Cable
Bay Length:	23.9 cm (9.0 in.)
Mechanical Properties:	
- Bending Stiffness:	62.8 kN-m^2 (21.96 x 10^6 lb-in.^{2})
- Bending Strength:	1.64 m-N (1456.3 in.-lb), Minimum Value Associated, with One Longeron In Compression
- Shearing Stiffness:	87.2 kN (19,620 lb)
- Shearing Strength:	134.8 N (30.33 lb)
- Torsional Stiffness:	1.453 kN-m^2 (5.08 x 10^5 lb-in.^{2})
- Torsional Strength:	970.7 N (218.4 lb)

Table 3.2.1-3 Present Technology 25-kW Array Solar Cell Design Features

Item	Value
Covered Efficiency (Based on Total Cell Area and 135.6 mW/cm^2):	12.3%
Diffusion Depth:	1200 to 2000 A
Cell Base Resistivity:	2 ohm/cm
Solar Cell AR Coating:	MLAR
Back-Surface Field:	No
Back-Surface Reflector:	Yes
Contact Material:	Cr-Pd-Ag or Ti-Pd-Ag
Cover Cut-On Wavelength:	350 nm,
Coverslide Material:	Fused Silica (Alternate: Ceria Stabilized Microsheet)
Cell Size:	2x4 cm, nominal
Cell Thickness:	200 micrometers (8 mils)
Cover Thickness:	150 micrometers (6 mils)
Coverslide Adhesive:	DC 93-500

Testing of the full-scale coilable longeron extension mast resulted in a mass-stiffness measurement of 15.15×10^6 lb-in.^{2} compared to the 19.6×10^6 lb-in.^{2} requirement. The associated weight increases along with the achieved cell assembly weights require a cell-efficiency increase from 11.4% to 12.3% to meet a specific power of 66 W/kg. This also reduces the number of panels per wing from 41 to 38 (25-kW array) and decreases the extension length from 32.0 m to 31.2 m.

Flight History - None; SAFE experiment is scheduled on shuttle orbiter flight in mid-1984.

Types/Manufacturer - Lockheed Missile and Space Company.

3.2.2 Ultralightweight Solar Array (Ref 17)

Description - Ultralightweight Solar Array is being developed by TRW for use in applications where existing technology is limited. This design is directed toward the following goals:

- Retractable, Redeployable
- Low Cost
- Modular/Scalable over 10 to 70 kW (BOL)
- Compatible with Automatic Fabrication/Assembly Processes

The array configuration consists of one or two flatpack foldout Kapton blankets contained in a graphite-epoxy stowage box attached to a strongback deployment structure. The blanket and container are integrated with a mast-stowage canister containing a coilable trilongeron mast for extension and retraction of the solar-cell blankets. Figure 3.2.2-1 shows the full-power two-blanket design. The total weight for the full-power design, made up of the blanket, blanket box system, and the blanket extension system combined, is 1262.8 lb (572.7 kg). Table 3.2.2-1 lists physical characteristics.

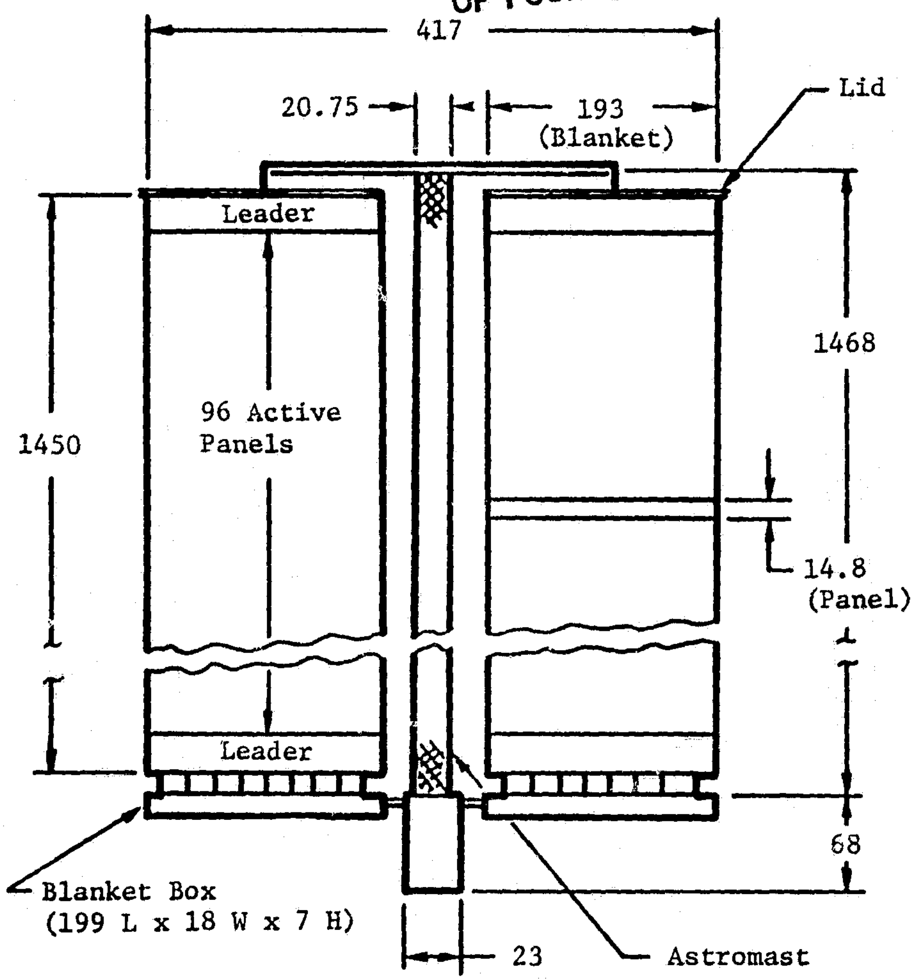


Figure 3.2.2-1
Two-Blanket Ultralightweight Solar Array (Ref 17)

Principal Operating Characteristics - The full-power, two-blanket design has a BOL power of 72 kW per spacecraft (68°C at 235 nmi, 60° inclination). End-of-life power (10 years) is approximately 17% less, or 61.7 kW per spacecraft. BOL open-circuit voltage is 425 V derating to an EOL voltage of 178 V (peak power at orbit MAX Temp of 80°C). Table 3.2.2-2 shows the array's performance analysis.

Table 3.2.2-1 Physical Characteristics, Full Power, 2 Blanket

Item	Value
No. of Wings/Spacecraft	2
No. of Blankets/Wing	2
No. of Active Panels (with Cells)/Blanket	96
Blanket Panel Size	178.3x14.8 in.
Blanket Size (Including Leader Panels)	178.3x1450 in.
Mast Deployed Length	1470
Mast Diameter	21 in.
Mast Canister Length	66 in.
Mast Canister Diameter	23 in.
Wing Width	396 in.
No. of Blanket Boxes/Wing	2
Blanket Box Size	180x18x7 in.
Deployed Wing Natural Frequency	0.04 Hz
No. of Panels/Electrical Module	2 Modules per 3 Panels
No. of Electrical Modules/Wing	128
Cell Type and Size	2 ohm-cm BSR; 4.08x2.35 cm x 8 mil
Cover Type and Size	Fused Silica, 6 mil
No. of Cells/Panel	174 x 8 = 1392
No. of Cells/Blanket	133,632
No. of Cells/Wing	267,264
Wing Weight	601 kg

State of the Art - Level 5 - 6 is estimated.

Flight History - None

Types/Manufacturer - TRW

Table 3.2.2-2 Array Performance Summary

Parameter, BOL	EOL Factor	Temp	BOL	EOL
Cell Efficiency (2-ohm-cm BSR)	--	28°C	13.3%	--
At V_{mp} = 0.49	--	28°C	490 mV	--
Cell Efficiency:				
[1-0.0046 (68-28)] 13.3%	0.85	68°C	10.9%	9.26%
At 490-2.2 (78-28)mV	0.96	68°C	402 mV	386 mV
Cell Output:				
8.57 cm^2 x 10.85% x 135.3 mW/cm^2	0.85	68°C	126 mW	107 mW
Half-Panel Output: 4p x 104s x 0.126 W	0.85	68°C	52.3 W	44.7 W
At 104s x 0.402 V	0.96	68°C	41.8 V	40.1 V
Module Output: 5 x 0.96 x 52.3W	0.85	68°C	251 W	215 W
At 5 x 0.96 x 41.8V	0.96	68°C	201 V	193 V
Blanket Output (36 Modules, 90 Panels)	0.85	68°C	9.04 kW	7.72 kW
Wing Output (4 Blankets)	0.85	68°C	36.2 kW	30.9 kW
Array Output (2 Wings)	0.85	68°C	72.3 kW	61.8 kW

Cell Size 4.08x2.10 cm = 8.57 cm^2

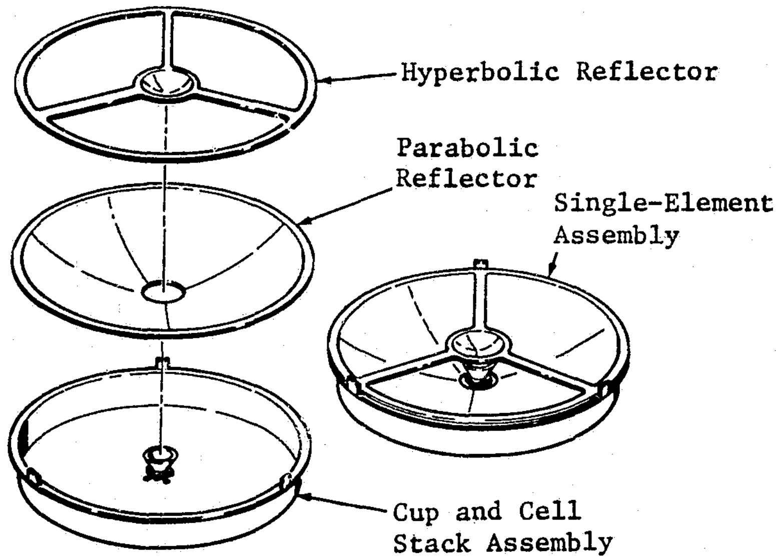
Output Values Rounded to 3 Significant Figures

Temperature Coefficient,
Power: -04.6%/°C
Voltage: -2.2mV/°C

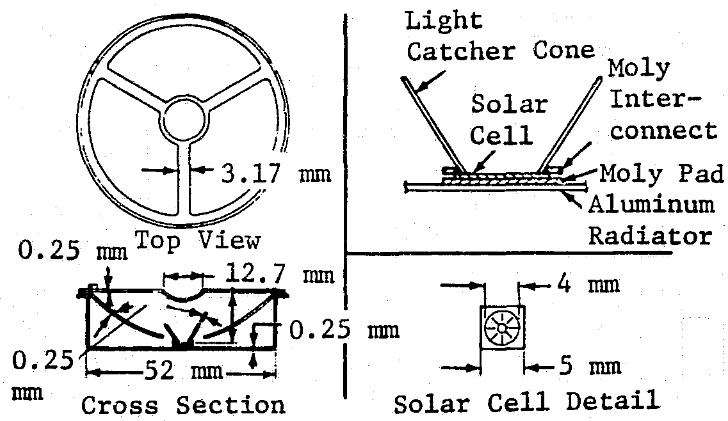
3.2.3 High Concentration Array - Cassegrainian (Ref 18)

Description - A development program is in progress (AF and NASA) for a miniaturized Cassegrainian concentrator solar array. The main interest in this type of array is to develop a multikilowatt solar array at a lower cost without sacrificing performance of present technology, and for hardening from weapon threats.

The Cassegrainian concentrator consists of a small solar cell centered in the base of a parabolic primary reflector with a hyperbolic secondary reflector mounted above the solar cell (Fig. 3.2.3-1). The solar cell is surrounded by a light-catching cone to improve performance under off-pointing conditions. Relief from thermal stress on the solar cell is accomplished by mounting it on a molybdenum base, which is then mounted to the aluminum radiator. The incident solar radiation is reflected from the primary parabolic reflector to the secondary hyperbolic reflector and finally to the solar cell.



Single-Element Assembly



Baseline Concentrator Element Design

Figure 3.2.3-1
Cassagranian Array Element Assembly (Ref 18)

The concentrator element described above is comparable in thickness to conventional panels; each element is 52 mm diameter and 13 mm thick. Several elements can be connected together for high-power use.

Principal Operating Characteristics - The Cassegrainian concentrator is in its early development stages. More testing needs to be completed before all the operating parameters are known. Table 3.2.3-1 lists present characteristics.

Table 3.2.3-1 Operating Characteristics

- Miniaturization action of concentrator results in excellent heat distribution.
- Passive thermal control provides low steady-state solar cell temperature range of 75° to 95°C.
- Effective concentrator ratio of 88 to 100.
- Reduction of recurring cost using very small solar cells in conjunction with low-cost optics.
- Primary and secondary reflectors have a common focal point, an f-number of 0.25, and a rim angle of 90 deg.
- Concentrator panel comparable area and performance (W/m^2 and W/kg) to conventional rigid solar array.
- Typical performance 100 W/m^2 and 20 W/kg with 20%-efficient solar cells.

Component-misalignment testing showed that performance falls by approximately 25% as the secondary reflector is moved 0.4 mm toward the primary reflector and remains constant as the secondary reflector is moved away from the primary reflector by as much as 0.5 mm.

State of the Art - Technology Level 4 is estimated. A nine-element demonstration module has been subjected to functional checkout tests. It has performed in a manner similar to the single-element module and is ready for comprehensive performance testing.

This type of array can use advanced high-efficiency cells for greater array performance. To date, effective concentration ratio is 88, future designs can be from 100 to 130. Future design will also have reduced blockage losses, presently at 21%.

Flight History - None

Types/Manufacturer - TRW

3.2.4 Low-Concentration Array - Trough/Pyramidal (Ref 19)

Description - The trough, or pyramidal, concept is based on a concentrator element having a four-sided, truncated pyramid configuration. Two of the reflector panels fold up with the solar panel for compact stowage. The element is designed for a geometric-concentration ratio of six suns, and can be used with silicon (Si) or gallium-arsenide (GaAs) solar cells.

The array consists of several rectangular modules with a total area of about 1400 m^2. Each module contains approximately 4400 pyramidal elements. Modules can be stored as cubes (3.24 m per side) in the Space Shuttle payload bay. The deployed module is 19.5x70.0x0.54 m. Figure 3.2.4-1 shows the module deployment stages and dimensions.

Three canister-and-mast assemblies extend from each side of the housing in two directions by connections to the end caps. The concentrator elements are supported by cables connected between the end caps and housing. The cables are maintained under constant tension through negator-cable extension mechanisms.

This type of array is expected to generate more than 300 kW of power in orbit by a single Shuttle launch. The array would comprise up to four solar-array panels, each having a power output greater than 75 kW.

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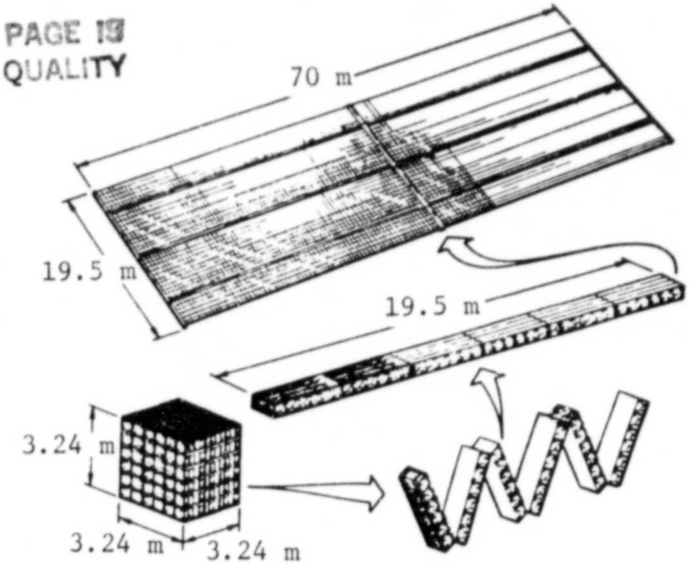


Figure 3.2.4-1
Concentrator Array Module Configuration

Principal Operating Characteristics - Two basic solar panel designs have been baselined corresponding to projected characteristics of silicon and gallium arsenide cells. Table 3.2.4-1 summarizes these characteristics.

State of the Art - This technology is estimated to be Level 3. Results to date indicate that a concentrator array module is a practical, low-cost approach for multihundred-kilowatt solar array systems for space applications. The modularity design concept can be extended to provide a hardened array configuration with gallium arsenide solar cells used for application to lower-power-level missions.

Flight History - None

Types/Manufacturer - Rockwell International

Table 3.2.4-1 Solar Panel Characteristics (Ref 19)

Parameter	Solar Cell Si	GaAs
Conversion Efficiency, % (AMO, 28°C)	14	18
Solar Absorptance	0.70	0.75
Low CR Optimized	Yes	Yes
Back-Surface Reflector	Yes	N/A
Back-Surface Field	No	N/A
Thickness, mm	0.25	0.30
Surface Dimensions, mm	50x50	19x19
Cover Type/Thickness, mm	Fused Silica, 0.2	Fused Silica, 0.2
Substrate Radiator Characteristics:		
Thickness, mm	0.6	0.5
A_R/A_P	2.0	2.0
Solar Absorptance	0.22	0.22
Emissivity	0.85	0.85

3.3 ENERGY STORAGE

Energy storage devices presented in this subsection are those that can be used for long-term operation. Included are Ni-Cd, Ni-H_2, and RFC systems.

3.3.1 Nickel-Cadmium

Description - The Ni-Cd battery consists of several hermetically sealed cells connected in series. The number of cells in a series is determined by the dc bus voltage. A 28-Vdc system usually has 22 cells, and a 240-Vdc system would require about 200 cells in series.

A typical cell is encased in a prismatic stainless steel container. It has a number of positive and negative plates insulated from each other and the metal case by separator material. Potassium hydroxide is normally used as the electrolyte. Reference 20 provides a detailed description of design, manufacturing, and operational characteristics of the Ni-Cd cell.

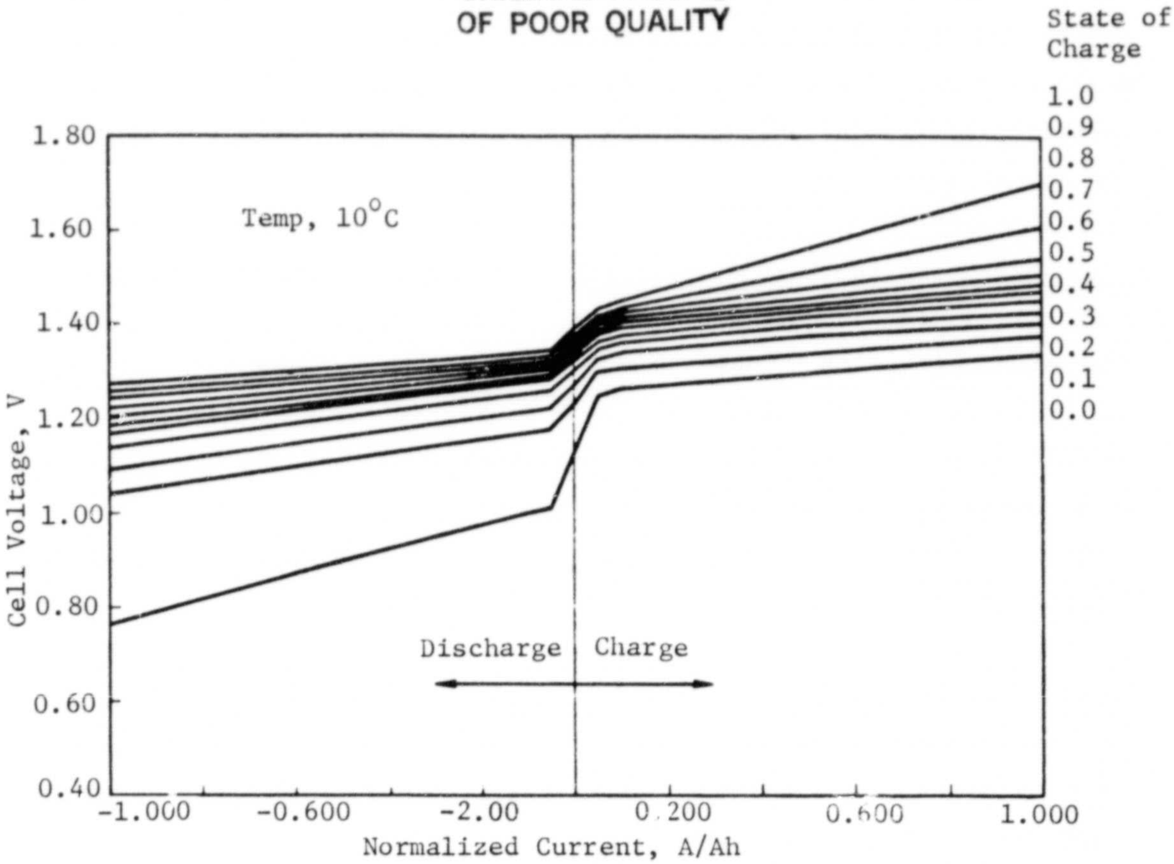
Principal Operating Characteristics - The operating characteristics of a nickel-cadmium battery are a function of state of charge, depth of discharge, number of cycles, the duration of charge/discharge cycles, and operating temperature. All these variables are controllable to a certain extent either directly or indirectly. Because of the large uncertainty in the performance behavior of Ni-Cd battery (and all others), battery operation management is one of the best candidates for automation via computers.

Typical charge-discharge voltage profiles are shown in Figure 3.3.1-1 as a function of state of charge. The desired range of charge voltage limit can vary from 1.40 volts to 1.60 volts, and discharge voltage is about 1.2-Vdc average per cell.

Figure 3.3.1-2 depicts one set of cycle-life data (Ref 20) available on an LEO mission. These data, as well as others in open literature, are based on 5-cell to 22-cell battery pack testing. Thus, a lot of uncertainties exist in projecting the life of a possible 200-cell battery pack configuration of the space station batteries.

Figure 3.3.1-3 shows the mass of the Ni-Cd cell from several suppliers as a function of rated capacity (36 to 41 gm/Ah).

State of the Art - Sealed nickel-cadmium cell batteries were developed for space applications. They have served as a reliable energy-storage system for the majority of spacecraft flown.



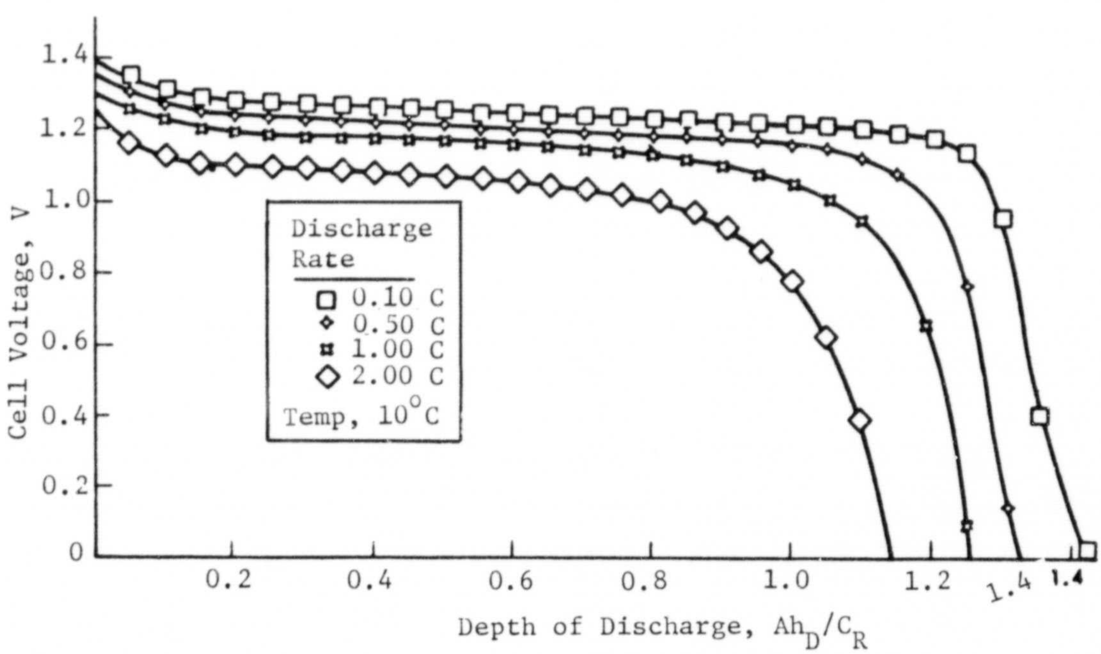


Figure 3.3.1-1
Typical Charge-Discharge Profiles of Ni-Cd Cell

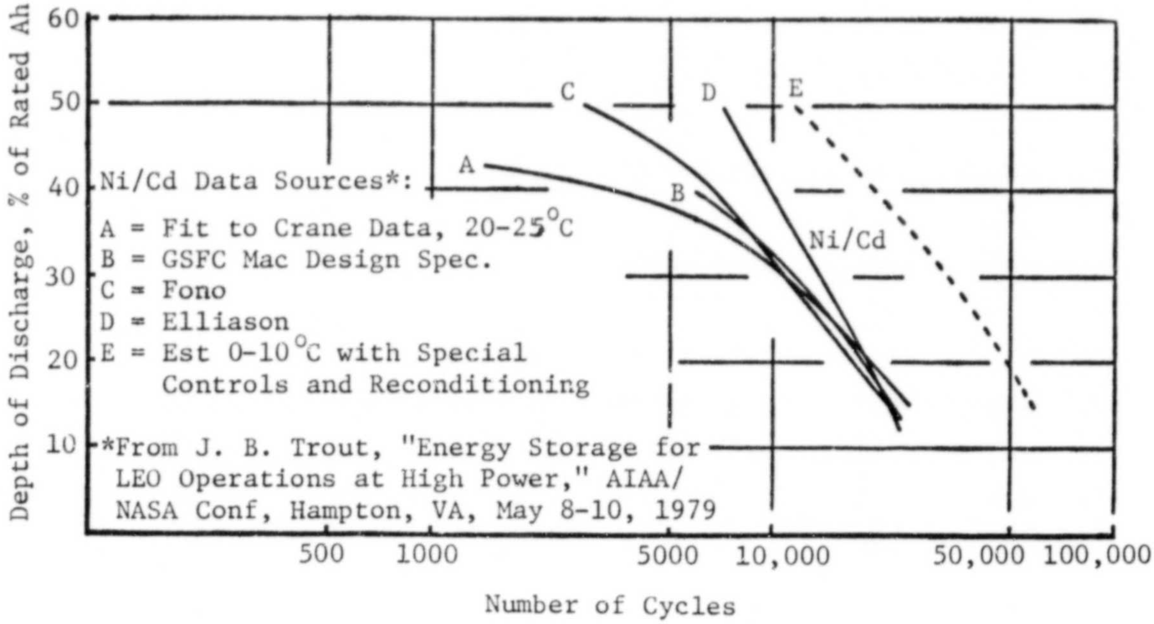


Figure 3.3.1-2
NiCd Battery Cycle Life Projection for LEO Application (Ref 13)

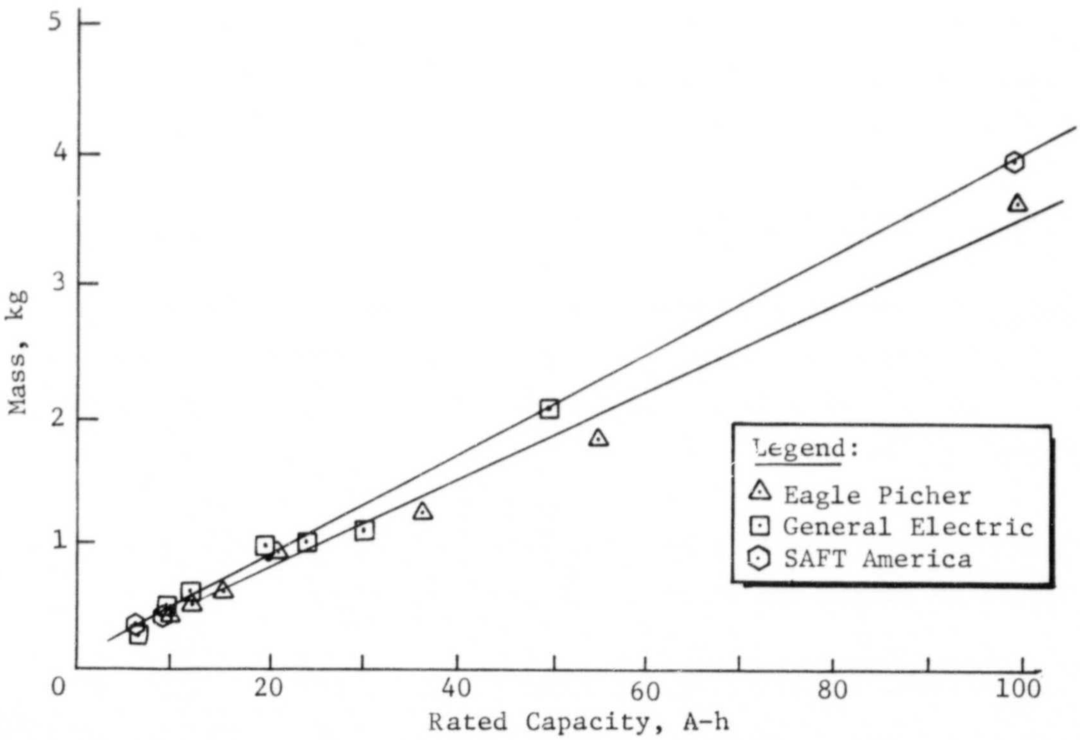


Figure 3.3.1-3 Relationship of Mass to Capacity for Spacecraft NiCd Cells

Recently, the primary advances have been in the areas of:

- Seal Improvement for Reliability
- Increased Cell Capacity
- Specific Energy Improvements
- Lightweight Container Designs

Major emphasis for advanced technical development efforts has been on: (1) reduced weight for geosynchronous and medium-altitude spacecraft, (2) increased life capability to more than 10 years at 85% depth of discharge for GEO, and (3) increased life to more than five years for LEO applications.

Flight History - Nickel-cadmium batteries have been flown on most spacecraft requiring long-life operation.

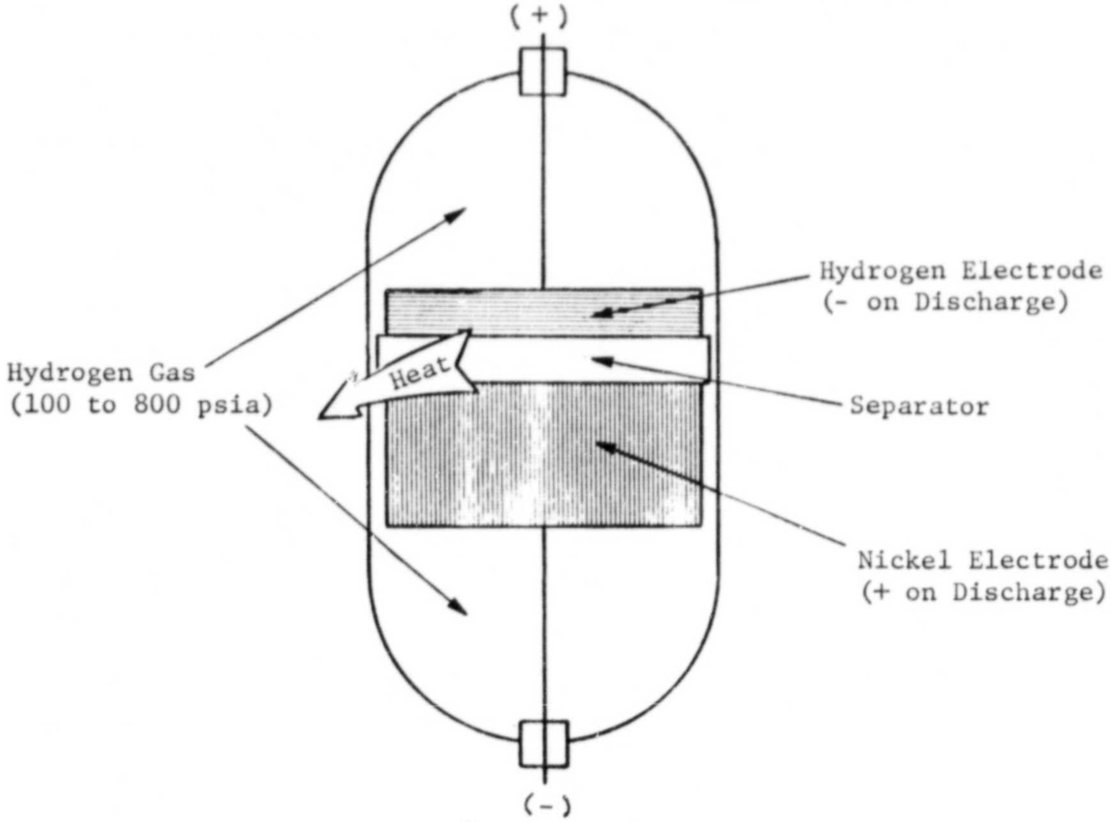
Types/Manufacturer - The primary suppliers of nickel-cadmium cells for aerospace use are General Electric, Eagle Picher, and SAFT America. Several sizes, up to 50 Ah, are now available.

3.3.2 IPV and CPV Nickel Hydrogen Battery

Description - The nickel-hydrogen cell is contained in a hermetically sealed pressure vessel (Fig. 3.3.2-1). It is a derivative of the Ni-Cd cell design via substitution of the negative electrode (from cadmium to hydrogen).

Nickel-hydrogen systems, like other batteries, require multiple cells in series to attain the necessary bus voltage.

Two basic types available are referred to as the individual pressure vessel (IPV) and common pressure vessel (CPV). The CPV design contains several cells connected in series within one common pressure vessel.



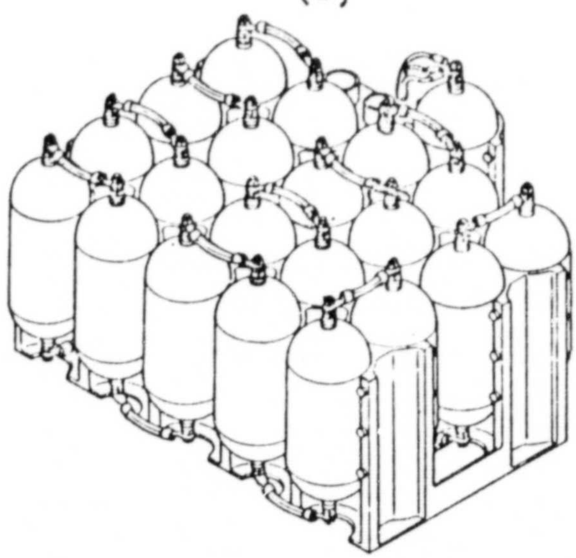


Figure 3.3.2-1
Schematic of a Ni-H_2 Cell and Typical Battery Arrangement

Principal Operating Characteristics - Figures 3.3.2-2 and 3.3.2-3 show charge/discharge curves for a typical Ni-H_2 cell. Internal pressure in a nickel-hydrogen cell varies linearly with state of charge.

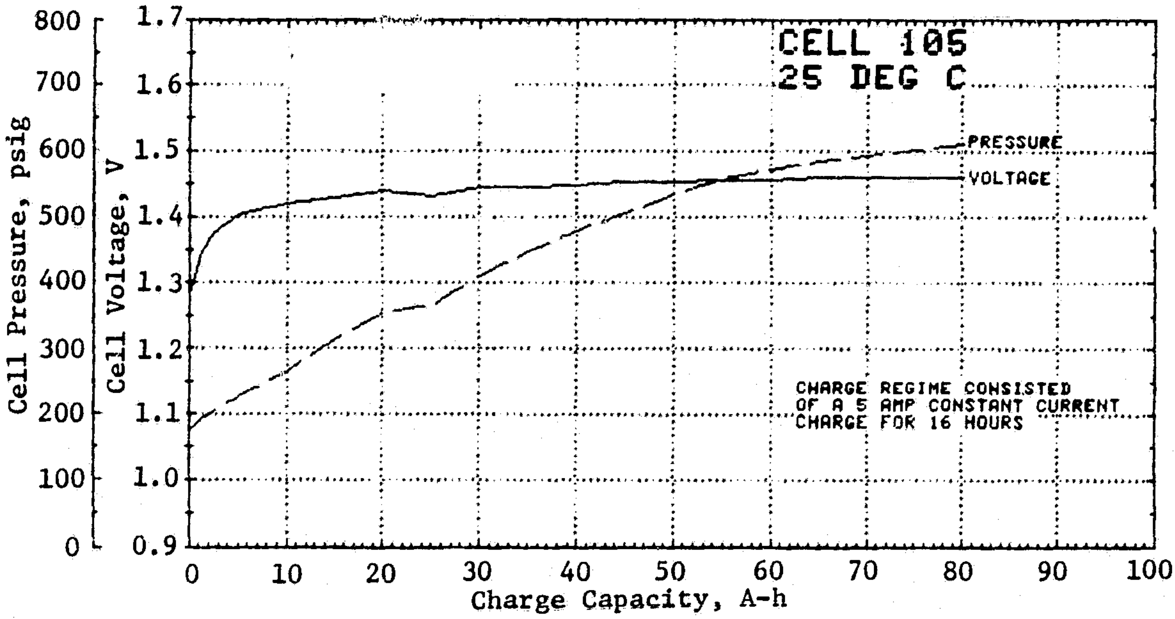


Figure 3.3.2-2 Typical Charging Characteristics of Ni-H_2 Cell (Ref 21)

Table 3.3.2-1 presents the physical characteristics for Yardney 30-A-h and 50-A-h nickel-hydrogen cells. These cells are similar in size and shape to cells of other vendors.

State of the Art - COMSAT Laboratories initiated the exploratory development of nickel-hydrogen cells in early 1970, followed by the Air Force in 1972. Since then, primary development occurred in the following areas:

1) Lightweight cells,

2) Basic cell design,

3) Production capability for electrochemically impregnated nickel electrodes,

4) Common pressure vessel.

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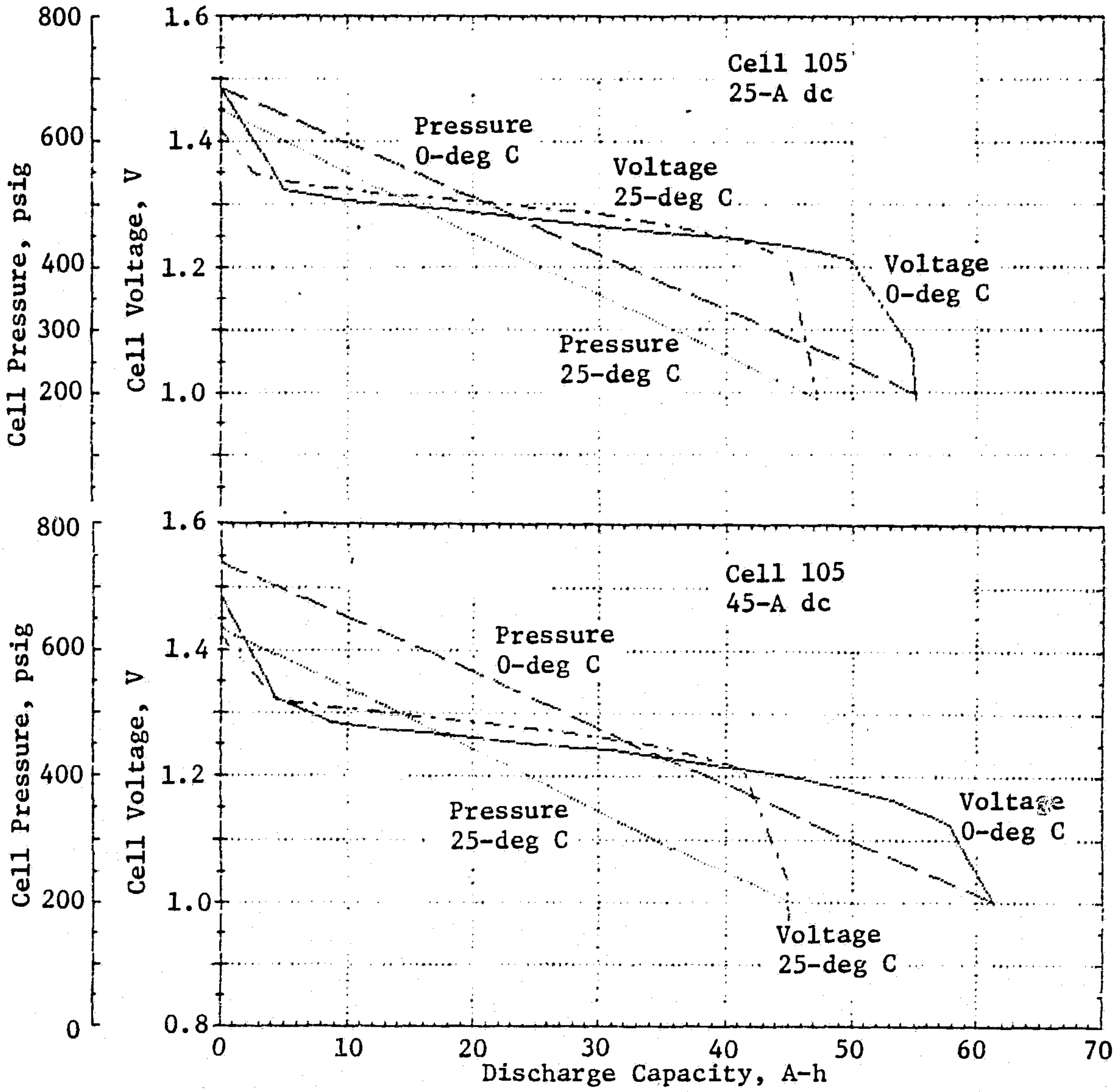

Figure 3.3.2-3 Typical Discharge Characteristics of Ni-H_2 Cell (Ref 21)

Table 3.3.2-1 Physical Characteristics

	YNH 30-2	YNH 50-3
Weight:	1.96 lb (887 g)	2.79 lb (1270 g)
Volume:	46.4 in.^{3} (715 cm^3)	52.3 in.^{3} (857 cm^3)
Length:	8.0 in. (20.3 cm)	9.0 in. (22.9 cm)
Diameter:	3.5 in. (8.9 cm)	3.5 in. (8.9 cm)

Flight History - Nickel-hydrogen batteries were launched in 1976 on the Navy NTS-2 satellite and the Air Force flight experiment satellite. Nickel-hydrogen batteries are planned for the following spacecraft:

1) Intelsat V and VI Communication Satellite;

2) U.S. Air Force SDS Satellite;

3) GTE "G-Start" Satellite;

4) Southern Pacific "Spacenet" Satellite;

5) ESA "L-Sat" Satellite.

Types/Manufacturer - Yardney Electric Corporation and Eagle Picher Co.

3.3.3 Bipolar Nickel Hydrogen Battery

Description - Bipolar NiH_2 cells provide a concept more closely resembling a fuel cell system than a traditional nickel-cadmium battery pack. This modular concept with projected energy densities of 44 to 53 W-h/kg (20 to 24 W-h/lb) and 700 to 900 W-h/ft^3, has significant potential improvements in reliability, energy density, cycle life, and cost (Ref 22, 23). The nickel-hydrogen battery using bipolar construction in a common pressure vessel is shown in Figure 3.3.3-1.

Principal Operating Characteristics - The basic specifications for a 35-kW battery are listed in Figure 3.3.3-2. The weight estimates for this battery are listed in Table 3.3.3-1.

State of the Art - A preliminary design of a 35-kW nickel-hydrogen battery featuring bipolar construction, a common pressure vessel and active cooling is being developed for possible applications requiring high power energy storage.

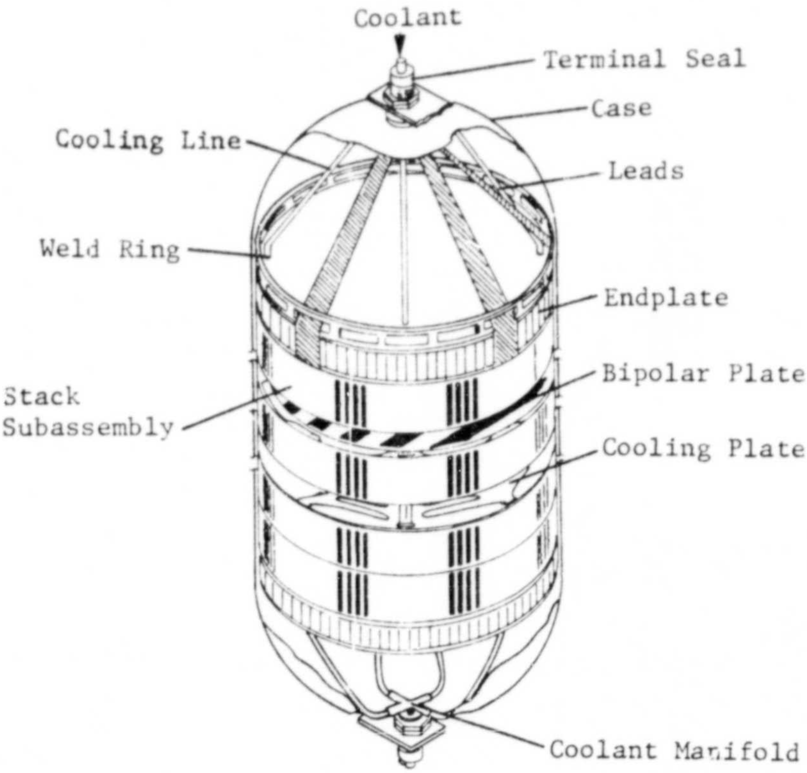

Figure 3.3.3-1 Bipolar Ni-H_2 Cell (Ref 22)

The inherent characteristics of the bipolar concept lends itself to a high voltage low current operation. Using a common pressure vessel for the entire battery offers significant improvement in both gravimetric and coulometric energy densities. In addition, spacecraft/battery integration is a simpler task when considering that this one 35 kW module (or a modified modular concept) would replace many cells in a series configuration.

Flight History - None

Types/Manufacturer - Hughes Aircraft Co.

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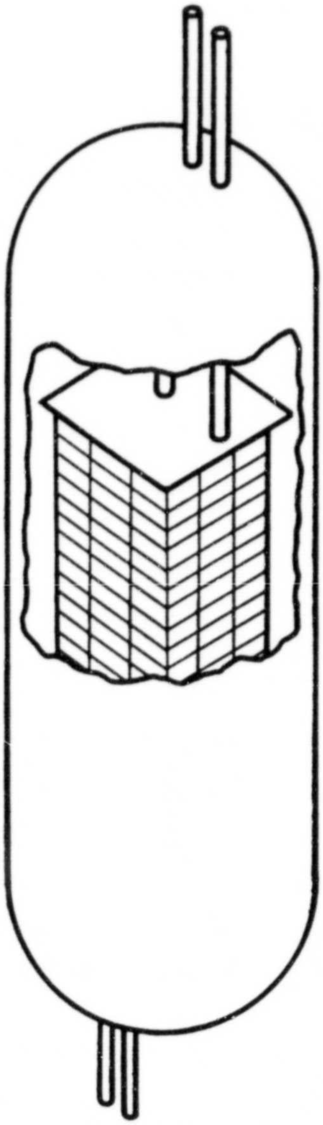

Battery Specifications

- Power	35 kW
- Load Voltage	275 V
- Current	127 A
- Dicharge Capacity at 0.6-h Eclipse	76
- Depth of Discharge	70%
- Theoretical Capacity	128 A-h
- Series Cells	229
- Plate Area	625 $in.^2$
- Cell Thickness	0.095 in.
- Stack Dimensions	27 x 27 x 28 in.
- Battery Weight	1583 lb
- Energy Density	19 W-h/lb at 100% DOD
- Volumetric Energy Density	780 $W\text{-}h/ft^3$
- Vessel Configuration	TBD

Figure 3.3.3-2 35-kW Bipolar Ni-H_2 Battery Specification (Ref 22)

Table 3.3.3-1
Estimated Weight Breakdown of a 35-kW Bipolar Ni-H_2 Battery

Component	Total Weight	% of Total
Nickel Electrodes	508 lb	32.5%
Hydrogen Electrodes	70	4.5
Separators	35	2.0
Electrolyte Reservoir Plates	185	12.0
Recombination Grids	15	1.0
Cooling Plates	180	11.5
Pressure Vessel	200	13.0
Electrolyte	246	16.0
Hardware (Tie Rods, Terminal Cables, Coolant Lines, Etc)	30	2.0
Foam	10	0.6
Frames	54	3.4
Coolant	20	1.2
End Plates	30	2.0
Total Weight	1585 lb	100.0%

3.3.4 Regenerative Fuel Cell (RFC)

Description - Regenerable fuel cell systems produce electricity by combining reactants by direct electrochemical process to generate electricity and water. The most well-developed system is H_2O_2.

The basic elements of a hydrogen-oxygen regenerative fuel-cell system are shown in Figure 3.3.4-1. The principal parts are the fuel cell and the electrolysis module.

The fuel-cell module converts H_2 and O_2 directly into dc power with water as the byproduct. The electrolysis unit essentially splits this water into gaseous H_2 and O_2, thus resulting in a reversible reaction. Heat exchangers remove waste heat from the electrolysis and fuel-cell modular water coolant loops, each having temperature-regulating valves. A condenser removes heat from the generated O_2 and H_2 gases such that the outlet saturation temperature or dew point is below the temperature of the storage tanks. Similarly, a product-water heat exchanger reduces the temperature of water discharged by the fuel-cell module to a desired value for storage. The process water outlet temperature of the heat exchangers is independently controlled by temperature-regulating valves.

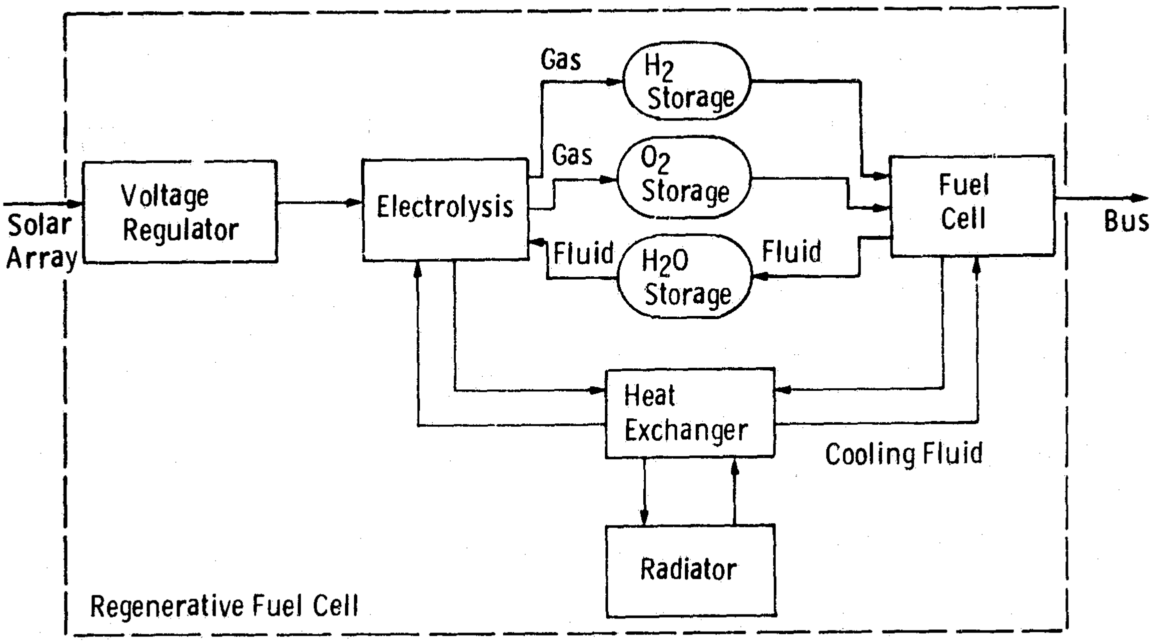

Figure 3.3.4-1 Block Diagram of a Regenerative Fuel Cell (Ref 24)

Principal Operating Characteristics - There are approximately ten contributors to energy-storage inefficiency with the RFC system (Ref 24): (1) fuel-cell voltage loss; (2) fuel cell faradaic inefficiency; (3) fuel-cell ancillary power; (4) fuel-cell discharge regulator power loss; (4) electrolyzer voltage loss; (6) electrolyzer faradaic inefficiency; (7) electrolyzer ancillary power; (8) electrolyzer input power regulator loss; (9) inefficient use of solar-array charging area; and (10) power consumption for temperature control.

For either a solid polymer electrolyte fuel cell or an alkaline fuel cell, a design energy-storage efficiency for the RFC system of 60% is considered possible without undue development risk.

One of the findings by United Technologies was that the specific weight did not change much for 35-kW and 250-kW systems which were 55.1 lb/kW and 51.1 lb/kW, respectively.

State of the Art - The basic space fuel cell after its emergence as a primary power source in the early 1960s has had, and continues to have, a steady and evolutionary technical growth. It very successfully provided the electrical primary power for the Gemini and Apollo programs and now must be examined as to its role in projected new large space power systems. It is expected that the large level of effort being directed to the development of fuel cells for terrestrial applications will indirectly affect space fuel-cell technology and could possibly affect its projected role in future space missions (Ref 25).

The state-of-the-art fuel cell of today is largely the product of technology-development efforts aimed at meeting particular mission requirements in a particular time frame. Fuel cells were developed in the early 1960s because of the special requirements of the Apollo vehicle. After this major step in technology advancement, the fuel cell became a more mature technology and made a steady technology growth toward lighter weight, higher specific power, lower cost, and longer life.

The specific weight decreased from 89 lb/kW for Apollo to 8 lb/kW for the Shuttle Orbiter (Ref 25). The advanced lightweight fuel cell has potentially greater specific weight reduction to 4 lb/kW. During this same period in which large reductions in specific weight and specific cost were achieved, there were corresponding increases in operating life from 100 to more than 2500 hours.

The fuel cell of today is an operational and reliable electromechanical power source. It was developed for NASA's manned missions in the 1960s because the conventional battery systems could not meet the energy-density requirements. Although the role of the fuel cell as a primary source for space power appears limited, it may have a much larger role as an energy-storage subsystem when combined with the electrolyzer.

Present studies have shown that the H_2O_2 space fuel cell with a dedicated electrolyzer can be competitive with NiCd and NiH_2 batteries as energy-storage subsystems for large space power-system applications.

Flight History - The basic fuel cells successfully provided the electrical primary power for the Gemini and Apollo programs. The RFC has not been flown.

Types/Manufacturer - GE and United Technology Corp.

3.4 POWER CONDITIONING

3.4.1 Series Resonant Converter

Description - The design of this type of converter is based on the controlled transfer and transformation of electric energy through series-resonant circuits at frequencies in excess of 10 kHz. Figure 3.4.1-1 is a schematic of a half-bridge converter. The high-Q series-resonant circuits continuously oscillate and are controlled by adjustment of the phase angle between the exciting voltage and the resonant current (Ref. 26). This topology is highly efficient because only a small fraction of the energy transferred to the load is absorbed by the resonant circuits. The system is suited for construction of low-cost, submegawatt, single-module converters using available components.

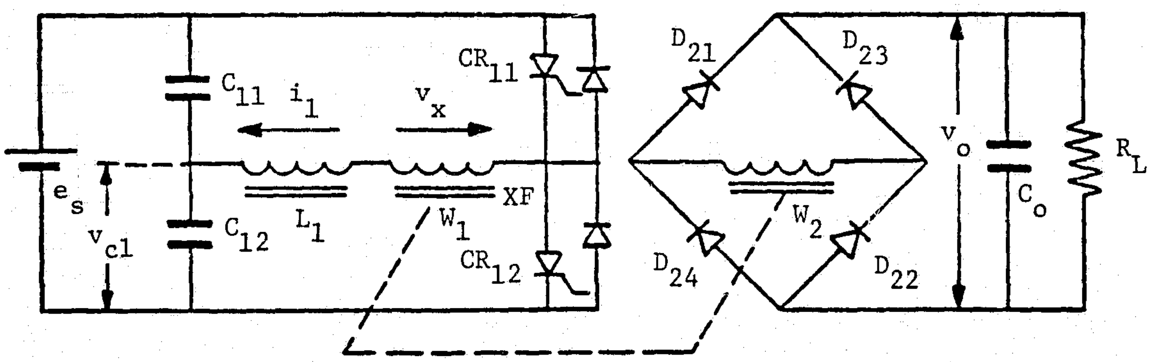

Figure 3.4.1-1 Half-Bridge Converter

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Principal Operating Characteristics - Higher energy density and efficiency are expected owing to high-frequency operation (10 to 30 kHz) than the lower-frequency rectangular-wave converter. High-frequency operation allows the inductive and capacitive energy-storage devices to be smaller than those used in lower-frequency converters, a reduction that results in significant size and weight savings. Higher-frequency operation in the series resonant converter is possible because a series-resonant current, rather than rectangular pulses, is conducted through the control-semiconductor power switch. The power switches are controlled so that they switch on and off when the current through the switch is very close to zero, thus allowing very low switching losses.

Figure 3.4.1-2 shows a simplified schematic of a twin-full-bridge version. Operation and control methods are similar in that the operating principle is merely an extension from the half-bridge operation.

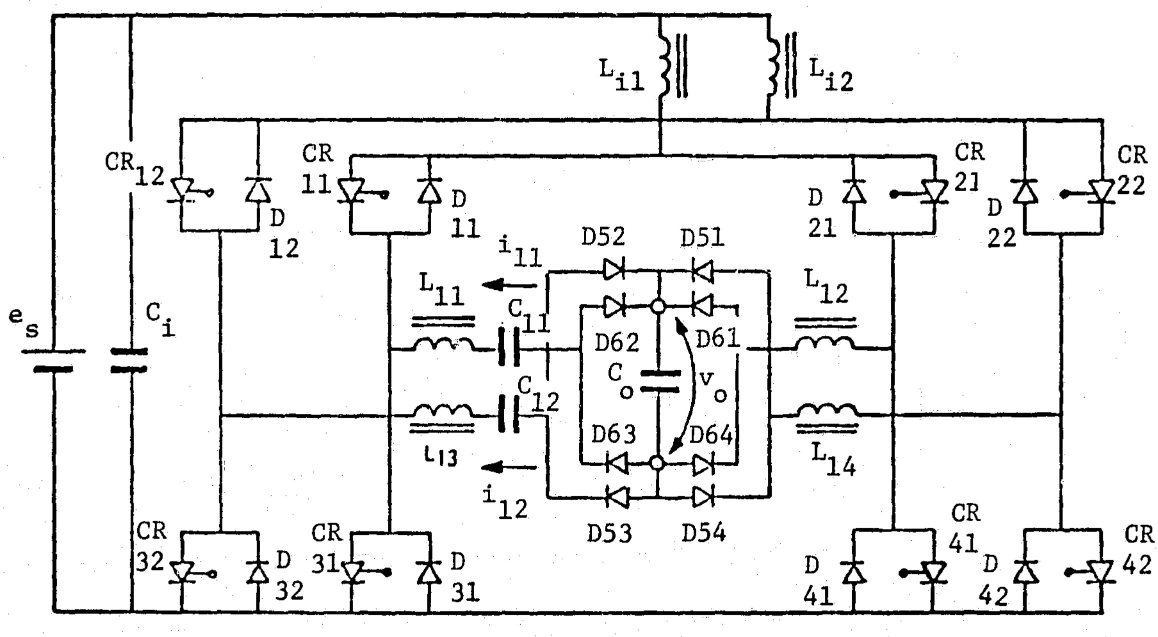

Figure 3.4.1-2 Twin Full-Bridge Converter Configuration

A dc-ac version and a 3-phase ac-dc version exist as well. Operating parameters for all these configurations are listed in Table 3.4.1-1.

Table 3.4.1-1
Operating Parameters of Existing Series-Resonant Configurations

Type	V_{In}	V_{Out}	Power
Half-Bridge Dc-Dc	200-400 V	200, 25 kV	100 kW
Twin Full Bridge	200-400 V	400, 25 kV	200 kW
Dc-Ac	200-400 V	208 Vac	5 kW
Ac-Dc	100-208 V	200, 25 kV	5 kW

Estimated efficiencies for the dc-dc types may range as high as 97 to 98% due to the reduced switching losses inherent in this topology.

State of the Art - The basic operating principles are known and have been demonstrated; however, development and improvement are still needed. Studies are presently underway that focus on developing standardized control and protection circuitry as well as to identify potential problems with space applications. Hybrid technology and microprocessor applications for control also are being examined by Martin Marietta under the AFAPL contract.

Flight History - None

Types/Manufacturer - None; under development by AFAPL.

3.4.2 Dc-Ac Inverter

Description - An inverter is a power-conversion device used to transform dc power to ac power. Power-conversion circuits consist basically of some type of "chopper" used to develop a waveshape that is acceptable to a transformer. The switching function in the inverter circuit is usually performed by high-speed transistors or silicon-controlled rectifiers (SCR) connected in series with the primary winding of the output transformer. Figures 3.4.2-1 and 3.4.2-2 show two different types of inverters, push-pull and resonant, respectively.

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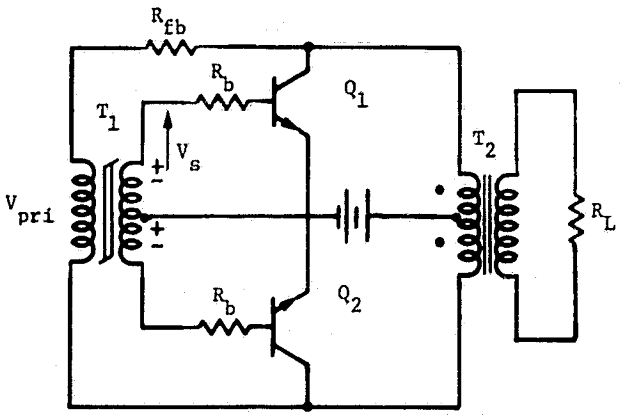

Figure 3.4.2-1
Two-Transistor, Two-Transformer Push-Pull Switching Inverter

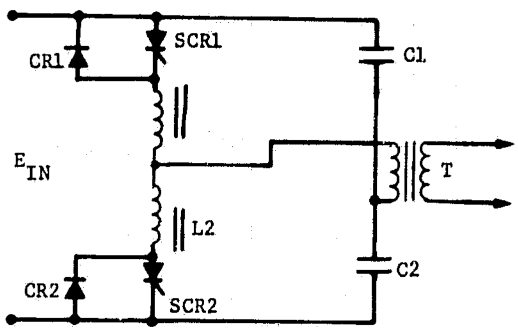

Figure 3.4.2-2
Series L-C Resonant Inverter

Transistor and SCR inverters can be made very lightweight and small in size. They are also highly efficient circuits and have no moving parts.

Principal Operating Characteristics - Dc-ac inverters show promise in applications involving large space-power systems. A study of the multihundred-kWe space system by General Dynamics (Ref 27) points out that the first choice for general-purpose, space-platform application is a hybrid-ac/dc, centralized, and distributed configuration (Fig. 3.4.2-3). This system's major features are listed in Table 3.4.2-1.

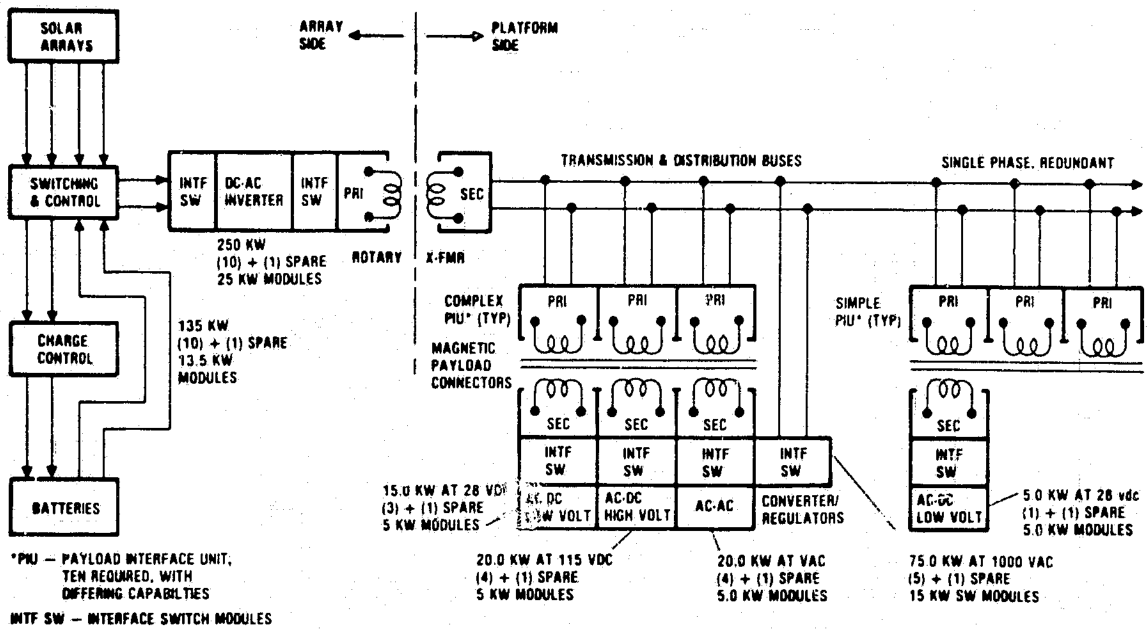

Figure 3.4.2-3 Ac-Dc Hybrid Resonant System (Ref 27)

Table 3.4.2-1 Ac-Dc Hybrid Resonant System Features

- Modular Design and Construction Sized for Minimum Weight/Life-Cycle-Cost
- High-Voltage Transmission (1000 Vac RMS)
- Medium-Voltage Array (440 Vdc)
- Resonant Inversion
- Transformer Rotary Joint
- High-Frequency, Single-Phase Transmission Line (20 kHz)
- Energy Storage on Array Side of Rotary Joint
- Fully Redundant
- 10-Year Life with Minimal Replacement and Repair
- Recurring Life-Cycle Cost = $28 per Pk Watt

State of the Art - The inverters for high-power space application do not exist.

Flight History - None

Types/Manufacturer - None; potential suppliers include:

- Helionetics, Inc
- General Dynamics and Astronautics
- Martin Marietta
- TRW

3.4.3 Switched-Mode Dc-Dc Buck Converter

Description - This type of converter is used often in spacecraft applications. Advances have been made toward automating this type of system, the best example being the Programmable Power Processor (P^3) (Ref 6). It is an autonomous, 18-kW power processor for use in large high-power spacecraft power systems. Operation as a voltage regulator, battery charger, shunt regulator, or power limiter is achieved by selection of the resident ROM. The P^3 is also flexible in other areas such as the command and data interface. With selection of the appropriate interface card, a single P^3 can operate in different modes and with almost any spacecraft interface. Table 3.4.3-1 summarizes its main features.

Table 3.4.3-1 P^3 Functional Capability

Battery Control
- Battery Charger
- Peak Power Tracker (Solar Array)
- Caution and Shutdown

Bus Voltage Control
- Voltage Regulator
- Caution and Shutdown

Power Limiter (Shuttle Power Extension Package)
- Peak Power Tracker
- Fuel-Cell Current Limiter
- Caution and Shutdown

Power Bus Overvoltage Protection
- Shunt Regulator
- Caution Shutdown

Figure 3.4.3-1 shows the functional block diagram of P^3. The input and output power are connected through two 4-pin, 50-A connectors. The 78-pin patchplug connectors and 15-pin analog measurement connector are provided. The package weighs 62 lb, and the volume is 1.17 ft^3.

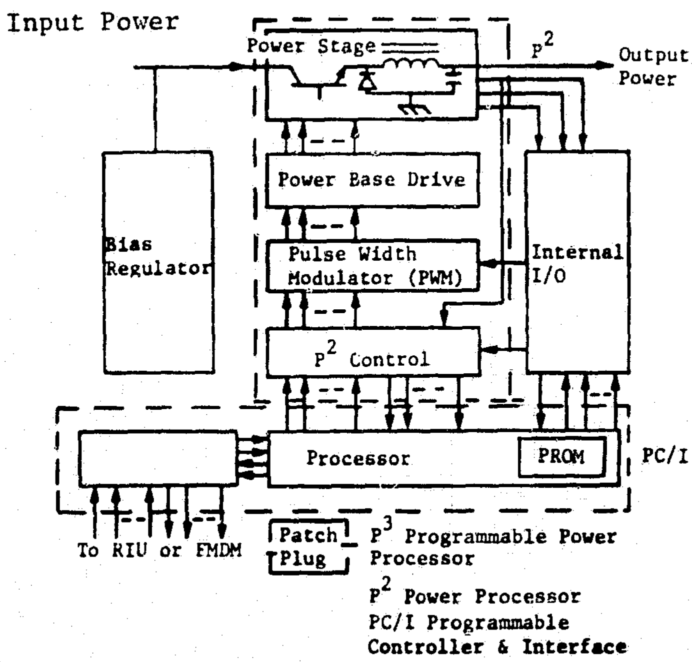

Figure 3.4.3-1
Functional Block Diagram of P^3

The power section contains three parallel power stages, which are controlled with a 100-kHz-pulse width-modulated drive circuit. Output voltage ripple is minimized by operating the three stages 120 degrees out of phase with respect to each other.

The microprocessor used in the P^3 is a TISB9900. This was selected because it was available in I^2L technology which has low radiation susceptibility. The 9900 uses a 16-bit data bus and hardware multiplication and division.

Control parameters and caution-and-shutdown parameters can be changed in flight by ground control using command-adjustable parameters.

Principal Operating Characteristics - High or low power levels may be achieved with P^3 by connecting several P^3s in parallel without hardware modification. Ten P^3s connected in parallel can produce up to 28 kW at 28-Vdc output; one P^3 may be used if 3 kW or less are required. Table 3.4.3-2 lists the electrical characteristics of P^3. Figure 3.4.3-2 shows the efficiency as a function of the output current at several input voltage levels.

State of the Art - The hardware and software for an autonomous 18-kW programmable power processor have been developed, integrated, and verified at ambient conditions. The power processor has been demonstrated to be capable of output voltages of 30 to 180 Vdc, at output currents of 0 to 10 Adc, and for input voltages up to 375 Vdc. Software for both the voltage-regulator and battery-charger/battery-management modes has been successfully tested. Mode selection and telemetry scaling via patchplug has been accomplished. The P^3 system has been demonstrated with both an RIU and an FMDM interface. An autonomous operation has been successfully demonstrated in the areas of automatic state transition, interface initialization, caution-and-shutdown monitoring, telemetry acquisition, processing and display, overload protection, battery management and protection, and peak-power tracking. A complete mechanical design for the P^3 has been developed. An engineering model has been electrically tested, and environmental testing is underway.

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Table 3.4.3-2 Summary of P^3 Capabilities (Ref 6)

Parameter	Level	Notes
Output Voltage, V_0	24 Vdc to 180 Vdc	Programmable
Output Current, I_0	0 to 100 Adc	
Input Voltage Steady State, V_{In}	26 Vdc to 375 Vdc	
Transient Voltage Limitation	400 Vdc, 20 s	
Output Voltage Ripple	50% of SL-E-0002A Conducted Susceptibility for V_0 = 30 Vdc	For V_0 = 30 Vdc Allowable Ripple Rises Proportionally
Internal Power Dissipation That Must Be Acceptable to Mechanical Design	600 W	
Fast-Response Hardware Overload Protection	105 to 115 Adc Limiting Occurs within 10 s of Overload	Protection Circuit Will Override Microcomputer
Hardware Overvoltage	Programmable between 26 & 200 V	Protection Circuit Will Override Microcomputer
Maximum Standby Power	140 W	

Flight History - None

Types/Manufacturer - Martin Marietta/NASA MSFC

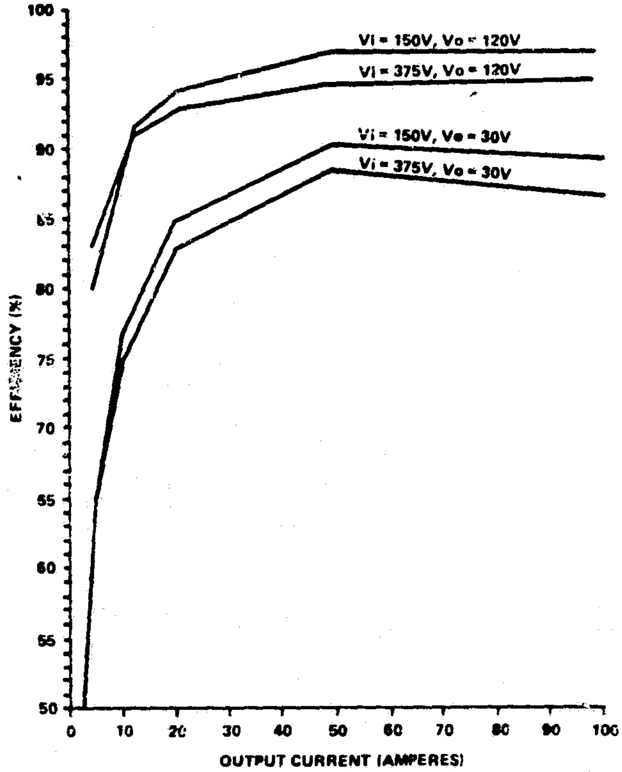


Figure 3.4.3-2
P^3 Efficiency vs Output Current, Input Voltage a Parameter

3.4.4 Transformer Coupled Converter

Description - The transformer-coupled converter (TCC) was developed by LMSC (Ref 31) for use on the Space Shuttle Power Extension program. This converter meets the weight and efficiency requirements for space applications and is capable of converting power from high-voltage solar arrays. The converter topology used is the full-bridge transistor-transformer-coupled design. The TCC block diagram is shown in Figure 3.4.4-1. The D60T high-voltage transistor is used in the baseline design because of its superior ratings.

The complete system consists of two independent bridge-converter modules having their own independent regulator, analog-control subsystem, digital-control subsystem, and peak power tracker. The unit dimensions are 20x20x7 in. and the weight is 67 lb.

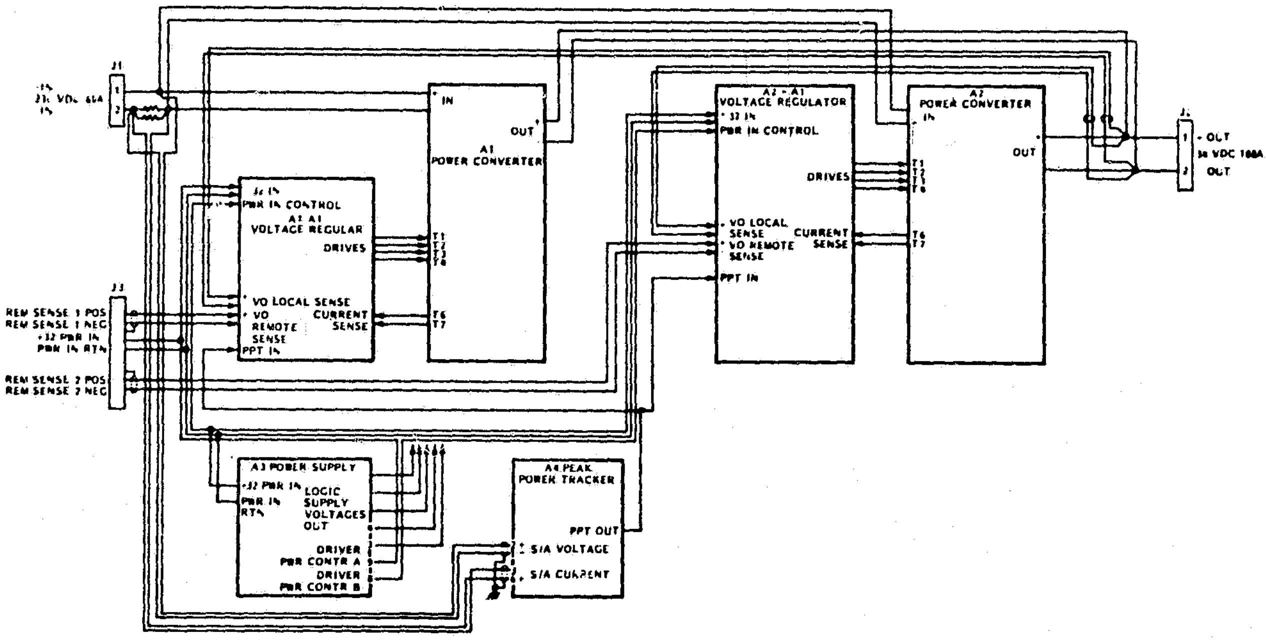


Figure 3.4.4-1 TCC Block Diagram (Ref 31)

Principle Operating Characteristics - The basic electrical characteristics of TCC are listed in Table 3.4.4-1.

Table 3.4.4-1 TCC Specifications

Requirements	Design Goals
Input Voltage: 111 to 234 Vdc PEP-Solar-Array Compatible	110 to 330 Vdc
Output Power: 5.0 kW 32.5 Vdc	6.5 kW 34.0 Vdc
Efficiency: - Overall 90% - Converter 92% - Peak-Power Tracker 98%	 91+% 92+% 99%

Output Paralleling

Shuttle-EPDC Compatible

The transistor bridge power converter stage is fully transformer driven with proportioned base drive. Current sensing is also transformer-coupled through current-sense transformers situated in the return-level emitter circuits. The secondary uses dual parallel rectifier-filters and the switching frequency is 20 kHz.

The principle feature of the TCC analog control circuitry is the active control of transformer flux balance through converter phase current sensing. The pulses of power-transformer primary current are sensed magnetically for each conduction phase.

Regulation breakup at very low output voltages in current limit mode, due to finite pulsewidth limitations, is reduced through foldback current limiting derived from the output voltage as shown. The TCC output I-V characteristic is shown in Figure 3.4.4-2.

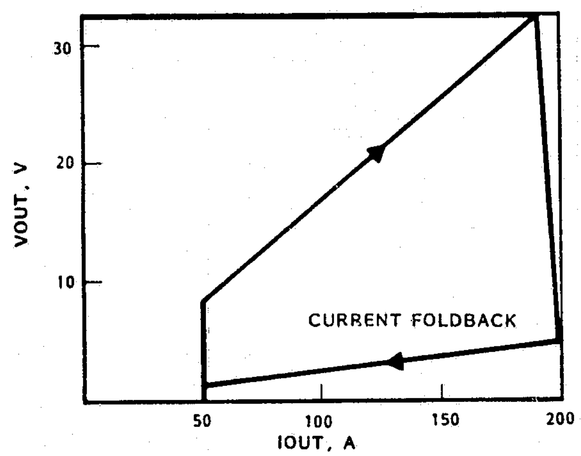


Figure 3.4.4-2
TCC V-I Output Characteristics

The digital-control subsystem handles common logic functions such as pulse phasing and enforcing a minimum offtime. This subsystem also coordinates phase turnon, current-sampling commands, normal phase turnoff, instantaneous phase turnoff, and limiting each phase to a single turnon event per clock cycle.

The peak-power tracker maintains maximum solar-array output power during system overload conditions. The peak-power tracker used is an analog type based on the principle of steepest descent with gradient estimation by means of input-voltage perturbation.

State-of-the-Art - LMSC has build two complete TCC units and operated them at full power (Ref 31). The prototype unit is scheduled for delivery to NASA Johnson Space Center for evaluation in their Shuttle Orbiter power system simulator. The prototype is intended to simulate the overall physical characteristics of a flight unit.

Flight History - None

Types/Manufacturer - LMSC

3.4.5 Partial/Full Shunt Regulator

Description - Shunt regulators are used to limit solar array and/or bus voltage at some value under varying spacecraft bus loading and array power conditions. This is accomplished by applying one or more proportionally controlled shunt elements across the bus as in the case of the full shunt regulator (Fig. 3.4.5-1A). Partial shunt regulators connect at an intermediate point on the array string to reduce power dissipation (Fig. 3.4.5-1B). Other types of shunt regulation schemes are shown in Figure 3.4.5-2 (Ref 30).

Principal Operating Characteristics - The partial shunt regulation approach is more relevant to high-power systems due to its lower dissipation. The binary-segmented, partial-shunt regulator, for example, uses both linear and digital control (Ref 14, 15, 28, 29). One of the unique features of this type of system is that the solar array is divided into binary segments that the shunt regulator controls. All shunt-regulator power stages are either open or saturated except for the first one. Each of the on-off power stages is driven by one of the up-down counter outputs. As a result, the bus current will decrease as the counter decreases. This type of control can be used with equal segmented arrays as well.

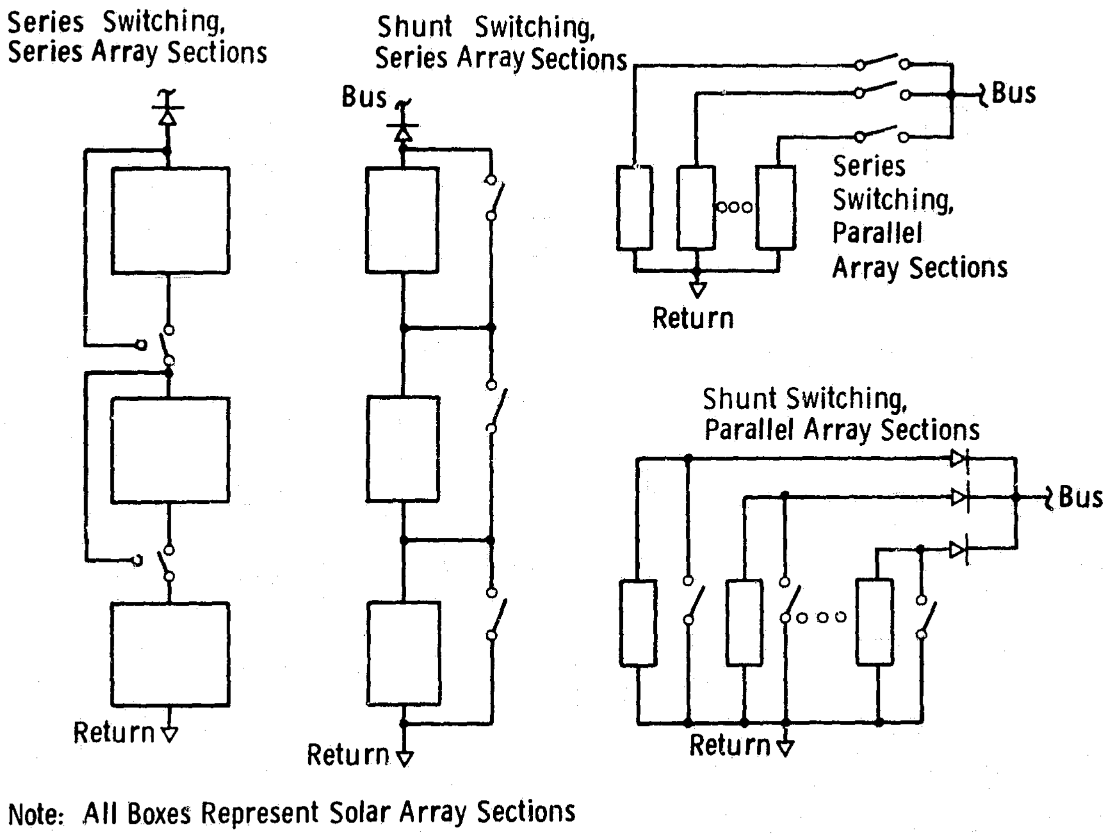
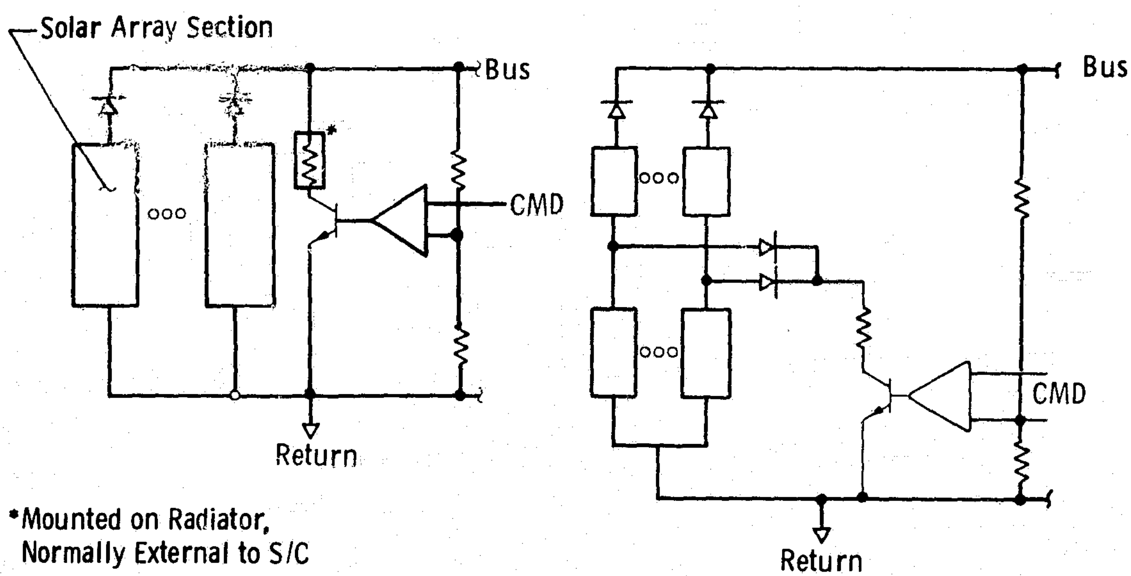


Figure 3.4.5-2 Array Voltage Regulation via Switching



(a) Full Shunt, (b) Linear Partial Shunt

Figure 3.4.5-1 Shunt Regulation Configuration

State-of-the-Art - To date, the shunt approach has been almost exclusively used for GEO and medium-altitude orbits and in low- to moderate- (100-to-2kW) power systems. The shunt regulator can be expanded so it can handle higher power levels by switching from a single-stage system to a multistage system, although growth capability is limited by circuit complexity and component limitations.

Flight History - Many spacecraft have used shunt regulators. Some examples are listed below:

Type	Spacecraft
Full Shunt	TACSAT, OJO, Pioneer Venus Orbiter, Multiprobe Bus, GMS, SCATHA
Partial Shunt	SEASAT, MARISAT, Satellite Business Systems, ANIK-C, NTS-2

Types Available - Typically custom-designed.

3.5 POWER DISTRIBUTION

3.5.1 Magnetic Latching Relay

Description - Magnetic latch relays are electromechanical power-switching components. They have two coils (A and B in Fig. 3.5.1-1 and 3.5.1-2), one for set and one for reset. They require only pulse power to transfer and do not require any steady-state coil power. All space-qualified units are in a nominal 28-Vdc contact rating.

Principal Operating Characteristics - Energizing Coil B produces a magnetic field opposing the holding flux of the permanent magnet in Circuit B. As this net holding force decreases, the attractive force in the air gap of Circuit A, which also results from the flux of the permanent magnet, becomes great enough to break the armature free of Core B, and snaps it into a closed position against Core A. The armature then remains in this position on removal of energy from Coil B, but

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will snap back to position B on energizing Coil A. Because operation depends on cancellation of a magnetic field, it is necessary to apply the correct polarity to the relay coil as indicated on the relay schematic (Fig. 3.5.1-2).

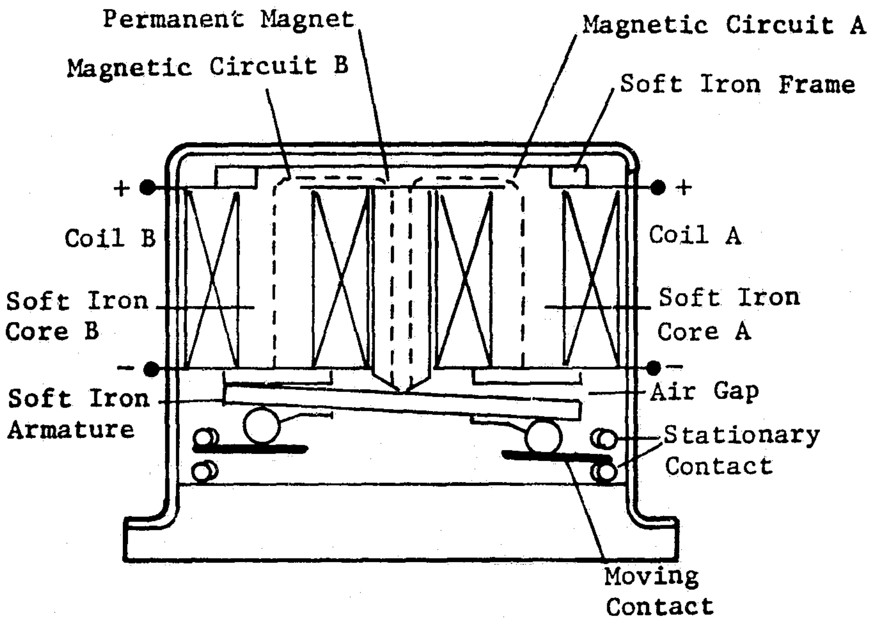

Figure 3.5.1-1 Cross Section of a Mag-Latch Relay

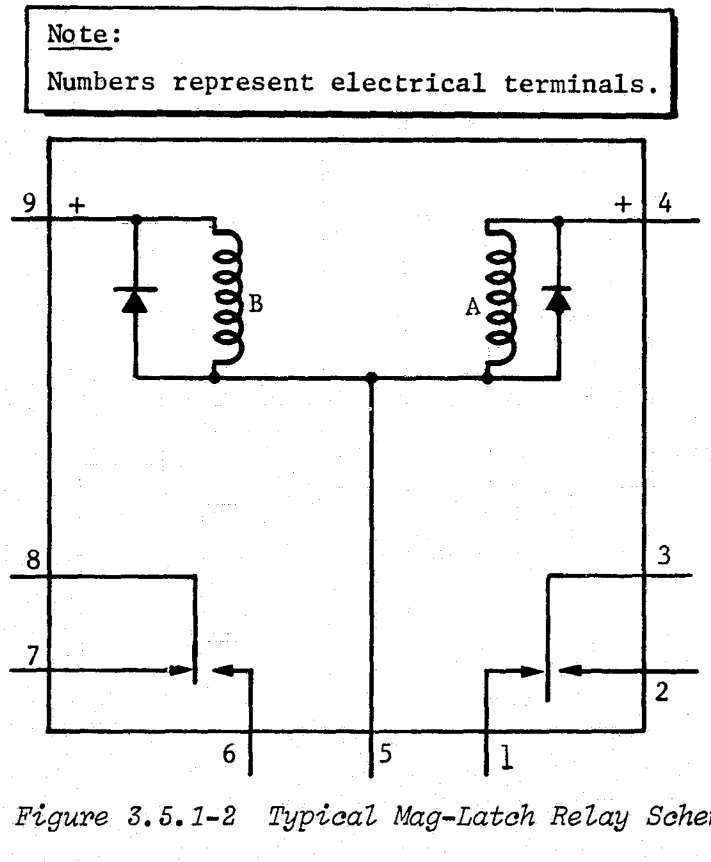

Figure 3.5.1-2 Typical Mag-Latch Relay Schematic

State of the Art - These are mature components with many space-qualified units for 28-Vdc systems. Development is required for 120-Vdc and 240-Vdc systems.

Flight History - These devices have flown on many spacecraft.

Types/Manufacturer - The following types are available for space applications:

Mfg	P/N	Contact Vdc	Adc	Weight, gm	Size, in.
Hartman		28	50	224	1.8x1.99x1.51
LEACH	ICCL Series	28	25	85	1x1x1
LEACH	JA Series	28	10	40	1x1x0.5
LEACH	X Series	28	5	15	0.4x0.8x0.65

3.5.2 Motor-Driven Switch

Description - These components employ a dc motor to make and break the contacts. Contacts are usually DPDT although the user can specify the form of the contacts.. Motor drive is normally 28 Vdc. Internal limit switch stops the motor after opening or closing the contacts.

Principal Operating Characteristics - Table 3.5.2-1 summarizes the electrical performance of a typical motor-driven switch.

Table 3.5.2-1 Motor-Driven Switch Electrical Performance

Parameter	Requirement
Contact Drop:	Less Than 100 mV
Dielectric Strength:	1000 V_{RMS} for 1 min, w/o Failure
Operate Time:	100 ms
Motor Current:	8 to 11 A, 32 V
Contact Rating:	28 Vdc, 200 A Continuous
Overload:	750 A (Make and Brake)
Rupture:	2000 A
Life:	2500 cycles at 28 Vdc, 200 A

State of the Art - Space-qualified components have been used on missiles and spacecraft for years.

Flight History - Flown on most missiles and many spacecraft.

Types/Manufacturer - Kinetics Corp., 10-, 20-, 50-, 100-, and 200-A ratings.

3.5.3 Solid-State Switch--RPC

Description - Solid-state remote power controllers (RPC) are switching devices that combine in one unit the capability to perform all the functions of load switching, overload protection, and direct indication of load status.

RPCs are designed to be located near the load and communicate control and status information remotely via low-level signals. Figure 3.5.3-1 is a functional block diagram of RPC in a typical application. The packages range from 3.8x3.8x2.3 cm, weighing 77 g, to 4.8x4.8x3.1 cm, weighing 142 g for the 28-Vdc version.

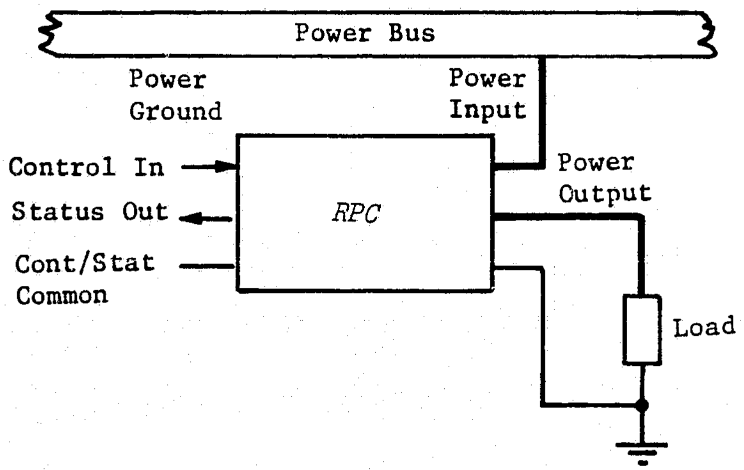

Figure 3.5.3-1 RPC in a Typical Application

Principal Operating Characteristics - Operation of an RPC is relatively straightforward. Bus voltage must exist at the power input to which the positive control voltage is applied. The control section is optically coupled to the logic and internal power supply. With the trip-and-latch circuit armed, the switch-driver circuit is activated to turn on the main power switch and energize the load in a controlled manner (Fig. 3.5.3-2). Once the RPC is activated, it sends back an "on" signal for status indication. In the event of a fault condition, the RPC will either limit, integrate, or trip, depending on the nature of the overload. A trip will result in de-energizing of the load and a trip indication on the status line. Table 3.5.3-1 lists operating parameters for the 28-V version.

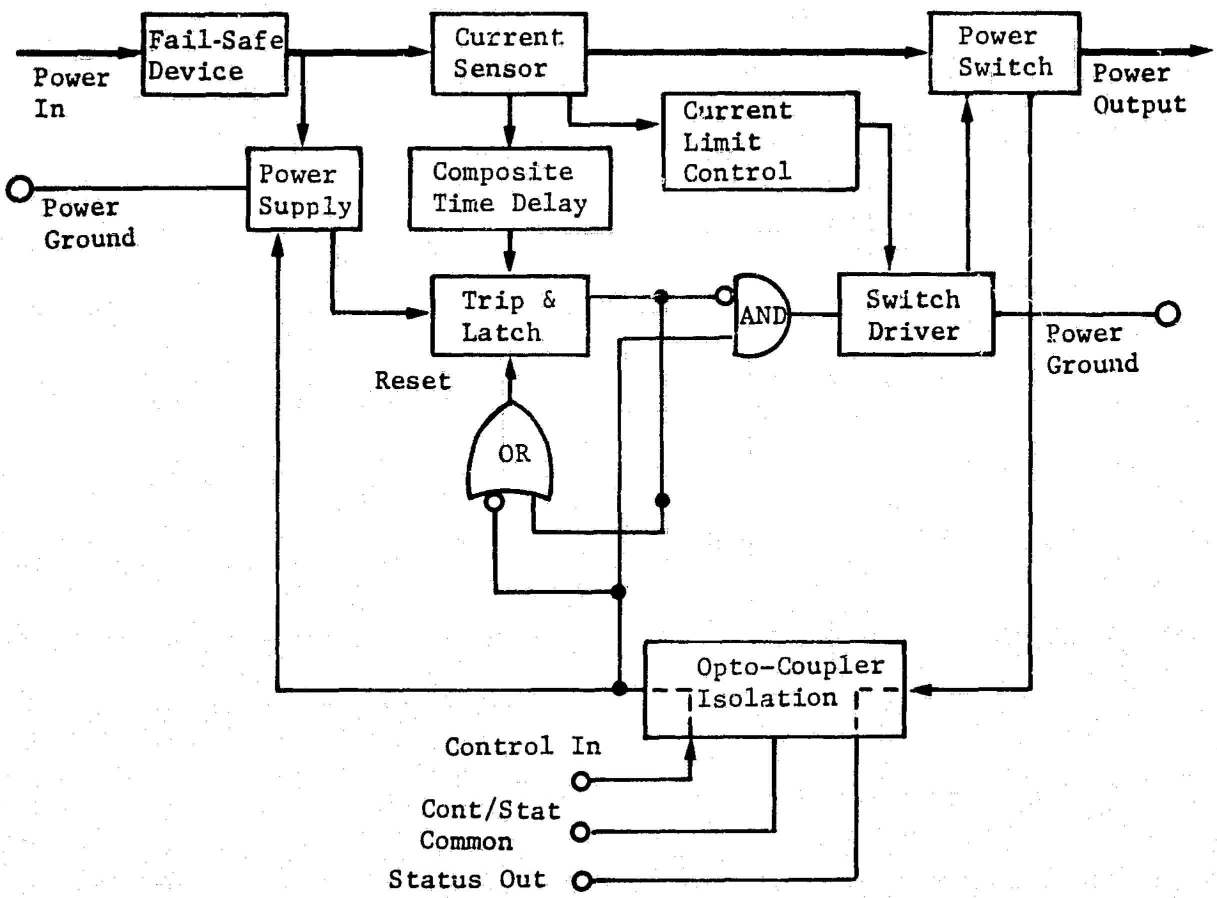

Figure 3.5.3-2
Functional Block Diagram of an RPC Showing Each Basic Function

Table 3.5.3-1 Operating Parameters

Operating Voltage:	24 to 34 Vdc
Current Ratings:	3 A, 5 A, 7 A, 10 A, 15 A, 20 A
Current Limiting:	125 to 150% of rated
Overload-Trip Time:	2 to 3 s
Rise-and-Fall Time:	0.3 to 6 ms
Control Voltage:	5 to 7 V (Off), 9 to 12 V (On)
Control Current:	10 mA max

State of the Art - Space-qualified units are available (see Fig. 3.5.3-3 for typical packaged RPCs).

Flight History - Each Space Shuttle Orbiter contains more than 500 RPCs in six ratings from 3 to 20 A.

Types/Manufacturer - Typical ratings and types available from Westinghouse are:

28 Vdc, 3 to 20 A

120 Vdc, 5 to 300 A

270/300 Vdc, 4 A, 2 A

230 Vac/400 Hz, 1.5 A

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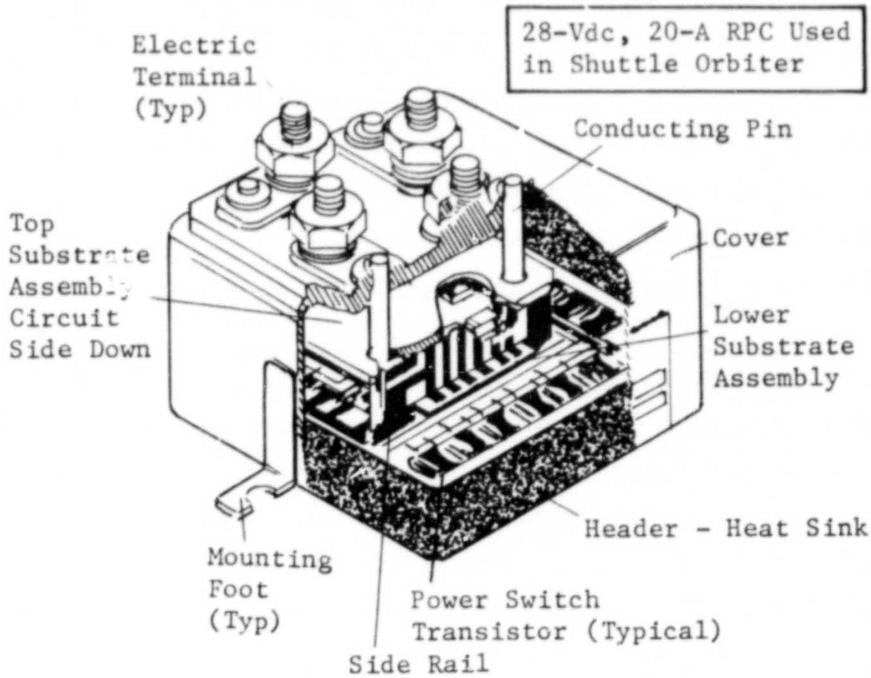

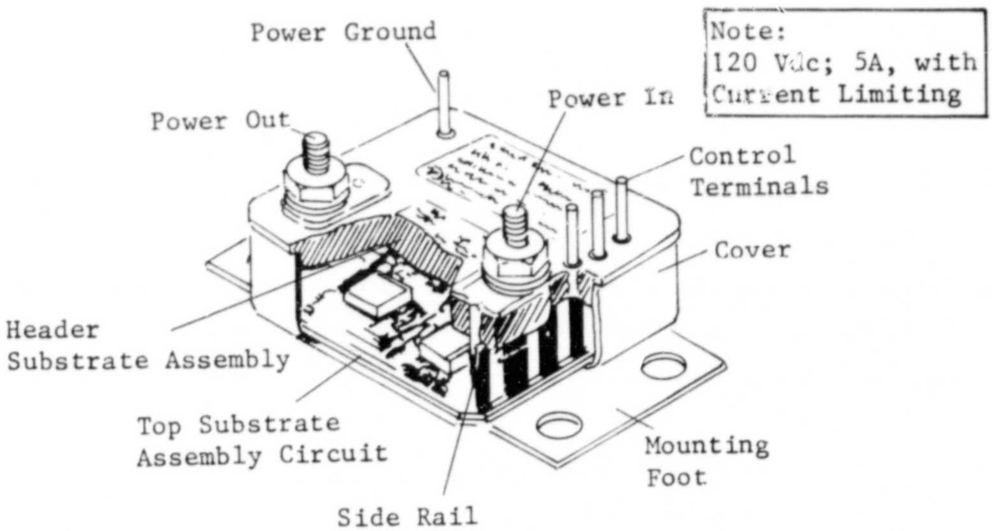

Figure 3.5.3-3
Cutaway View of Packaged Remote Power Controllers

3.5.4 Fuses

Description - A fuse is a device used to protect electrical-system components from fault currents. Two conditions exist where a fuse will open. The first is an overload current, where the current rating is exceeded by any marginal percentage. The second is in the event of a direct short circuit, in which the fault current, (in the absence of a protection device), would exceed the rated current by many orders of magnitude. The possibility exists that a component such as a circuit breaker can be completely destroyed under short-circuit conditions while the fuse opens and protects the user from the fault current. The current-limiting capability of the fuse should allow components with low short-circuit tolerances to be specified.

Principal Operating Characteristics - Fuses are characterized by their rated current voltage and "let-thru" current values (Ref 33). Current rating is a nominal value expressed in amps to which the fuse can be loaded based on a controlled set of test conditions. Voltage rating indicates the value at which the fuse can safely interrupt a fault current. Peak let-thru current is the current value that flows at the time the fuse blows (Fig. 3.5.4-1).

The area under the curve indicates the amount of short-circuit energy being dissipated in the circuit.

Magnetic forces and thermal energy are directly proportional to the square of the current. This implies that the fault current must be limited to as small a value as possible in as short a time as possible. Figure 3.5.4-2 shows a typical relation of blow time versus fault current in percent of rated current.

State-of-the-Art - Fuses are a mature technology.

Flight History - These devices have flown on several spacecraft.

Types Available - A large number of different types exist from several suppliers.

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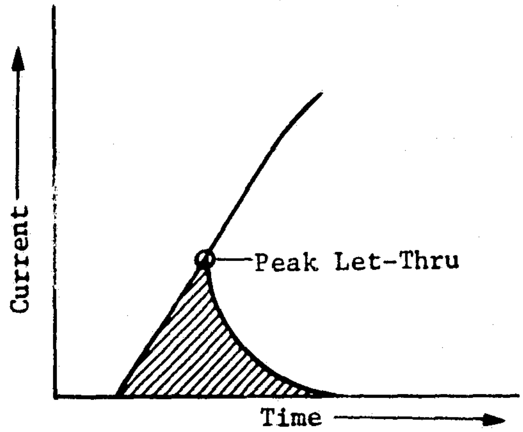

Figure 3.5.4-1
Typical Current-Limiting Characteristics of Fuses

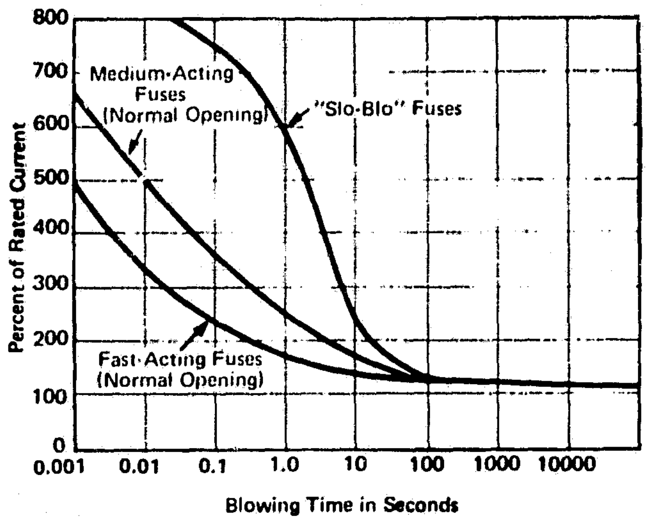

Figure 3.5.4-2
Typical Fuse Blow Time Characteristics

3.5.5 Circuit Breaker

Description - Circuit breakers, like fuses, are a protection device and function to protect the power wiring. The type used on the Space Shuttle Orbiter are thermal circuit breakers. This type of breaker is dependent on temperature rise in the sensing element for actuation. Temperature rise in the sensing element is caused from load-current I^2R heating. This causes deflection of the element (e.g., bimetal), which will cause the circuit to open. The size of the thermal element, its configuration, physical shape, and electric resistivity, determine the current capacity of the breaker.

Principal Operating Characteristics - The Series-4310 ambient temperature-compensated miniature circuit breaker is a lightweight single-phase breaker. This device is designed to operate under severe environmental conditions. Table 3.5.4-1 lists operational data.

C-2

Table 3.5.4-1 Typical Circuit-Breaker Characteristics

Minimum Limit of Ultimate Trip:	No trip within 1 h at 110% load, 25°C.
Maximum Limit of Ultimate Trip:	Trip within 1 h at 145% load, 25°C.
Overload Cycling:	Minimum of 100 cycles at 200% rated current.
Interrupting Capacity:	1 to 20-A models: 6000 A at 28 Vdc.
Dielectric Strength:	1250 Vac
Insulation Resistance:	100 megohm at 500 Vdc.
Weight:	25 g.

The breaker characterized above was built to Rockwell specifications for use in the Space Shuttle orbiter. Other types were used as well.

State-of-the-Art - Space-qualified units are available.

Flight History - Circuit breakers have been used on manned missions (Skylab and Space Shuttle Orbiter).

Types/Manufacturer - Many types are available; for example, see Mechanical Products, series 4310 and Series 4330, used on Shuttle Orbiter.

3.5.6 Cabling

Description - Cables are insulated conductors used to transmit electrical energy to all the various subsystem components. The most common material used is copper because of its high electrical conductivity, ductility, and resistance to wear and fatigue. Copper-alloy conductors are desirable because they permit significant size and weight reduction. Aluminum conductors could represent a great weight savings (50%); however, they have low tensile strength, poor flexibility, and crimp poorly to terminals.

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There are many types of insulation available that are suitable for aerospace applications. The best of these and their properties are shown in Table 3.5.6-1.

Table 3.5.6-1 Characteristics of Various Insulation Materials

	Polyvinyl Fluoride Kynar	FEP Fluoro-plastic	Polyimide Kapton	Teflon	Polyimide Nylon 6
Tensile Strength, psi	7000-18,000	2500-3000	25,000	3000	9000-18,000
Elongation, %	115-250	300	70	250-330	250-500
Burst Strength, Mullen Points, 1-mil Thick	19-70	11	75	11	Elongates
Tearing Strength, lb/in.	997-1400	600	232 mil	600	1000-1200
Water Absorption, 24 h, %-Wt Gained	0.5	0.01	2.9	Neg	9.5
Temperature Limits, °F					
- High	220-250	440-525	750	392	200-400
- Low	-100	-425	-450	-112	-100
Dielectric Constant at 10^3 Hz	8.5	2.0-2.05	3.5	2.1	3.7
Dielectric Constant at 10^9 Hz	1.6	2.05	3.4	2.05	3.4
Dielectric Strength, V/mil	7000	3500	7000	7000	1300-1500

Thermal derating is based on the wire-bundle configuration. The derating factor considers the temperature rise due to reduced thermal view and thermal conductivity of the bundle. For example, flat conductor cable requires the least derating, owing to a greater surface area not common to the other conductors (Fig. 3.5.6-1).

A cylindrically assembled bundle requires more derating to keep operating temperatures low.

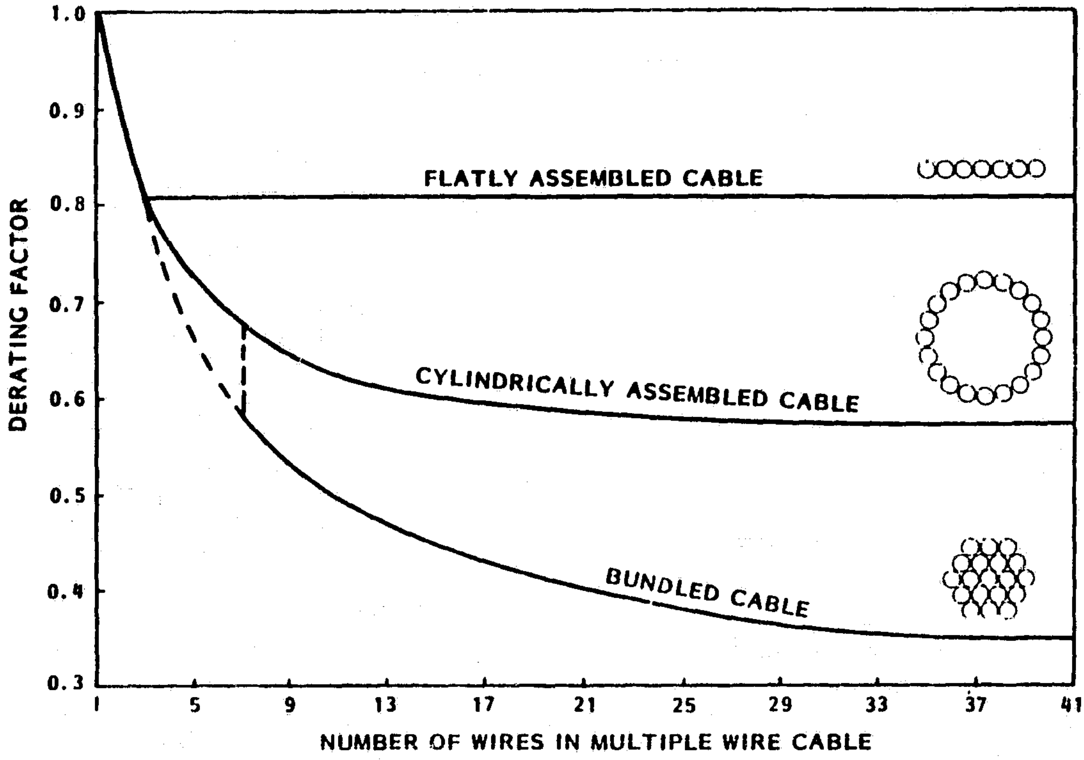

Figure 3.5.6-1 Derating Curves for Multiple Cable Assemblies

Principal Operating Characteristics - Power conductor parameters are listed in Table 3.5.6-2 for different materials. Table 3.5.6-3 shows performance information for these types of materials.

State-of-the-Art - Copperclad aluminum cables and bus bars are presently used in space programs. Sodium and intercalated carbon fibers represent new technology (Level 3). Sodium conductors would be extremely lightweight, and intercalated carbon would reduce cost as well as lower the weight.

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Table 3.5.6-2 Power Conductor Characteristics

Parameter	Copper*	Aluminum+	Sodium#	Intercalated Carbon Fibers#
Relative Conductivity, %	100 (Ref)	61	40	TBD
Volume Resistivity, ohm-cm	1.72×10^{-6}	2.82×10^{-6}	4.3×10^{-6}	3.5×10^{-6} To Date
Density, g/cc	8.89	2.70	0.97	2.7
Temp Coefficient of Resistance	-0.00393	-0.00410	-0.0044	TBD
Coefficient of Linear Expansion/°C	17×10^{-6}	23×10^{-6}	62×10^{-6}	1×10^{-6}
Melting Point, °C	1083	659	97.5	N/A
Electrical Resistivity	15.3×10^{-6}	7.61×10^{-6}	4.17×10^{-6}	9.5×10^{-5}
Relative Density to Conductivity Ratio %	100	50	27	618 To Date

*Present
+Near-Term
#Far-Term

Flight History -

Copper Types Used Extensively

Sodium - None

Intercalated Carbon - None

Types Available - Copper

Table 3.5.6-3 Power-Conductor Performatnce Information

Parameter	Copper	CdCrCu	Aluminum	Sodium	Intercalated Carbon
Tensile Strength (Also Improved By Insulation), psi	32,000	68,000	15,000	N/A	300-1000
Flexibility	Reference	3X Copper	1/3 Copper	Depends On Sheath	TBD
Crimp Terminability	Excellent. Crimping Tools Designed Around Copper	Very Good. Greater Crimping Force Required	Poor. Tends to Creep, Causing Looseness and Arcing	N/A	N/A
Solderability	Excellent. Mild Flux Usually Required	Very Good. Stronger Flux Required with Alloys	Very Poor. Special Flux Required	N/A	N/A
Stability	Fair. Prone to Oxidation and Chloride and Sulfide Tarnish	Same as Copper, Except Alloying Decreases Rate of Attack	Excellent, Except in Chloride Environment	Good Only in Space Environment	Excellent.

3.6 POWER-TRANSFER DEVICES (GIMBALS)

3.6.1 Slip Rings

Description - Slip rings are used to transfer electrical power and signals from the solar-array and sun-sensor preamps to a stationary portion of the structure. Under NASA contract NAS3-22266 on power management technology, Poly-Scientific Corp. evaluated the feasibility of producing a slip-ring capsule assembly (Ref 35). This module design serves as a good example of present slip-ring technology.

The slip-ring capsule was designed in 25-kW sections to be combined into a 100-kW capsule. Table 3.6.1-1 lists physical/mechanical characteristics.

Table 3.6.1-1 Physical and Mechanical Characteristics

Length:	11 in.
Outside diameter:	5.5 in.
Weight:	13 lb
Rings, Number:	8 Total, 4 +, 4 -
Material:	Coin Silver, (9 Ag-10 Cu) or Hard Silver Electrodeposit
Brushes, Number:	6 per Ring
Material:	Silver, Molydisulfide, and Graphite
Life:	5 Years
Current Density:	62.5 A/in.^{2}, Normal; 150 A/in.^{2}, Emergency
Drive Torque:	8 in.-lb

Principal Operating Characteristics - The slip-ring capsule assembly may be used to reliably and efficiently transfer 100-kW of power in space. Table 3.6.1-2 summarizes the electrical operating parameters.

Table 3.6.1-2 Electrical Parameters

Voltage:	400 Vdc
Current:	62.5 A per Module; 250 A Total
Power:	100-kW, 4 to 25-kW Modules
Contract Drop:	0.090 V
Power Loss:	45 W

State-of-the-Art - Slip rings are a mature technology (Level 8) and are applicable for 100-kW range.

Flight History - See Table 3.6.1-3.

Types/Manufacturer - See Table 3.6.1-3.

Table 3.6.1-3
Flight History Space Slip Rings and Poly-Twists Preliminary Poly-Scientific Data

P/N	Application	Customer	Type
FK1806,7	Nimbus SA	TRW	Sep
D1836	Tiros	BBRC	Cap
BQ1946	Not Defined	Cap	Comp
ET2010	OSD	BBRC	Cap.
EW2063	Apollo Ant.	Dalmo Victor	Cap.
F12076	INT IV A	HAC	Sep
BN2098	Mars Probe	GE	Cap.
ET2189	Scoop	BBRC	Cap.
D2255	Skylab	Bendix	Cap.
FK2334	Viking	TRW	SW
ET2374	Atm Exp	BBRC	Cap.
FL2391	OSO	HAC	Cap.
AS2431	Dom Sat.	RCA	Sep
ET2445	CTS	BBRC	Cap.
FK2450	FLT SAT. COM	BBRC	Sep
FK2470	Solar Array	TRW	Cap.
DQ2614	Not Defined	LMSD	SW
DQ2615	Solar Array	LMSD	Cap.
D2634	ELMS	Bendix	Cap.
AS2646	TEL SAT.	RCA	Cap.
JP2650	OTS	HSD	Sep
AC2737	Not Defined	-	Cap.
DQ2769	Sea Sat.	LMSD	Cap.
ET2793	P78-2	BBRC	Cap.
KU2832	INT V	FACC	Sep
FK2857	TDRSS	TRW	Sep
FL2907	SBS ANIK-C,D	Hughes	Sep

Legend:

Cap. - Capsule SW - Switch
P - Pancake Tape - No Contacts, Tape Conducts

3.6.2 Roll Ring

Description - The roll ring is a device for transferring power across a rotary joint. This approach incorporates a complex structure of mechanical parts (Fig. 3.6.2-1), which significantly reduces friction. The dimensions of a developed device are 25 in. long, 10-in. diameter, and it weighs 30-kg.

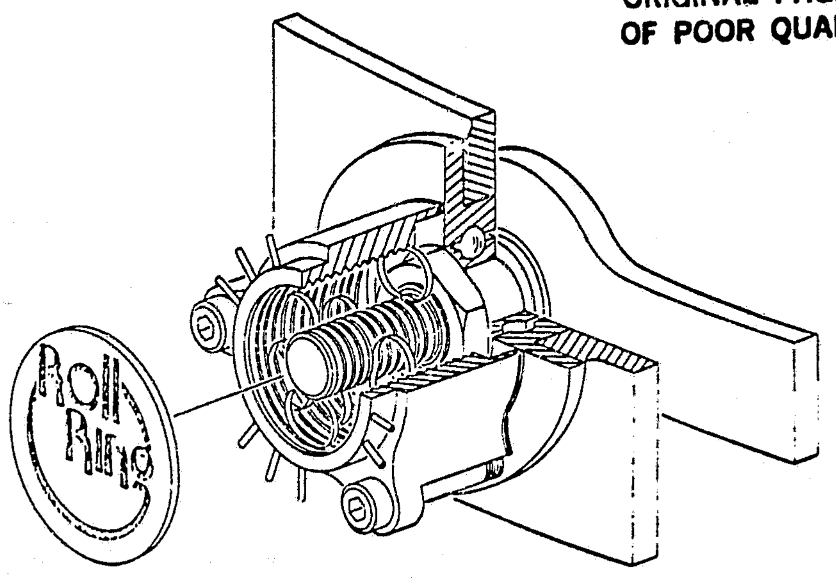

Figure 3.6.2-1 Cutaway of 11-Contact Roll-Ring Capsule (Ref 34)

Principal Operating Characteristics - Table 3.6.2-1 lists roll-ring performance characteristics identified in Reference 34. Its design goals are:

1) Provide transfer of power ranging from 10 kW to 100 kW;

2) Be capable of handling high voltage independent of the environmental pressure; a 1000-V criteria was used to force a solution of the high-voltage corona problem for high-power systems in vacuum, with potential operation pressures in the critical-pressure zone;

3) Transfer power with a minimum size and weight;

4) Meet long-life operating requirements ranging from three to 10 years, with rotation up to 56,000 revolutions;

5) Provide redundancy in the power-transfer lines;

6) Ensure wearout-failure modes are open-circuit type;

7) Transfer power with unlimited angular rotation.

Table 3.6.2-1 Roll-Ring Performance Characteristics

Parameter	Capability
Rotation Limit	Unlimited
Internal Pressure, mm Hg	780×10^{-8}
Voltage Limit	200
Max Current, A	10
Corona Problem	Yes
Life	Millions of Rev
Conductor Size and Number	Fixed 10 (Bearing Friction)
Angular Rotation	X1

State-of-the-Art - This is a new technology device (Level 4).

Flight History - None

Types/Manufacturer - None

3.6.3 Rotary Transformer

Description - A rotary transformer designed by GE (Ref 35) consists of a primary core with windings and a secondary core with windings in a cylindrical configuration. The secondary core encloses the primary core, which has a shaft through the center. The secondary can be supported by a housing that is connected to the spacecraft structure. The primary core/shaft assembly can rotate freely within the secondary core. This configuration allows energy to be transferred through a rotary joint by magnetic induction once power conditioning electronics are connected to the rotary transformer. The transformer characteristics are listed in Table 3.6.3-1.

This device is being developed for use with a series resonant converter. The power per module is based on a 25-kW design, however there are no inherent limitations to the power levels.

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Table 3.6.3-1 Rotary Transformer Characteristics (Ref 35)

Core	
Outside Diameter	9.0 in.
Air-Gap Diameter	5.35 in.
Inside Diameter	2.0 in.
Air-Gap Length	0.01 in.
Width of pole	0.6 in.
Winding	
Primary Resistance, 100°C	0.0053 ohms, dc 0.0136 ohms, ac
Secondary Resistance, 100°C	0.029 ohms, dc 0.120 ohms, ac
Primary Inductance	19 H
Secondary Inductance	51 H
Weight, lb	
Copper	7.1
Core	15.7
Losses	
I^2R	141
Core	89
Efficiency	99%
Thermal	
Primary	
Sink Temperature	60°C
Core Temperature	100°C
Coil Temperature	105°C
Secondary	
Sink Temperature	60°C
Core Temperature	63°C
Coil Temperature	66°C

Principal Operating Characteristics - Four 25-kW modules combine to provide 100-kW capability. A drive module provides a rotational capability from one revolution per day to one revolution every 90 minutes using a stepper motor, speed reducer, and clutch. Table 3.6.3-2 lists the basic operating characteristics of the system.

State-of-the-Art - This is a new technology item (Level 3).

Flight History - None

Types/Manufacturer - None

Table 3.6.3-2 Operating Characteristics of 100-kW Rotary Transformer

Input from Solar Array	
Power	100 kW
Voltage	440 V
Output from Rotary Power Transfer Device	
Voltage	1000 V
Frequency	20 kHz
Power Conditioning Electronics	
Resonant Circuit (Schwartz)	
Rotary Transformer	
Power	100 kW
Input Voltage	400 V
Input Current	70 A
Output Voltage	1000 V
Frequency	20 kHz
Inductance	75 H
Configuration	Concentric Cylinder 4- to 25-kW Modules Two Parallel Secondary Windings per Module
Rotational Period	90 minutes to 24 hours
Efficiency	Greater than 95%
Environment	Shuttle Launch
Temperature	
- Nonoperating	-20° to 80°C
- Operating	80° Heat Sink, Rotary Transformer 60° Heat Sink, Power Conditioning Electronics
Life	5 years

3.6.4 Flex Cable

Description - A simple approach to rotational power transfer, is the Lockheed designed and developed twist flex unit (Ref 34). This technique permits power transfer through insulated wire bundles from one rotating disk to a second rotating disk. The disks are mounted on a shaft (torque tube) that connects to a bulkhead.

The wire bundle is made up of 40 pairs of 16-gauge wire, 72 pairs of 24-gauge wire, and eight twinax. The unit is 13-in. diameter, 25-in. long, and weighs 10 kg.

Principal Operating Characteristics - Table 3.6.4-1 is a list of the primary characteristics.

Table 3.6.4-1 Twist Flex Characteristics

Parameter	Capability
Rotation Limit	$\pm$205 Deg
Internal Pressure, mm Hg	780 p 10^{-8}
Voltage Limit	400 V
Max Current, A	15 A
Corona Problem	None
Life	0.4 x 10^6 Rev Demonstrated
Particle Generation	None
Major Failure Mode	Open
Conductor Size	Simple to Revise
Angular Rotation	X2

State of the Art - The design has been fully developed.

Flight History - None.

Types/Manufacturer - LMSC.

3.7 SENSORS AND SIGNAL CONDITIONING

3.7.1 Ac Voltage and Current Sensors

Description - Ac voltage and current sensors are devices (usually magnetic) that provide a calibrated analog signal acceptable to conditioning or control electronics.

Principal Operating Characteristics - A common method of sensing alternating current involves a current transformer. The conductor carrying the current to be measured is taken to be the primary winding. The voltage developed on the secondary is proportional to the primary current. Figure 3.7.1-1 shows a typical current-transformer approach (Ref 7).

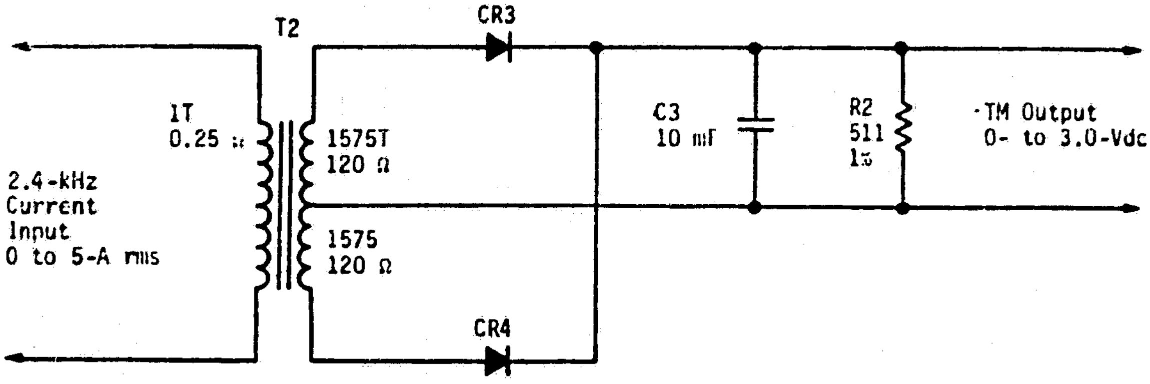

Figure 3.7.1-1 Current Tranformer Approach

True RMS current can be detected using the circuit shown in Figure 3.7.1-2. In this case, current is sensed with a shunt, another common sensing element. A 3-V p-p signal input to the true RMS converter produces a 3-Vdc output signal.

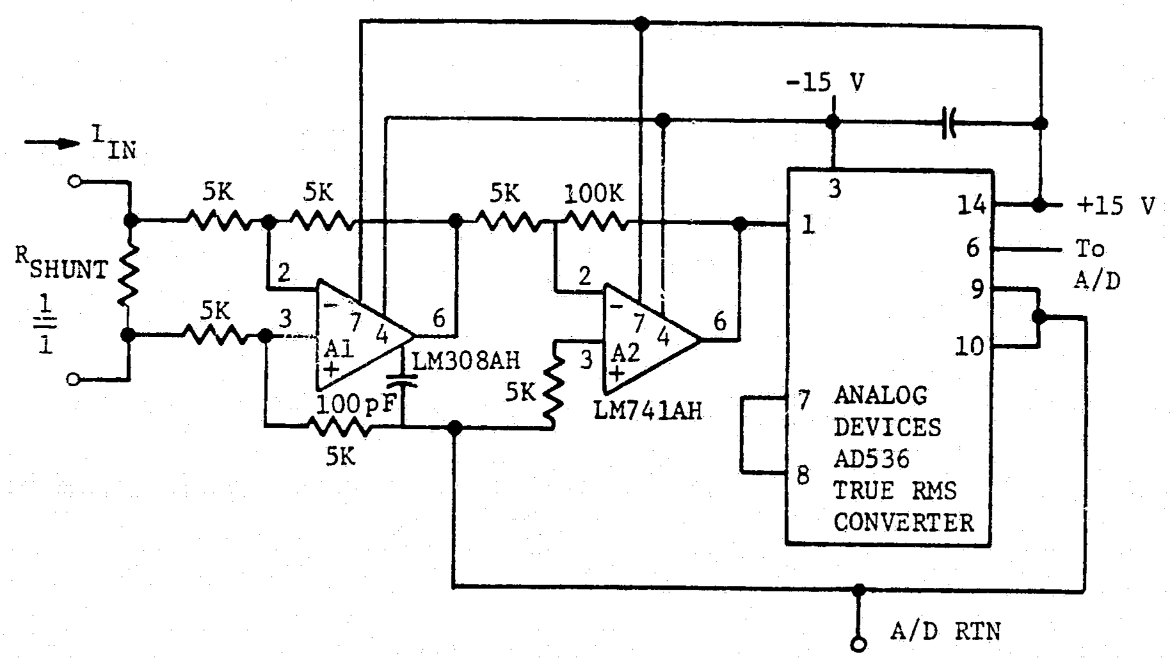

Figure 3.7.1-2 Shunt and True rms Converter

Standard operational amplifiers scale the shunt signal to the appropriate values. Ac voltage monitoring is also sensed magnetically. A transformer easily scales the voltage down to a small signal that can be rectified and filtered.

State of the Art - These devices have been fully developed (Level 8).

Flight History -

Types/Manufacturer - These devices are custom-made items.

3.7.2 Dc Voltage and Current Sensors

Description - These devices provide a calibrated analog signal to the conditioning system. Voltage measurement usually involves a resistor divider and an op amp. Dc-voltage measurement is somewhat simpler than ac, whereas the opposite is true for dc-current measurement. A current can easily be transduced with a shunt; however, this method is only practical at the lower levels. Mag-amps are used for nonintrusive sensing of high currents and are more complicated.

Principal Operating Characteristics - A dc voltage sensor can be made simple and reliable. Figure 3.7.2-1 is a schematic of a typical voltage transducer. The variable divider is R1 and R2. Amplifier A1 is used as a difference amplifier; that is, it rejects common-mode voltages when R1 and R2 are at the source. A2 is a unity-gain inverting amplifier. For positive input voltages, the output is taken from the output of A2. For negative input voltage, the output is taken from the output of A1. The output impedance of this transducer is low because, for both positive and negative source voltages, the output is an operational amplifier with a gain of -1. Table 3.7.2-1 shows the principal features of a dc voltage transducer.

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Note: 1. A1 and A2 LM741-CN
2. All R 1%, 1/4 W RN 55

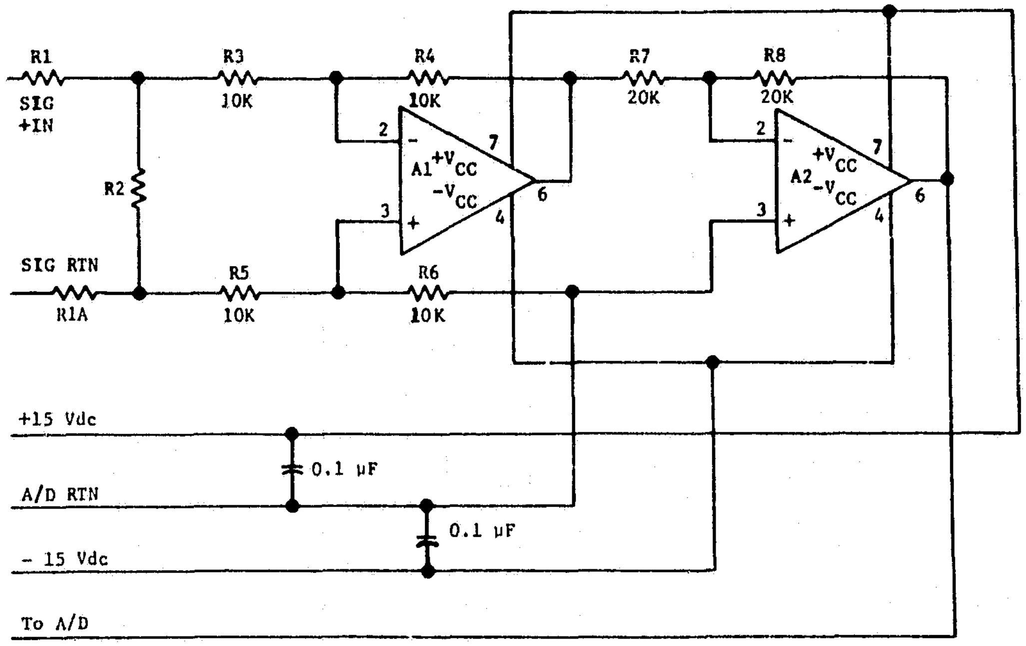

Figure 3.7.2-1 Dc Voltage Transducer, R Divider, and Operational Amplifier

Table 3.7.2-1
Dc Voltage Transducer Design Details

Source Voltage, Full Scale	Source Voltage, Nominal	R1	R2	E(Out) Full Scale
65	56	103 kohm	20 kohm	3 V
40	30	61 kohm	20 kohm	3 V
8	5	8.3 kohm	20 kohm	3 V
+20	15	28.3 kohm	20 kohm	3 V
-20*	-15	28.3 kohm	20 kohm	3 V

*For -20 V, delete R7, R6, R8, and A2. Use E1 as output.

Figure 3.7.2-2 shows the type of mag amp used on the Viking Orbiter '75 (Ref 7). Each toroid core (A & B) has an excitation/reset coil that is connected to the drive circuitry as shown. CR1 and CR2 always steer the current through coil 6,5 in the same direction, while alternately resetting the cores on opposite half cycles. Dc load current passes through the toroids via coil 7,8. The output voltage is determined by the product of the turns ratio times the load current times the resistance of R3.

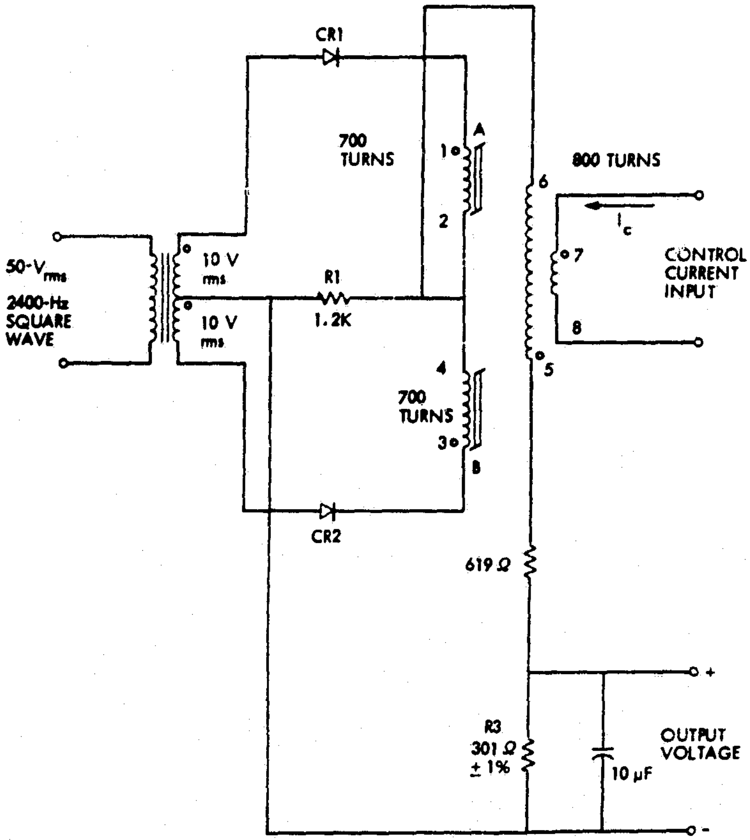

Figure 3.7.2-2
Viking Orbiter '75 Type of Magnetic Amplifier Current Transducer

State of the Art - These devices have been fully developed.

Flight History - N/A.

Types/Manufacturer - These devices are custom-made items.

3.7.3 Temperature Sensors

Description - Materials that change resistance by some function of temperature are normally used as temperature transducers. Typical ones are platinum wire segments, resistors, and copper.

Principal Operating Characteristics - These devices are commonly used in a balanced resistive-bridge configuration. Imbalance due to temperature change can be sensed differentially across the bridge. Figure 3.7.3-1 is a schematic of this type of circuit. It is scaled to produce zero output at 32°F and +3V at 150°F. The thermistor R1, R2, and R3 form a bridge. Amplifier A1, along with R4, R5, R6, and R7, convert the common-mode voltages across the thermistor and R3 into a single-ended voltage. A2 is an adjustable-gain amplifier used to set the scaling in a precise manner. An amplifier with a guaranteed low offset voltage is used for A1 to preclude trimming of offset voltages and to achieve minimum error due to A1 offset voltage.

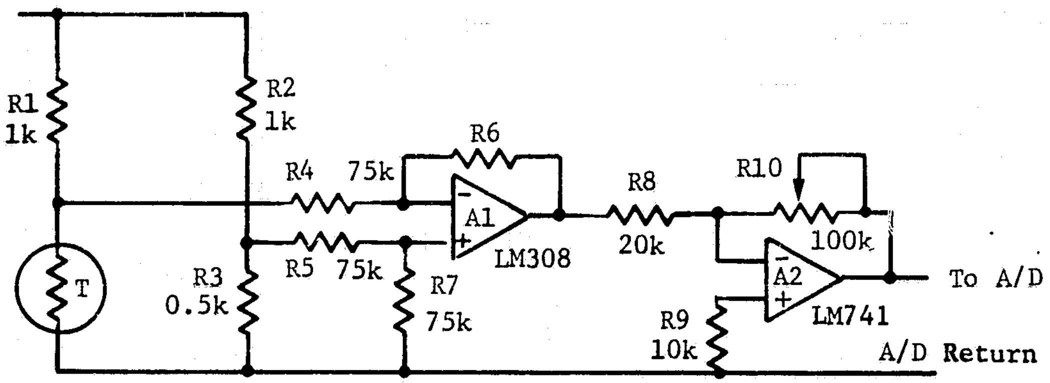

Figure 3.7.3-1 Thermistor-Bridge Temperature Sensor

State of the Art - These devices have been fully developed.

Flight History - Used on all spacecraft.

Types/Manufacturer - All ranges are available for custom design.

3.7.4 Pressure Sensors

Description - Pressure measurements can be accomplished reliably by using a metallic strain gauge (Ref 36). Pressure in a container will induce stress on the solid, constraining material, which can be measured using a strain gauge. Metallic strain gauges are formed from thin resistance wire or are etched from thin sheets of metal foil. Figure 3.7.4-1 shows a bondable wire-grid strain gauge.

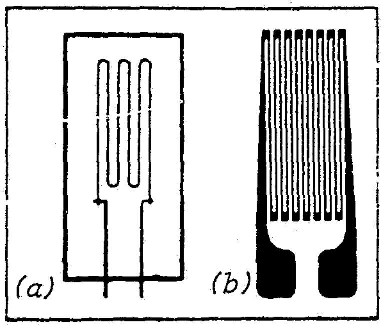

Figure 3.7.4-1
Uniaxial Strain Gauge,
(a) Wire, (b) Foil (Gould Inc, Mesaurement Systems Division)

Many types of material are used to fabricate these devices, such as Constantan, Nichrom V, and Stabiloy. Typical sizes range from 1/8x1/8 in. to 1x1/2 in.

Principal Operating Characteristics - In the usual application, the strain gauge is cemented to the structure whose strain is to be measured. The adhesive material must hold the gauge firmly to the structure, yet it must have sufficient elasticity to give under strain without losing its adhesive properties. The adhesive should also be resistant to temperatures, humidity, and other environmental conditions.

Connecting four gauges in a bridge configuration is the most common method of electrically sensing the changing resistance. Having two gauges active and two gauges inactive provides a balanced, temperature-compensated bridge circuit. Signal amplification and scaling are performed in the usual manner.

State of the Art - These devices have been developed, tested, and used extensively. Present development is directed toward microminiature semiconductor versions.

Flight History - Intelsat 5 and 6 have used strain gauges for NiH_2 battery pressure sensing.

Types/Manufacturer -

- Uniaxial, Wire or Foil

- Two- and Three-Element Rosettes

- Signal Conditioning Custom Design

4.0 TASK 2 - DEFINITION OF FAULTS AND FACTORS AFFECTING EPS PERFORMANCE

OBJECTIVE AND SCOPE

The objective of this task is to (1) develop a comprehensive list of electrical power system (EPS) faults, activities in other subsystems, and other factors that could prevent the power subsystem from functioning properly, and (2) define their operational impact on the EPS.

SUMMARY

Inputs to this task were the components of a generic EPC developed in Task 1. A "fault" is defined to include all types of failures and degradation modes.

A summary of the major EPS failure and degradation modes is shown in Table 4-1. The only EPS failures that could result in catastrophic loss of the spacecraft are explosion of the NiH_2 pressure-vessel and failure of a series-resonant inverter capacitor. Both of these potential failures must be eliminated by design, worst-case analysis, and test, and not by automation. Table 4-2 is a list of operational impacts resulting from failures.

A summary list of other subsystems and activities that could affect the EPS is given in Table 4-3. A summary of unknowns that could affect the EPS is given in Table 4-4. There are two methods for considering failures:

1) Undetected and uncorrected;

2) Timely detection and correction.

Table 4-1 Major EPS Component Failure and Degradation Modes

EPS Component	Major Failure Modes	Degradation Modes
Photovoltaic Array	- Open - Short	- Filter, Antireflective Coating - Arcing - Power Loss Due to Plasma Interaction & Charged-Particle Radiation
Slip Rings	- Short	- Particle Generation from Brushes (Major)
Roll Rings	- Open	- Particle Generation from Rings (Minor)
Twist Flex	- Open	
P3 (Dc/Dc Converter)	- Shorted Series-Pass Transistor - Output Overvoltage	- Efficiency - Ripple
Transformer-Coupled Converter (Dc/Dc Converter)	- Output Overvoltage	- Efficiency - Ripple
Series-Resonant Inverter(Dc/Ac Converter)	- Shorted Semiconductor Power Switch - Shorted Commutating Diode - Output Overvoltage - Input Cap Destruction By Overvoltage	- Efficiency
Photovoltaic Array Voltage Controller	- Loss of All Output from an Array	- Partial Loss of Control & Regulation
Magnetic Latching Relays	- Fail to Operate - Transfer when Not Commanded	- Increased Contact Resistance
Remote Power Controllers	- Fail to Transfer - Spurious Transfer - Oscillation - Fail to Limit Rise & Fall Time of Current - Fail to Limit Fault Current	- Increased Contact Resistance - Loss of Status Indication

Table 4-1 (concl)

EPS Component	Major Failure Modes	Degradation Modes
Fuses	- Opens at Current Less Than Spec - Does not Open at Spec Current	
Cabling	- Open - Short	- Insulation Life Degraded Due to Excessive Temperature or Voltage
Sensors	- No Output	- Accuracy Out of Spec - Out of Calibration
Chemical Turbo Machinery	- Reactant Leakage - Turbine Mechanical Failures - Generator Electrical Failures	
Regenerative Fuel Cell, Electrolysis and Fuel Cell	- H_2 in O_2 Manifold - O_2 in H_2 Manifold - V/I HI/LO - Absolute Pressure HI/LO - Excessive H_2 and O_2 P - Temps Hi/Lo - Voltage Regulator Out of Spec	Separator Electrode
Nickel-Cadmium Battery	- Shorted Cell - Open Cell Due to Cell Reversal	- Loss of Capacity - Low Voltage - Overpressure Failure
Nickel-Hydrogen Battery	- Pressure Vessel Leak Resulting in Open Cell/Cells - Overpressure Failure Due to Overcharge	- Loss of Capacity - Low Voltage
Lithium-Thionyl-Chloride Primary Battery	- Open Cell - Shorted Cell (Which Can Cause Other Failures, Including Overpressure)	- Low Final Voltage - Loss of Capacity

Table 4-2 List of Operational Impacts

- Catastrophic Loss of the Spacecraft - Complete Loss of Mission Functions - Partial Loss or Degradation of Mission Functions - Loss or Degradation of a Subsystem Function - Loss of Fault Management or Maintenance Capability - No Significant Impact
Note: Above definitions are from JPL Report SD-TR-82-58, Autonomous Spacecraft Design and Validation Handbook. April 30, 1983.

Table 4-3 Other Subsystems and Activities That Can Affect the EPS

Subsystem	Failure/Activity	Effect	Operational Impact*
Structures	Modular Buildup	Reduced Power	3,4
Thermal Control	Impaired Capacity to Jettison Waste Heat	Reduced Power	3,4
User Loads (All Subsystems and Payloads)	Shorts or Overloads	Bus Undervoltage	3,4
	Large Differences in Day and Night Power at Buses	May Reduce Bus Power Capability; Excessive Battery DOD	3,4
Attitude Control	Gravity Gradient Attitude Mode	Reduced Power	3,4
EPS/Crew Interface	Crew Commands, Displays, New Crew, Interface Ambiguity, Mistakes	Reduced Power Capability; Unintended Shutdown	3,4
EPS Ground Operations Interface	Power Management Configuration History; Audit Trail or Automated Activities; Training; Commands/Displays	Reduced Power Capability	3,4
Attitude Control	Failure to Maintain Required Stable Attitude Because of Unknowns in Controlling Large, Flexible Structures	Reduced Power Capability	3,4
Command	Degraded TM Data Transmission	Reduced Information	3,4
	Loss of CPU Power	Reduced Automation Capability	3,4
Data	Software Maintenance	Reduced Power Capability	3,4

*See Table 4.4.1-2.

Table 4-4
List of Other Factors That Could Affect EPS Design and Performance

Factors	Primary Effects On:
- Orbital Environment and Parameters: - Charged-Particle Degradation - Thermal Cycling - UV Losses - Solar Flare - Solar Intensity Variation - Plasma Interactions	Solar Array
- Station Orientation	Solar Array
- Station Growth	Array, Batteries, Power Distribution
- Life	Solar Array, Batteries
- Onorbit Maintenance, Rendezvous and Docking	Checkout and Diagnostic Abilities
- Assembly and Buildup	Solar Array, Batteries
- Mission Operations	All Subsystem Elements

The most serious failure is one that is undetected and uncorrected. This could arise from a lack of redundancy, or a double or triple failure. The operational impact of an undetected and uncorrected failure can range from complete loss of mission functions to loss of EPS functions. One object of automation is to provide the resources, monitoring, and control to ensure that all admissible failures are detected and corrected in a timely manner. When there is timely failure detection and correction, the operational impact can be lowered to that of loss of fault-management capability. The possible impacts of the two kinds of failures are summarized as follows:

1) Undetected and uncorrected failure impacts,

   a) Damage to user loads,

   b) Loss of mission capability,

   c) Safety hazards,

   d) Wiring damage,

   e) Schedule, mission operations, and planning,

   f) Possible drive of SS into shutdown, survival mode,

   g) Time required to bring SS back up to operational mode,

   h) Time required for damage assessment,

   i) Time for maintenance, resupply, STS future flights,

2) Timely fault detection and correction impacts,

   a) No damage to user loads,

b) Minimum user-load downtime, loads shifted to backup,

c) Immediate decrease in backup capability,

d) Requirement for maintenance resupply,

e) Possible impact on operations that require more backup capability than exists,

f) Minimized impact on mission by timely fault detection and correction,

The key conclusion drawn from Task 2 is that automation is essential in correcting the problems identified and that automation is an enabling technology.

4.1 PHOTOVOLTAIC ARRAY FAILURE MODES AND OPERATIONAL IMPACT

Failure Modes - A photovoltaic array usually consists of a number of series and parallel strings of solar cells. Each string requires an isolation diode. For articulating solar arrays, power transfer from the array to the power-conditioning equipment may require a slip ring, a roll ring, or a "flex ring."

A catastrophic, single-point failure is the slip ring. A short or open in the slip ring causes a loss of all power from the array served by that slip ring. An open failure of an interconnect wire (or open isolation diode) in a series string causes a loss of that string. This failure results in loss of a fraction of the array power. There are other long-term degradations that result in loss of solar array power, such as slow degradation of the cover glass or lens by micrometeorites, outgassing, or process failure.

Environmental impacts on the solar array are possible arcing and loss of array power owing to parasitic currents set up in the plasma. If there are gimbals and slip rings on the solar array, this implies a tracking servo with commands, electronics, and a stepper motor. There are catastrophic failures, degraded accuracy failure, and failures that result in oscillation of the servo motor, with premature wear-out associated with the elements of the sun-tracking servo system. Attitude control and operational mode can affect the solar array by shadowing the array. Shadowing reduces the output of the array and can lead to solar-cell failures from excessive heating or reverse-voltage breakdown.

Operational Impact - A summary of the solar array and associated components and the operational impact of the failure modes is given in Table 4.1-1. The operational impacts used are listed in Table 4-2.

Table 4.1-1 Solar Array Failure Modes and Impacts

Failure Mode	Cause	Effect	Operational Impact
Solar Array Section			
- Open	Broken Interconnect, Shadowing	No Power	2-4
- Short	Insulation Breakdown, Arcing	No Power	2-4
Cover Slide, Loss of Transmissivity	Micrometeorites, Outgassing from S/C, Process Failure	Reduced Power	4-6
Loss of Cover Glass Transmissivity	UV Degradation; Cover-Glass Erosion; Plume Deposits	Reduced Power	4-6
Isolation Diodes Open	Process Failure, Lack of Redundancy	No Power	2-4
Failure to Track Sun, Catastrophic	- CMD Fail - Servo Fail - Motor Fail	Reduced Power	4-6
Degraded Ability to Track Sun	- Pointing Impairment, Structure - Servo Oscillation	Reduced Power	4-6
Slip Ring Open/Short	- Lack of Redundancy - Inadequate Test	No Power	2-4

There are no solar-array failures that will cause a catastrophic loss of the spacecraft. This assumes there is sufficient redundancy that loss of a solar-array section or ring can be tolerated. Depending on the amount of redundancy present, the impact of losing a solar-array section can range from complete loss of mission functions to loss or degradation of EPS functions. Degradation of the cover slide or anti-reflective coatings can range from degradation of EPS capability to no significant impact.

4.2 ENERGY STORAGE FAILURE MODES AND OPERATIONAL IMPACT

4.2.1 NiCd Cell and Battery

Failure Modes - A summary of failure modes for NiCd cells is given in Table 4.2.1-1. To be useful, the cells must be assembled in series and parallel interconnections. Approximately 200 series-connected cells would be required for a 300-Vdc system, and about 22 cells in series would be required for a 28-Vdc system.

A battery requires operational control and auxiliary systems control. Operational control consists of the following three categories:

1) Charge Control

2) Discharge Control

3) Offline Operations

Typical charge-control limits cell or battery voltage as a function of temperature. Amp-hour integration is usually required for depth-of-discharge determination. Discharge control involves limiting the maximum DOD. For a battery with several hundred cells, individual-cell or multiple-cell module monitoring may be required to guard against cell reversal during discharge. Cell reversal can result in gas generation, case rupture, and loss of battery.

Table 4.2.1-1 Basic Failure Modes of Nickel-Cadmium Battery Cells

Failure	Causes	Effect
Low discharge Voltage	Loss of capacity; reduction in active material within cell.	Possible Bus undervoltage during discharge.
Loss of Capacity	Redistribution of electrolyte or active material within cell. Overcharge or cell reversal.	Possible unexpected bus-voltage drop during discharge.
Open Cell	Seal failure; break in electrode-terminal connection.	Possible unexpected bus voltage drop and electrolyte or power loss during discharge--whole string of cells deactivated.
Shorted Cell	Electrode bridging by conductive active discharge. Contact between electrodes caused by separator deterioration.	Possible bus undervoltage active material; power loss during charge and discharge. Can cause excessive overcharge of the remaining cells, leading to premature failure.
Cell Overpressure (Limited to Sealed Cells)	Gas generation by overcharge or cell reversal.	Possible cell explosion or rupture.

Offline operations include capacity measurement, reconditioning, and equalization charging (in the case of several batteries connected to one bus). Thus, it is seen that a NiCd battery has traditionally required extensive operational controls owing largely to uncertainties in its performance with time. A summary of battery-operational control failures, their effects, and criticality, is given in Table 4.2.1-2.

When batteries are charged or discharged, they generate heat. If this heat can not be removed, the battery will overheat. NiCd batteries are generally constrained to operate within narrow temperature limits, e.g., 5°C to 15°C, to assure mission life. The upper temperature limit is sometimes controlled by minimizing battery discharge or minimizing or terminating the overcharge.

Table 4.2.1-2 NiCd Battery Operational Control Failures

Failure	Effect	Criticality
Charge Control		
- Overcharge	Reduction of Life	4
- Undercharge	Undercapacity	5
- DOD Determination Failure	Loss of Ability to Accurately Charge and Discharge	5
Discharge Control		
- Cell Reversal	Cell Overpressure Failure	4
- Excessive DOD	No Significant Impact if not Repetitive	6
	Repeated, premature battery failure	5
Offline Operation		
- Capacity Measurement Error	Erroneous Information about Battery State of Health, Possible Future Over/Under Use	5
- Reconditioning Failure	Cells Not Rejuvenated or Equalized	5

Operational Impact - A summary of generic battery failures for the three basic operating modes is given in Table 4.2.1-3. Under the assumption that the batteries would not have any function during launch or initial orbital assembly, there is no impact from failure here. It is possible that loss of battery capacity could cause a partial loss or degradation of mission functions, depending on the amount of capacity safety factor initially used.

Table 4.2.1-3 Other Battery Failures

Failure	Causes	Effect
Complete Battery Loss	Cell Failure. Cell Reversal Due to Discharge Failure	Loss of EPS capability.
Battery Capacity Degradation	Excessive DOD Due to Control Failure. Insufficient Charge Due to Control Failure	Degradation of EPS capability (bus power).
Thermal Control Failure, High Temperature	Inability to Reduce Loads	Degradation of EPS capability.
Heater Blanket	Broken Electrical Leads	Degradation of EPS capability.

The impact of a single-cell failure will depend on whether there is onboard cell-level* sensing, switching, and replacement available. If onboard cell-replacement is not available, then the impact will be loss of EPS battery capability. There would be a further schedule, maintenance, and STS flight impact to remove and replace the bad cell. If onboard-cell replacement were available, the bad cell would be automatically replaced and the EPS would have full capability. The impact of the failure would be loss of fault-management capability in the EPS. The number of spare cells would have been reduced by one. When all of the spare cells are switched online, then the next cell failure would result in a battery loss. This is an example of how active redundancy management can reduce the severity of a fault impact.

4.2.2 NiH_2 Cell and Battery

Failure Modes - A summary of the failure modes of a NiH_2 cell is given in Table 4.2.2-1. NiH_2 has all the generic failure modes of any battery cell such as open, short, and loss of capacity. NiH_2 batteries

*For a battery string containing 200 cells in series, "cell-level" can be "module-level," with the module consisting of 10 to 20 cells that can serve as the lowest replaceable unit.

require operational control similar to NiCd batteries and are susceptible to control failures. A unique feature of NiH_2 batteries is that their available capacity is proportional to the internal pressure, and, therefore, pressure can be used as a control parameter. They require pressure vessels and thus are susceptible to a mechanical failure that permits H_2 to escape from a given cell. A control failure that can cause loss of a NiH_2 battery is overcharging. Overcharging causes a pressure buildup that can cause a pressure-vessel failure and loss of a battery. Pressure-vessel rupture presents a potential hazard. Worst-case analysis and qualification of the pressure vessel are mandatory to guarantee that there would not be a safety hazard from an exploding pressure vessel.

Operational Impact - All failure impact identified for the NiCd battery applies to the NiH_2 battery also.

Table 4.2.2-1 Failure Modes of Nickel-Hydrogen Battery Cells

Failure	Causes	Effect
Open Cell	Seal failure; escape of hydrogen gas; break of electrode terminal connection.	Possible unexpected bus-voltage drop and loss of power during discharge; loss of battery.
Shorted cell (Primarily a Common Pressure Vessel Cell Failure).	Electrolyte and active material redistribution.	Possible bus undervoltage and loss of power during charge and discharge.
Pressure Vessel Failure	Excessive gas generation due to overcharge, charge-control failure.	Same as open cell. Cell-case rupture hazard.

4.2.3 Regenerative H_2O_2 Fuel Cell

Failure Modes - A regenerative H_2O_2 fuel cell consists of an electrolysis module that separates H_2 and O_2 from H_2O; a fuel cell to generate electrical power from H_2 and O_2 and run auxiliary equipment; a source of power for the electrolysis module (assumed to be a solar array); storage for H_2O, O_2, and H_2; a heat exchanger; a radiator; pumps; and a voltage regulator for the electrolysis module.

Failures can be grouped in three areas:

1) Electrolysis unit;

2) Fuel-cell unit;

3) Auxiliary equipment.

A potential hazard exists when free oxygen and hydrogen are present in a system. However, there is general agreement among the fuel-cell manufacturers that a catastrophic failure is highly improbable. By design, they keep the volumes of free hydrogen and oxygen as small as possible. The electrolysis and fuel-cell units are quite similar, their main difference being the catalysts used to optimize operation as an electrolyzer or fuel cell. The major failure mode in the electrolyzer or fuel cell is a membrane failure that allows O_2 into the H_2 manifold or H_2 into the O_2 manifold. Considering present designs, the highest unreliability is in auxiliary equipment. Pumps are known to wear out from mechanical failure. The voltage regulator for the electrolysis unit is subject to all the standard failure modes of power-processing electronics.

A summary of the failures that can cause shutdown of the electrolysis and fuel-cell subsystems is given in Table 4.2.3-1. These failures are detected by the following types of sensors:

1) Absolute pressure;

2) Differential pressure level;

3) Temperature;

4) Voltage and current.

Table 4.2.3-1 Regenerative H_2O_2-Fuel-Cell Failure Modes

Electrolysis Subsystem	Fuel-Cell Subsystem
H_2 in O_2 Manifold	H_2 in O_2 Manifold
O_2 in H_2 Manifold	O_2 in H_2 Manifold
Module Current High	Module Current High
Module Voltage High	Module Voltage Low
Cell Voltage Low/High	Cell Voltage Low
H_2 Separator Level Low/High	Product H_2O Level Low/High
O_2 Separator Level Low/High	Module Coolant Pressure Low
O_2 Separator Level High	H_2 Outlet Pressure Low/High
H_2O Circulation Low	O_2 Outlet Pressure Low
Circulating Pump Pressure Low	O_2 Inlet Pressure High
H_2O Resistivity Low	(O_2 Out - H_2 Out) Pressure Low
H_2 Pressure Low/High	(O_2 In - O_2 Out) Pressure Low
O_2 Pressure Low/High	Piston Pressure Low
O_2/H_2O Outlet Temp High	Pad Pressure Low
H_2 Condenser Temp High	H_2 Temp Low/High
O_2 Condenser Temp High	O_2 Temp Low/High
H_2O Temp Low	
H_2O Pump Pressure Low	
Module Coolant Temp High	

Operational Impacts - A summary of the regenerative fuel-cell failure modes and operational impacts is given in Table 4.2.3-2. The operational impact of the failure is highly dependent on the amount of redundancy available to correct the failure. If there were n units available and only n-1 were required to satisfy all requirements, then the impact of the first unit failing would be only a loss of fault-management capability. On the other hand, the operational impact of the second unit failing would be a loss of EPS capability.

Failure of an electrolysis unit would mean (1) loss of capability to store solar-array energy, and (2) loss of functional redundancy to produce breathable oxygen from water and electrical power. If the electrolysis unit were used to convert wastewater in a closed system, then there could be a buildup of wastewater. Loss of a fuel cell would result in loss of electrical-power capability and loss of ability to produce potable water from hydrogen and oxygen.

Table 4.2.3-2
Regenerative Fuel-Cell Failure Modes and Operational Impacts

Failure Mode	Cause	Effect	Operational Impact
Electrolysis Unit Failure	Membrane Failure	Can not convert water into hydrogen & oxygen.	4, 5
Fuel-Cell Unit Failure	Membrane Failure	No electrical output. Can not convert hydrogen and oxygen into electrical power.	4, 5
Auxiliary-Equipment Pump Failure	Mechanical Failure	Degradation or loss of water circulation in electrolysis unit, loss of ability to store solar-array energy.	4, 5
Solar Array Voltage-Regulator Failure	Lack of Redundancy	Degradation or loss of electrical input to electrolyzer. Loss of ability to store solar array energy.	4, 5
Thermal Control Not Able to Maintain Temperatures	Lack of Redundancy	Loss of capacity in electrolysis & fuel-cell units. Can not store energy, can not make electrical power from H_2 and O_2.	4, 5

Loss of solar-array capability directly affects energy-storage capability. The regenerative fuel subsystem generates waste heat in both the electrolysis and fuel-cell units. If the thermal-control subsystem can not dissipate this waste heat, then both the energy storage and electrical power output of the regenerative fuel cell are directly affected, causing a reduction in available bus power.

4.3 POWER CONDITIONING FAILURE MODES AND OPERATIONAL IMPACT

4.3.1 Programmable Power Processor (P^3), Buck Dc-Dc Converter

Failure Modes -

1) Shorted pass transistor, P3 in voltage-regulator mode, driven by voltage source. A shorted series-pass transistor is an admissible failure mode for a P^3. The effect is that the source is connected to the output. The P^3 design includes a system-level overvoltage sensor and shunt switch to keep the voltage below unsafe levels and cause the input fuse to open. If the load bus voltage drives up to the overvoltage limit, the external shunt switch turns on, and the input fuse on the P^3 opens. This prevents possible damage to the user loads.

   If there is a double failure, the shorted series-pass transistor and the overvoltage sense fails, and then the source would be connected to the loads. The input fuse might or might not open. This double failure may damage the user loads.

2) Shorted pass transistor, P3 in battery charger mode, driven from a solar array. In the battery-charger mode, the P^3 would be driven by a solar array. The effect of the failure would be to connect the battery across the solar array. The battery would change the operating point of the solar array and the array voltage would decrease to that of the battery. The P^3 can not correct this condition because all it can control is its pass transistor. This will generally not be a safety problem. Detection and correction times of minutes probably will be acceptable. The P^3 detects a shorted pass transistor. This status signal can be used to open a contactor to remove the P^3 from the solar array.

3) Input over voltage or current, output over voltage or current, and internal over temperature. The effects of any of these failure modes are:

a) P^3 is sent to shutdown state by its internal microprocessor,

b) An external reset is required before P^3 will turn back on,

c) Output overload current is caused by user loads.

The P^3 will support an overload for a programmed length of time, then it will automatically turn off and wait for a programmed length of time. It will then automatically turn on. If the overload is gone, it will continue normal operation. If the overload is still present, it will continue cycling on/off/on until it receives an external command. The net effect is that the P^3 turns itself off.

Operational Impacts - A summary of P^3 failure modes, causes, effects, and operational impacts is given in Table 4.3.1-1. The most serious of these is loss of mission functions owing to an undetected and uncorrected shorted pass transistor that results in connecting the high-voltage input to the low-voltage output loads. This results in destruction of the user loads. Normally, this fault will be detected and corrected by a system-level shunt regulator. In this case, the user loads are not destroyed, and the operational impact is reduced to loss of fault-management capability.

The operational impact of low output-power can range from degradation of mission function to loss of fault-management capability, depending on the amount of redundancy available. If there were no redundancy, and the P^3 with low output-power could not be replaced, then the impact would range from degradation of mission function to loss of EPS function. If there were a redundant component that would allow replacement of the failed P^3, then the operational impact of the failure would be reduced to loss of fault-management capability.

The other faults shown will generally result in an operational impact of a degraded EPS function if there is no standby redundancy in which to switch. If there is standby redundancy, then the impact would be lowered to loss of fault-management capability.

Table 4.3.1-1 P3 Failure Modes and Impacts

Failure Mode	Cause	Effect	Operational Impact
V_{Out} Hi	Shorted Pass Transistor, Failed OV Sensor	Damage Loads	2
	Shorted Pass Transistor (Corrected)		5
Low Output Power	Control Circuit Failure	Partial Loss of Power	3,4
Efficiency	Filter Capacitor Leakage, Pass Transistor Switching Loss Increase, Saturation Voltage Increase	Assembly Overheats	4
V_{In}Hi	System Anomaly	Assembly may Fail	4
I_{In}Hi	Hi-Leak Input Filter Capacitor	Assembly Overheats	4
High Temp	Thermal Subsystem Failure	Assembly Overheats	4
I_{Out}Overload	Component Degradation, Load Fault, or Overload	Output Overheats	4

4.3.2 Transformer-Coupled Converter (TCC), Buck-Derived Dc/Dc Converter

This type of power converter can be used for main or local (housekeeping supply) power-conversion functions. This configuration has a transformer to isolate input from output. The configuration can be that of a buck-derived converter or a Cuk Converter.

Failure Modes -

1) Shorted series-pass transistor. This is a major failure mode. Because there is a transformer between the input and output, the input is not connected to the output. The load voltage does not go up; instead, it decays to zero. At the input, there will be a short across the source and an input fuse must open to clear the fault.

2) Control electronic. A second major failure mode is associated with the control electronics. One type will cause 100% duty-cycle operation and try to drive the output into overvoltage. Whenever the overvoltage detector works, the TCC will be turned off. If there is a double failure, the control circuit fails to 100% duty cycle and the overvoltage fails to operate, and then loads can be destroyed. The cause of this failure is inadequate redundancy. Possible fixes are redundant control circuits, redundant local overvoltage detectors, or a system-level overvoltage detector combined with a shunt switch.

The TCCs, as they are known to exist today, do not have the extensive self-protection and local automation features that the P^3 has, but they could be added.

Operational Impacts - A summary of the TCC failure modes and operational impact is given in Table 4.3.2-1. An undetected and uncorrected output overvoltage can result in loss of mission functions. Also, a failure where no power is provided to the user loads can result in an operational impact or loss of mission functions. If there are standby redundancy and timely detection and correction, then the operational impact of the above two failures can be reduced to loss of fault-management capability. The operational impact of converter-efficiency degradation can range from degradation of EPS capability to no significant impact. The actual impact will be strongly affected by the degree of converter overheating and how closely the converter shutdown limits are approached.

Table 4.3.2-1 Transformer-Coupled-Converter Failure Modes and Impacts

Failure Mode	Cause	Effect	Operational Impact
V_{Out} High	Control Fail, Overvoltage Protection Failure	Damage Loads	2
No Output	Shorted Pass Transistor or Open Component	No Power to Loads	2
Degraded Efficiency	Filter-Capacitor Leakage Increase, SW Transistor Loss Increase	Assembly Overheads	4,5,6

There are failure modes where the efficiency or ripple voltage is degraded. For these failure modes, the TCC will function, but will not result in optimum operation. The heat-rejection requirement will be increased owing to lower efficiency. This degraded component can be relegated to backup status. The longer-term impacts would be schedule, maintenance, STS flights, and failure analysis to determine the reason for the degradation.

4.3.3 Series-Resonant Inverter (SRI), Dc to Ac

Failure Modes -

1) Shorted power semiconductor. This is a major failure mode that results in a short across the input and a control-circuit failure that results in an output overvoltage. There are control failures that result in loss of output. A load fault does not harm the SRI, because inherently it is a current source and can supply shorts without damage.

2) Control circuit malfunction resulting in simultaneous conduction of power switch. The SRI uses power semiconductors as switches in the full-wave rectifier bridge. An inherent failure occurs if the control circuitry allows both power semiconductors to conduct at the same time. When both power switches conduct, they are across the power source and can be destroyed. Electronic protection circuits for this failure mode are required of all SRI circuits. When a power switch is shorted due to either a control or switch failure, there is a fault across the source. The fault must be cleared by a fuse. Should the fuse fail to clear the fault when there is a battery connected to the bus, there is a potential fire hazard due to wire overheating. If the source is only a solar array and no battery, then the fault currents would be limited and there would not be a safety hazard to wires. An external evaluation would be required to sense the failure, remove the SRI, and switch a backup online.

3) Commutating diode fail shorted. There are commutating diodes across each power switch. If one of them were to fail shorted, the next time the other power switch is turned on, there would be a short across the source. The impact would be the same as item 2 above.

4) Control circuit failure causing output to be overvoltage or no output. A second major safety failure mode is a control-circuit failure that allows the output to go overvoltage. There are two areas affected by this failure: first, user loads can be damaged; second, the SRI input capacitors can be driven overvoltage and destroyed. Further work is required to define how the input capacitors fail when they are driven overvoltage. The safing for shorted-input capacitors is for an input fuse to open.

   There are control circuit and wiring failures that will result in no output from the SRI. For these failures, external analysis is required to sense the failure and switch a backup unit online.

Operational Impact - A summary of the SRI failure modes and their operational impact on the mission phases is given in Table 4.3.3-1. A shorted power semiconductor is a safety hazard if it is not detected and corrected. The safety hazard occurs when the fault across the source is not cleared and wiring may be destroyed. An output overvoltage failure is also one that can propagate from the converter to the wiring and user loads if it is not detected and corrected. The output overvoltage could cause destruction of the user loads. To assess the impact of these failures, an assumption about redundancy must be made. If sufficient redundancy is provided to eliminate single-point failures, the impact would be partial loss or degradation of mission functions. In addition, there would be these impacts: time to assess the damage, delay assembly, immediate decrease in spacecraft capability, schedule/maintenance, and future STS flights.

Table 4.3.3-1 Series-Resonant-Inverter Failure Modes and Impacts

Failure Mode	Cause	Effect	Operational Impact
Power SCR Short	SCR Fail, Control Failure, Fuse Fail	No Output	2
Power SCR Open	Control or SCR Fail	No Output	2
Commutating Diode Shorted	Diode Fail, Fuse Fail	No Output	2
Load Short	Load Fail	No Output Power, SRI Not Harmed by Short	3
V_{Out} High	Control Fail, OV Protection Fail	Damage Loads	3
No Output Voltage	Wire Open	No Power Output	2,3
Degraded Efficiency	Filter-Cap. ESR Increase	Assembly Overheats	4,5,6
Resonant Caps. Fail on Overvoltage	Lack of Redundancy, Lack of Margin	No Output	2,3

If there were to be timely detection and correction for a source short or output overvoltage, then the impact would be lowered to loss of fault-management capability. The fault would be detected and corrected, and a redundant unit would be brought online. In this case, mission functions would not be affected. The EPS would function normally. There would be an impact on the reserve capacity of the EPS owing to the fact that a redundant unit was brought online. There would be a future impact on schedule/maintenance and STS flight to replace the failed component. Additionally, the loss of reserve capacity in the EPS could affect future space station operations if there were rules that required a certain level of reserve capacity. A SRI has a unique failure mode where a control-electronics failure can cause the input capacitors to fail on overvoltage. This failure is noted to ensure that (1) a thorough analysis of the overvoltage failure mode of the capacitors used is examined, and the package is sufficient to contain debris from a failure, and (2) there is not a catastrophic loss of the spacecraft.

An SRI will have multiple piece-part failures that will cause it to have no output. Assuming some redundancy, the impact of this failure should be limited to loss of fault-management capability. A decrease in efficiency of the SRI would result in less-than-optimum operation. This could result in higher demands on the thermal subsystem. Depending on the degree of efficiency degradation, the SRI would be acceptable for use. A good configuration-management philosophy would require the degraded SRI be placed on standby and the backup unit used.

4.3.4 Solar-Array Voltage Controller

Failure Modes - There are several design configurations and concepts for controlling the upper limit of the solar-array bus voltage. The main ones are the following:

- Multiple-Array Segment Switching
    - Series-Switch/Series-Array Segments
    - Shunt-Switch/Series-Array Segments
    - Series-Switch/Parallel-Array Segments
    - Shunt-Switch/Parallel-Array Segments
- Full Analog Shunt Regulator
- Partial Shunt Regulator
- Hybrid Shunt Regulator

Table 4.3.4-1 lists the major failure modes, effects, and operational impacts that are summarized below.

Table 4.3.4-1 Array-Voltage-Control Failure Modes and Impacts

Configuration	Failure Mode	Effect	Operational Impact
Series-Switching, Series Array	Switch Fail Closed Switch Fail Open	+ *	4 6
Series-Switching, Parallel Array	Switch Fail Open Switch Fail Closed	* +*	4 6
Shunt-Switching, Series Array	Switch Fail Closed	*	4
	Switch Fail Open	+	6
Shunt-Switching, Parallel Array	Switch Fail Closed Switch Fail Open	* +	4 6
Full Shunt	Shunt Fail Shorted	No Power	2,3,4
	Radiator Failure	+	3,4
Hybrid Partial/ Full Shunt	Failure of One of n Digitally Controlled Switches - Closed - Open Failure of One of n Linear Shunt Regulators	 * + +	4 6 4
Series-Switching, Series Array with Full Shunt	Switch Remains in One Position	+	4
	Single Component Failures That Will Cause Oscillations	Lose Control	3,4
Partial Shunt	Shunt Fail Shorted	Reduced Power, Lose Control	4
	Piece-Part Failure Causing Oscillation	+	4

* Partial Loss of Power
\+ Partial Loss of Control

a. Multiple Array Segment Switching Failures. For the array-switching configurations, the major failure mode is a switch stuck in one position or stuck in the middle with no contact at all. The switch failure can be caused by an open or short in the switching element itself, or a control or interface failure. The cause of all these failures is insufficient redundancy in the switches and control circuits. The effect of the failure would be loss of a string, or loss of control of a string. The impact of the failure or array output power would depend on the number of strings present.

b. Full Analog Shunt Failures. A full shunt can fail by shorting or opening. If the full shunt shorts, there is no array output voltage. If the full shunt fails open (shunt switches fail open, or control failure) the array output voltage is present, but it can not be limited by the full shunt. A full shunt is required to dissipate the total array power; therefore, it is strongly affected by the thermal-control subsystem. If the thermal-control subsystem is not able to accept all the waste heat from the full-shunt regulator, then the EPS output capability would be reduced.

c. Partial-Shunt Regulator. A partial-shunt regulator will be subject to all the failures of a full shunt except that the thermal-dissipation control problem will not be as severe. The partial shunt is not required to dissipate the full-array power. Therefore, the demands on the thermal control subsystem are not as severe as with the full shunt.

d. Hybrid-Shunt Regulator. The hybrid-shunt regulator will contain both discrete and continuous shunt switches. There can be both full and partial shunt switches. The array will be partitioned into different groups of series and parallel solar cells for control. The strings can either have equal or unequal power. One method is to use binary weighting of the power.

A generic hybrid system could have binary-weighted parallel strings with discrete, partial-shunt switches on all but the smallest string. The smallest string could have a continuous shunt.

The generic hybrid system is not required to dissipate the entire array power. Because this system contains a number of binary-weighted, discrete, partial-shunt switches, the loss of any one switch will result in either a loss of control for an open switch, or loss of power from a parallel branch for a shorted switch. The continuous shunt switch is used for a fine control. Because it generally will have the smallest power-handling capability, loss of the continuous switch will result only in loss of fine-control capability, and not in loss of the array.

Operational Impacts - Failures in the photovoltaic-array switching will not affect launch because these components are not operational during launch. These components generally will not have an initial onorbit assembly function. The impact of a failure during onorbit assembly could cause an assembly delay, schedule and maintenance impact, and an impact on future STS flights. If no single-point failures are assumed, one failure in an array voltage control unit would result in the loss of only a fraction of the total array. Therefore, space station operations could be affected by less-than-expected solar-array power. The impact of the failure during an orbit assembly could be described as partial loss or degradation of mission functions until the faulty unit is replaced.

Failures in the switched controllers will result in loss of a fraction of the array power or some loss of control. The impact will be a loss of EPS capability. This should not result in a loss of mission function. A decision will be required as to when to replace and repair.

A real full-shunt regulator would be modular and redundant. A single-point failure causing loss of all array power or ability to limit the array voltage would not be allowed to happen by designing in redundancy. If a second failure causes a full-shunt switch to fail shorted, then the array voltage could be held at some low value until the fault were corrected. In a modular redundant system, an open failure would result in some loss of capability to limit the array voltage under light-load conditions. These failures would probably not affect operations during sunlight or eclipse. A loss of voltage-limit capability can occur during an eclipse-to-sun transition under light-load conditions. The array would be cold coming from eclipse, and its voltage would go to maximum at the eclipse-to-sun transition. Inability to limit the maximum voltage of a cold array could possibly cause damage to loads or require the array to be unloaded until it warmed in the sun and its open-circuit voltage decreased. It is expected that passive radiators would be used to get rid of heat from the full shunt. Attitude constraints or abnormal vehicle-orientation modes could restrict the ability to dissipate waste heat and could affect the EPS. The net impact of these failures would be classified as loss of EPS capability.

A flight-type hybrid photovoltaic-array voltage controller is also expected to be modular and redundant. A hybrid controller would have a graceful failure mode, where each failure would result in a specified loss of control capability or power from the array. If the array parallel strings were binary weighted, loss of the largest branch could be one half of the array. If n equal branches were used, then loss of one would result in only loss of 1/n of the total array. The impact of hybrid controller failures on orbital operations is classified as loss of EPS capability.

4.3.5 Housekeeping Power Supplies

Failure Modes - Housekeeping supplies are usually contained within an EPS component such as an array-control unit or within a power converter. The purpose of these supplies is to provide multiple regulated voltages to a specific black box. They can be either linear, dissipative devices for onboard regulation, or switched-mode topologies. These supplies are subject to all the failure modes of switched-mode converters and linear-dissipative regulators. These supplies are subject to over/undervoltage, oscillations, out-of-specification ripple, and frequency failures in clock-drive circuitry. The basic causes of these failures are usually attributed to insufficient redundancy, lack of worst-case design, and insufficient test.

Operational Impact - The effect of failure of a housekeeping supply will be the loss or degradation of an EPS black box that it is powering. The ultimate impact of the housekeeping-supply failure will thus be determined by the impact of losing the EPS black box. The impact of the failure of a specific housekeeping supply will be limited to loss of EPS capability, or loss of fault-management capability.

4.4 POWER-DISTRIBUTION DEVICE FAILURE MODES AND OPERATIONAL IMPACT

4.4.1 Magnetic Latching Relays

Failure Modes - A summary of the generic failure modes of a magnetic latching relay is shown in Table 4.4.1-1. The failures on a relay must be considered along with failures of the relay drivers and loads.

Table 4.4.1-1 Magnetic Latching Relay Failure Modes

Failure Mode	Cause
Fail to Transfer	- Relay Coil Open - Interface Failure - Control Electronics Failure
Relay Oscillates	- Control Failure That Powers Set and Reset Coil at the Same Time
Relay Driver Fails	- Voltage Suppression Diode across Coil Opens, Driver Fails on Inductive Overvoltage the Next Time It Interrupts Coil Current
Contacts Burnt Open or Welded Shut	- Excessive Fault Current, Voltage Suppression Diode across Inductive Load Opens, then Relay Tries to Interrupt Inductive Current, Contact Failure due to Inductive Voltage Transient
Spurious Transfer	- Command Failure - Control Electronics Failure

A major magnetic-latching relay-failure mode is failure to transfer on command. This can be due to internal relay failure (open coil, mechanical contact failure, welded contacts), interface failure, or driver-electronics failure. Another failure mode is relay oscillation. This can be caused by a control failure that commands the set and reset coils at the same time. A magnetic latching relay has both set and reset coils. These coils require parallel diodes to prevent an inductive voltage rise when the current is interrupted. Should a diode open, it would not be detectable until the driver tried to turn off the coil current. At this time, the driver would fail owing to the inductive voltage transient. This is an example of a propagating failure. Contacts can be burnt open or welded shut by fault currents or by interrupting an unprotected inductive current. Spurious transfer of a relay can be caused by a command- or control-electronics failure. Relay position can be determined directly by inference. A failure in the direct position indicator (sense voltage across a spare set of contacts) can cause a good relay to be indicating bad. This failure could then require the use of inference (conclusion based on indirect sensing) to resolve an anomalous situation.

Operational Impact - Relays have recognized failure modes. It is expected that a space-station-wide criticality classification of loads and redundancy requirements for relays will be made. For this reason, the impact of the failure of a relay in a specified redundancy configuration are discussed below.

System-level analysis normally classifies loads and establishes redundancy requirements for each load class. Possible relay redundancy requirements are as follows:

1) Failure of a single relay will not result in more than TBD signal or power-connection failure. Example--a single relay;

2) Failure of a single relay will not prevent connecting a load. Example--two parallel relays;

3) Failure of a single relay will not prevent disconnecting a load. Example--two series relays;

4) Failure of a single relay will not prevent normal operation of a load. Example--four relays, two in parallel in series with two in parallel.

Table 4.4.1-2 lists the operational impact of a single relay failure in each of the above relay-redundancy configurations. For a single relay, a failure can result in not being able to connect or disconnect a load. For two series relays, a fail-closed mode has no effect; the load can be removed and an open failure always causes load removal. For parallel relays, an open failure has no effect other than loss of redundancy. A closed failure means the load is always connected. For four relays in series and parallel, a single relay has no operational effect. Its impact is loss of redundant backup.

Table 4.4.1-2 Relay Failure Impact by Redundancy Configuration

Relay Redundancy Configuration	Failure Mode	Effect	Impact
Single Relay	Fail Open Fail Closed	Does not connect load. Does not remove load.	3-4 3-4
Two Relays in Series	One Fail Open	Does not connect load. An open failure always causes load removal.	3-4
	One Fail Closed	None.	5
Two Relays in Parallel	One Fail Open	None.	5
	One Fail Closed	Does not remove load.	3-4
Four Relays, Two Parallel in Series with Two in Parallel	One Relay Always Closed or Open	None, normal operation.	5

4.4.2 Motor-Driven Switches

Failure Modes - The generic failure modes of a motor-driven switch are as follows:

1) Fail to transfer on command (motor failure, control-electronics failure);

2) Spurious transfer (command or driver-electronics failure);

3) Mechanical damage to unit if both engage/disengage coils are activated simultaneously (control-electronics failure).

A motor-driven switch is an electromechanical device with motors, gears, and limit switches. Failures associated with a motor are open, shorted, or partial shorts of the coils. These can result in failure or degraded operations. Gear trains are subject to tooth wear-out, particle generation, and bearing failure that can result in the device failing to transfer. Limit-switch action is essential to turn off power to the drive coils after the unit has engaged or disengaged. Limit-switch failure can result in the motor driving too far and mechanical failure of the gear train. The electrical contacts are subject to being burnt open or welded shut by fault currents or interruption of unprotected inductive currents.

Operational Impact - A motor-driven switch performs the same functions as a relay, except the loads it switches are generally much longer than relay loads. The remarks for impacts of magnetic-latching-relay failure are applicable to motor-driven switches.

A major use of motor-driven switches is to connect and disconnect high-current sources (e.g., ground supply and batteries) from buses. For this type of application, failure of a motor-driven switch to engage would be the same as the loss of a battery. Once a battery is connected to a bus, the motor-driven switch would not normally be operated. A failure could occur that would prevent the switch from disengaging, but it would not normally be detectable until a disengage command is given.

A scenario for assessing the impact of a motor-driven switch failing to disengage and remove a battery from a bus during orbital operations is as follows. Suppose that it were required that a battery should be removed from a bus, either for maintenance on the battery or on the load side of the bus. When the switch fails to disengage, the battery is not removed from the bus. With 270-Vdc batteries, a safety hazard would exist when performing maintenance on the load side of the bus. Depending on the space station safety requirements, maintenance could be prohibited with this failure. The battery would be composed of a large number of cell modules. Battery maintenance would consist of replacing these modules. It is expected that safety requirements would require that the battery be floating so one side of a module could be grounded. When the motor-driven switch fails to open, the battery can not be isolated from ground. A safety hazard would exist for the removal of modules and the battery. Space station safety requirements could prohibit maintenance in certain cases.

The impact of a motor-driven switch failing to open and remove a battery from a bus and ground during orbital operations could create a safety hazard for maintenance. Safety requirements would probably require that there be a manual means of isolating the battery from the bus and return before maintenance is allowed to proceed.

During maintenance operations when a motor-driven switch is required to be disengaged, the impact of spurious engaging could create a safety hazard. The impact of a computer command connecting a battery to a bus while maintenance is in progress is such a serious hazard that design rules may require a manual disconnect of motor-driven-switch power during maintenance.

4.4.3 Remote Power Controllers

Failure Modes - A remote power controller (RPC) is a solid-state switch that performs all the functions of a magnetic latching relay plus the additional functions of circuit breaker, fault current limiter, current-rise-time limiter, and current-fall-time limiter. An RPC is controlled by a logic-level signal. An RPC has all the generic failure modes of a magnetic latching relay plus several additional failure modes unique to an RPC. RPCs can be used in redundant configurations in the manner of magnetic latching relays. A summary of the RPC-unique failure modes and operational impacts are shown in Table 4.4.3-1.

Table 4.4.3-1 RPC-Unique Failure Modes and Impact

Description	Failure Mode	Effect
RPC is hi-gain feed feedback circuit.	Piece-part failure in stabilization loop.	Output of RPC oscillates. Load may not operate. Possible overdissipation in RPC.
Redundant series-pass transistors, individually fused emitters.	Pass transistors short, emitter fuses open.	No measurable impact. Nondetectable loss of redundancy. Graceful failure mode.
Limit rate of current rise (di/dt).	Piece-part failure, no rate-of-current-rise limit.	Bus transient undervoltage. EMI.
Limit rate of current fall (-di/dt).	Piece-part failure, no rate-of-current-fall limit.	Transient voltage rise due to inductance. EMI. Opening Of a Voltage Suppression Diode On An Inductive Load Could Result in Voltage Rise Sufficient to Destroy RPC.
Limit fault current for approximately 3 s. Built-in thermal mass to absorb heat.	Piece-part fail, timer does not turn off current. All pass transistors short, all internal fuses open.	Fault current is cleared, but RPC is destroyed.
RPC is mounted on a cold plate to control steady temperature.	Thermal subsystem Failure or degradation. Rise in cold-plate temperature.	Rise in cold-plate temperature can impose limits on dissipation in RPC.

A remote power controller has a high-gain, electronic-feedback circuit meant to control multiple parallel power transistors. A description of the unique failure modes of an RPC is shown in Table 4.4.3-1. Because many of the RPC functions depend on analog circuitry, piece-part failures in the analog circuitry can cause RPC functional failures. The causes of the piece-part failures are insufficient worst-case design and analysis, process failure, or lack of redundancy.

An RPC will normally be mounted on a cooling plate to maintain desirable operating temperature. A failure in the thermal-control subsystem can affect the EPS by not controlling the plate temperature. An increase in the plate temperature could restrict the dissipation in the RPC.

Operational Impacts - RPC application is similar to magnetic latching (mag-latch) relays. Their redundancy requirements and impact of a relay failure in a redundant configuration is the same as mag-latch relays.

An RPC has more functions than a mag-latch relay. In addition to having a relay function, it is also used as a circuit breaker, fault current-limiter, and limiter for rate of current rise and fall. The impact of these unique RPC failure modes is shown in Table 4.4.3-1.

There is an undetectable degradation in an RPC. This occurs when one of the parallel series-pass transistors fails and its emitter fuse opens. The RPC can function normally, but some margin would be lost. The operational impact of this failure ranges from a loss of fault-management capability to no significant impact. The operational impact of other faults owing to piece-part failures will be in the loss-of-EPS-capability category. RPC degradation owing to failure of the thermal-control subsystem to maintain the cold-plate temperature for the RPC will range from loss of EPS capability to no significant impact.

4.4.4 Fuses

Failure Modes - A fuse has three major failure modes. First, it may fail to open at its specified rating. Second, a fuse may fail by opening at a current less than its specified rating. Third, a fuse may open owing to mechanical failure.

A limit of a fuse is its fault-clearing capability. If a fuse is used in an application where the fault current exceeds the fuse-clearing rating, then the fuse may not clear the fault. Also, fuses have maximum voltages for which they can be used in clearing. If a fuse is used at a higher than design voltage, it may not clear a fault.

Operational Impact - If no redundancy is provided (i.e., one fuse), the impact of a premature fuse opening is loss of the user load. The impact will be a partial loss of mission function. This would be an acceptable condition because the decision would have been made to tolerate loss of that load because it was classified low priority and was purposefully not provided with fuse redundancy. Failure of a single fuse to open at its rated current could result in a possible bus undervoltage. The operational impact could be a degradation of EPS function and affected user loads.

For series-redundant fuses, the effect of one fuse opening at less than its rating is to lose a user load. The impact can range from degradation of mission function to loss of EPS function. There is no significant impact from one series-redundant fuse not opening at its rating because it is assumed the other fuse will open.

For parallel-redundant fuses, there is no significant impact from one opening prematurely. It is assumed the other fuse will carry the load current. If one of the parallel redundant fuses fails to open at its rating, more current would be required from the source to clear the fuse. If the source were limited, the fuse might not be cleared, and an overload or undervoltage condition could result.

Table 4.4.4-1 summarizes the failure modes and impacts for several redundancy configurations:

1) One fuse, no redundancy;

2) Two fuses, series redundancy;

3) Two fuses, parallel redundancy.

Table 4.4.4-1 Fuse Failure Modes and Operational Impacts

Redundancy Configuration	Failure Mode	Effect	Operational Impact
One Fuse, No Redundancy	Premature Open	A user load removed.	3, 4
	Fail to Open At Rating	Fault or overload not cleared. Possible bus undervoltage.	
Two Fuses in Series	One Premature Open	Lose a user load.	3, 4
	One Fail to Open at Rating	None.	5, 6
Two Fuses in Parallel	One Premature Open	None.	5, 6
	One Fail to Open at Rating	Higher current required from source to clear both fuses. If source limited, fuses might not clear.	3, 4

4.4.5 Circuit Breakers

Failure Modes - Circuit breakers serve the same basic function as a fuse, but they are capable of resetting, either by manual or electrical means. Therefore, circuit breakers have all the failure modes of fuses plus additional failure modes unique to circuit breakers.

If a circuit breaker has a manual switching capability, then it is susceptible to a man's incorrect operation. If the circuit breaker has an electrical operation, then it can be affected by command errors, driver-electronics failures, and interface failures. Electromechanical circuit breakers have limits on the fault currents they clear. Grossly exceeding these limits can result in explosive destruction of the circuit breaker. Information desired about the operational state of circuit breakers is "open" or "closed." The state can be sensed directly by using an extra set of contacts as in mag-latch relays. The direct-sensing state indicator is subject to failures. These failures can give a false indication of the circuit-breaker state.

Operational Impact - Circuit breakers are generally used with fuses and controlled switches such as mag-latch relays or RPCs. The circuit breaker is usually an enabling function. The relay is generally used for repetitive switching. The impact of a circuit-breaker failure is thus similar to that of a fuse.

The failure of a circuit breaker to open could have a safety impact on maintenance similar to the failure of a motor-driven switch to open (see motor-driven switch failure impacts).

4.4.6 Cabling

Failure Modes - The generic failure mode of cabling and connectors is conductors or connections opening and insulation failing, with a resulting wire-to-wire or wire-to-structure short. Operational environments that cause mechanical damage are not included here. The principal operational environment that can cause degradation of insulation is temperature. Overvoltage can cause failure. Overtemperature would not cause an immediate insulation failure, but it could decrease the useful life of the insulation and require abnormally early maintenance or replacement.

For a 250-kW-class space station, power cables may require heat sinking to structure, or active cooling. For such a configuration, failures or degradation of the thermal-control subsystem could affect the EPS through power cabling.

A space station will experience modular buildup over a number of years. During this expansion, there is the potential for the change in cable locations that could affect thermal properties of the cable. Also, attitude-control modes such as gravity gradient have the potential for exposing cables to sunlight or darkness, both of which could affect cable thermal and insulation properties.

Operational Impacts - A summary of cable failures and other activities and their operational impacts is given in Table 4.4.6-1. Under the space station design requirement to eliminate single-point failures, the severest impact from a cable failure would be a partial loss of EPS capability or mission function. Insulation shorts from wire to wire, or intermittent insulation failures, can cause anomalous operation that could require partial shutdown for troubleshooting. Intermittent shorts in cables have the potential for extensive and time-consuming effort to discover, isolate, and correct.

Insulation can be degraded by overtemperature. Monitoring could prevent this failure mode. The immediate operational impact of insulation degradation is probably not significant. As the degradation progresses to the point where cable failure occurs, the operational impact will be loss of fault-management capability (it is assumed there is sufficient cable redundancy that a failure can be tolerated).

Table 4.4.6-1 Cabling Failures/Activities and Impacts

Failure or Activity	Effect	Cause	Operational Impact
Cable opens.	Lose loads.	Insufficient redundancy.	3, 4
Insulation shorts, wire-to-return.	Fault currents present, fuse or RPC must open to clear.	Insulation fault.	3, 4
Insulation shorts, wire-to wire.	Anomaly, a load energized spuriously, arcing.	Insulation fault.	3, 4
Insulation degradation due to overtemp.	None.	Lack of monitoring.	5, 6
Thermal subsystem failure.	Increase cable temperature, decrease allowable power thru a cable.	Failure in another subsystem.	4, 6
Modular buildup, or attitude-control mode.	Cable moved or thermal characteristics altered.	Activity of modular buildup or attitude control.	4, 6

4.5 SENSORS AND SIGNAL CONDITIONING FAILURE MODES AND OPERATIONAL IMPACT

Failure Modes - The primary sensors for the EPS are to monitor the fundamental or dc component of the following parameters:

1) Dc voltage and current;

2) Ac voltage, current, and frequency;

3) Temperature;

4) Pressure;

5) Solar irradiance.

All sensors have catastrophic-failure modes where they fail saturated or open. Each of the sensors has an error band. A sensor can degrade when its error exceeds its specified error band.

Sensors in ground-based applications require periodic calibration. If a periodic calibration requirement is imposed on sensors for the space station, then a sensor is good as long as its calibration date is valid. Once a sensor has exceeded its calibration date, then it may be considered bad and perhaps not useable for a manned application. Thus, there exists the possibility of sensors affecting the space station operation simply because of the exceeding of calibration dates or uncertainties about their accuracy.

Some sensors have well-known and predictable drifts due to temperature. This would constitute an accuracy degradation that could be removed by real-time adjustment if correction factors can be accurately determined.

Signal-conditioning circuits will use electronic piece-parts to convert the raw analog measurement into a single-ended dc voltage of a given range such as 0 to +5V, suitable as the input to an analog-to-digital converter. The signal-conditioning circuits are subject to catastrophic failure, drift, and accuracy degradation. Generally, the signal-conditioning circuit will be inseparable from the sensor for calibration and failure analysis.

Sampling circuitry involves multiplexers and analog-to-digital conversion. This signal conversion can fail catastrophically or can degrade. Signal-conversion circuitry is quite susceptible to grounding problems that could inject noise into an analog-digital converter.

Sampling implies bandwidth limits on the signal being sampled to ensure that Shannon's Sampling Theorem is satisfied. This means there may be an antialiasing filter in front of the sampler. An antialiasing filter may be either passive or active. Thus, the EPS can be affected by failures in an antialiasing filter in the sampling section of the data system. If an antialiasing filter fails by not restricting the bandwidth of the sampled signal, anomalies in the sampled data can result by frequencies greater than one half the sampling frequency being present in the input.

The impact of transducer failure will depend on whether the transducer is active, if its output is being monitored, and what weight is given to its output. If a transducer is active and its output available, then its failure would be loss of information about the EPS. The failed transducer could present an anomaly.

A summary of transducer failures and the resulting operational impact are given in Table 4.5-1. There should be sufficient sensor redundancy built into the space station so that the failure of a single sensor will have no significant impact. System-level trade studies will be required to identify how many sensor failures are permissible before EPS or mission functions are lost or degraded. When the vehicle is operated with failed sensors, it has a reduced fault-management capability. Requirements for fault management may require maintenance after a failure of particular sensor.

Degradation of a sensor by drifting outside of its error band can cause a lack of confidence in the measurements. The lack of confidence could cause overly conservative operating safety margins.

A sensor failing by exceeding its calibration due date is an example of a planning failure. The impact on orbital operations will depend on the quality control and safety requirements for the space station. If the philosophy is that an out-of-calibration sensor can not be used, then EPS capability can be lost or degraded. If all sensors were to go out of calibration on the same date, there could well be a requirement to curtail operations and make sensor calibration the highest-priority item.

Table 4.5-1
Sensor and Signal Conditioning Failure Modes and Operational Impacts

Failure Mode	Cause	Effect	Operational Impact
One Transducer Open	Lack of Testing; Inadequate Worst-Case Analysis; Lack of Process Control	None, If Redundant	5
One Transducer Out of Spec, Drift	Inadequate Operating Process Control; Piece-Part Quality Not Adequate	Decreased Information	4
Sensor Calibration Time Exceeded	Inadequate Planning	None for Short Times	6
ADC* Intermittent, Noisy, Ground; Antialiasing Filter Failure	Packaging, Manufacturing Test; or Installation	Decreased Information, Error	4

*ADC Analog-to-Digital Computer

4.6 POWER-TRANSFER-DEVICE FAILURE MODES AND OPERATIONAL IMPACT

Failure Modes - The components classified as power-transfer devices are slip rings, roll rings, twist flex, and rotary transformer (power electronics based on a series-resonant circuit). The major failure a "twist flex" of a slip ring, a roll ring, or a "twist flex" is an open-circuit condition that results in loss or reduction of array power.

The twist flex has a limited angular rotation. Vehicle operations could potentially affect the twist flex by commanding it beyond its allowable angular rotation. Assuming normal limit switches and safety interlocks, the impact of this operations failure would be to stop the orientation drive. This would result in degraded output from the solar panel.

The rotary transformer includes a series-resonant inverter with control and protection electronics along with the rotary transformer. This device will have all of the failure modes associated with a series-resonant inverter discussed in section 4.3.3. Failures associated with the transformer itself include open and shorted windings.

Operational Impact - A summary of the power-transfer-across-rotary-joint failure modes and operational impact of the failure modes is given in Table 4.6-1.

Table 4.6-1
Components for Power-Transfer-Across-Rotary-Joints Failure Modes and Operational Impacts

Failure Mode	Cause	Effect	Operational Impact
Slip Ring - Noise	Particle-Generation Brush-Plug Wear	Degraded Power	4
Slip Ring - Short	Insulation Failure	Loss of All Power thru Slip Ring	5-4
Roll Ring - Open	Mechanical Failure	Loss of All Power thru Roll Ring	5-4
Twist Flex - Open	Mechanical Failure of Flex Wire	Degradation of Full Loss of Power thru Twist Flex	5-4
Rotary Transformer - Open	Electronics Failure in Series-Resonant Inverter	Loss of Power from an Array Section	5-4

There are no failure modes that would result in a catastrophic loss of the spacecraft under the assumption there would be sufficient redundancy to tolerate the loss of power across a rotary joint. The operational impact of noise generated in a slip ring will probably result in a degraded EPS capability. There is also the possibility of electromagnetic interference with payloads. The complete failure of a component to transfer power will result in loss of all power from a solar-array section if there is no redundancy in the rotary-joint's power-transfer components. If there is no redundancy, the operational impact will range from degradation of mission functions to loss or degradation of EPS function. Assuming power transfer component redundancy is provided, the impact of the loss of power transfer component would be loss of fault-management capability.

4.7 AUXILIARY POWER SOURCES FAILURE MODES AND OPERATIONAL IMPACT

4.7.1 Lithium Thionyl Chloride ($LiSOCl_2$) Battery

Failure Modes - A summary of the failure modes is given in Table 4.7.1-1. This primary battery has no function during long-term, normal operations. Its intended use is that of an auxiliary, or emergency, power source. A significant shortcoming of this type of battery design is lack of state-of-health monitoring during the normal-operations period when it is not used. Should a failure occur during a long standby period, then the battery could fail or be degraded when it is activated to supply power. This condition can not be tolerated if the required power is for emergency purposes.

Table 4.7.1-1 Failure Modes of Lithium Thionyl-Chloride Battery Cells

Failure	Causes	Effect
Low end of discharge voltage and/or loss of capacity.	Cell operation at low temperature.	Abnormally early bus voltage drop, possible bus undervoltage following.
Low beginning of discharge voltage.	Long dormancy, generally at an above-normal temperature; cold temperature.	Possible transient bus voltage drop or power delay at the beginning of a discharge period.
Cell shorted.	Electrode or terminal bridging.	Possible bus undervoltage and loss of power.
Cell open.	Terminal-electrode break.	Possible bus voltage drop and loss of of an entire string of cells.

4.7.2 Chemical Turbomachinery

Failure Modes - Chemical turbomachinery or, more commonly, auxiliary power units (APU), can have failures associated with leaks in the reactant reservoirs, clogged tubes preventing reactant flow, pump failures, turbine mechanical failures (blades, bearings) and all the known failure modes of an electrical generator. A significant degradation of the energy capacity of chemical turbomachinery can occur by leaks of the reactants during periods of disuse.

Operational Impact - A summary of the generic failures of an auxiliary power unit and the operational impact are given in Table 4.7.2-1. If the state of health of an APU is not monitored during normal operations and it fails, there is no impact as long as it is not needed. If a situation arises where the APU is needed but has already failed, the next level of APU backups will have to be activated.

Table 4.7.2-1 Auxiliary Power Unit Failure Modes and Impact

Failure Mode	Effect	Impact Orbital Assembly	Operational Impact
Normal operation, failure not detected.	None	None, if no additional failure. Safety hazard if required.	5
Normal operation, failure detected.	None	None, if no additional failure. Schedule work around when failure known.	5
Auxiliary power active, then fail.	Switch over to backup.	Schedule maintenance impact.	5
Emergency shutdown system, false emergency.	Deplete battery capacity.	Operations. Schedule. Future STS Flights.	5

A most serious impact appears to be an undetected failure during normal operations. The impact is that fault-management capability has unknowingly been lost. Operating safety margins are not what they seem. If an APU failure is detected during normal operation, APU not needed, then operations could be changed to minimize the impact of the loss of the APU and timely maintenance repair, or replacement could be scheduled to restore the fault management capability.

If an APU is active and fails, and a backup of APU is activated, the operational impact of the failure is a loss of fault-management capability.

If limits in the emergency-shutdown system are too tight, or an invalid emergency is declared and the APU is activated, the reactants can be consumed. A rapid string of false emergencies and activation of the APU can result in APU-reactant depletion. The impact of failures in the emergency-shutdown system (false emergencies) is loss of EPS fault-management capability, future STS flight impact, and maintenance time for APU replacement. There can also be an operational impact by constraints owing to depleted APU backup capability.

4.8 OTHER ACTIVITIES AND FACTORS AFFECTING EPS PERFORMANCE

Table 4.8-1 is a list of basic space station operational characteristics and impacts on the EPS design, performance, and operation. A summary of other subsystem faults and activities that affect the EPS is given in Table 4.8-2. A brief discussion of major activities is given in the following paragraphs.

A key conclusion that can be made is that EPS automation is mandatory, in meeting the initial space station's basic requirements.

Table 4.8-1
Basic Space Station Operational Characteristics and EPS Design Implications

Activities/ Unknowns	Implications in Power Subsystem Design, Performance, and Operation
Long-Duration Manned Facilities	- Assure Crew Safety and Reduce Ground Support Requirements - Incorporate Flexible Fault Detection and Correction Capabilities - Replace Battery Modules and Array Sections Periodically - Accurately Keep Maintenance Logs and State of Health of Identifiable Elements or Sections - Accommodate Old- and New-Technology Components
Build and Repair in Space	- Facilitate/Simplify Capability to Add Key Components - Be Able to Determine Sate of Health Quickly and Accurately, and Predict Failure (e.g., Based on Trend Date) - Provide a "Turn-Key" Operation Similar to Large Terrestrial Photovoltaic Power Systems as Solar-Array Sections and Batteries Are Installed
Incremental Growth in Power	- Flexibility in Power-Hardware Designs and Additions in Orbit - Be Able to Quickly and Accurately Verify Performance after Assembly and Update Power-Capability Information - Be Able to Reconfigure Easily and Operate in Reconfigured Arrangement
Economical Payload Support	- Reduce Power Subsystem Maintenance, Monitoring, and Other Housekeeping Roles by Flight and Ground Crew to a Minimum - Accommodate Unproven (on Long Life) or New-Technology Hardware to Reduce Development Cost - Overcome Technology Limitations (e.g., Lack of Long-Duration Battery Life Testing and Uncertainties in Life of High-Voltage Batteries) - <u>In Situ</u> Learning of Capabilities and Limitations, e.g., Large Number of High-Voltage Batteries Operating in Parallel, in Lieu of Extensive Ground Testing
Verify Performance of Large Components along with Multiple Components Operating in Parallel	- Need to Develop Technology for Onorbit Checkout Techniques and Analytical Tools for Performance Determination - Resort to Analytical Approach in Predicting or Calibrating Performance - Solar Array Strings and Battery Strings May Have to Operate with Mismatched and New or Old Elements. This poses a Special Problem in Performance Optimization and Prediction

Table 4.8-2
Other Subsystem Faults and Activities That Can Affect EPS Performance

Subsystem	Failure/Activity	Effect	Operational Impact
Structures	Modular Buildup	Reduced Power	3,4
Thermal Control	Impaired Capacity to Dissipate EPS Waste Heat	Reduced Power	3,4
User Loads (All Subsystems and Payloads)	Shorts or Overloads	Bus Undervoltage	3,4
	Large Differences in Day and Night Power at Buses	May Reduce Bus Power Capability; Excessive Battery DOD	3,4
Attitude Control	Gravity Gradient Attitude Mode with No Solar Array Articulation;	Reduced Power	3,4
	Failure to Maintain Required Stable Attitude Because of Unknowns in Controlling Large, Flexible Structures	Reduced Power	3,4
Command	Degraded TM Data Transmission	Reduced Information to Ground	3,4
	Loss of CPU Power	Reduced Autonomy and Automation	3,4
Data	Software Maintenance	Reduced Power Capability	3,4
EPC/Crew Interface	Crew Commands, Displays,New Crew, Interface Ambiguity, Mistakes	Reduced Power Capability; Unintended Shutdown	3,4
EPS/Ground Operations Interface	Power-Management Configuration History; Audit Trail or Automated Activities; Training; Commands/ Displays	Reduced Power Capability; Inefficient Mission Planning	3,4

4.8.1 Flexible-Structures and Control-Subsystem Activities

A space station will contain a large, flexible structure. Knowledge of the low-frequency dynamics of large, flexible structures will be critical to the design and performance (stability envelope) of the control system. There is a probability that some in-situ characterization of the structure dynamics will be required. A significant mass in a space station will be in the solar panels. Hence, possible impacts on the EPS from flexible structures and the control system are low-frequency mechanical oscillations, solar-array pointing-accuracy degradation, and constraints on solar-panel slew rates.

4.8.2 Data Management Subsystem (DMS) Activities

Assuming that the EPS incorporates a reasonable amount of automation, it is expected that the EPS will not be highly dependent on the space-station DMS. Loss of channels or degradation of data rates in the DMS can result in loss of information about the EPS for ground use. If, for some reason, sampling times become larger than normal, information about the state of the EPS decreases. Preprocessing of critical EPS performance data by the EPS computer would significantly minimize the impact of DMS failures of this type.

Loss of space-station CPU capacity could result in some high-level EPS automation software being bumped out by higher-priority flight software. This could mean the PES high-level automation software would have to be run either in the STS or on the ground, or it would be cancelled and the functions performed by the ground.

The extent to which the EPS is automated, especially in handling and processing raw engineering performance data and commands, affects the cost of the data-management subsystem. If the EPS transmits only the significant engineering data, e.g., power, energy, and average quantities, rather than real-time voltage, current, and temperature, then DMS support requirements to the EPS will be significantly lower than in past spacecraft. Also, the local computing and data-storage capability of the EPS processor will minimize the requirements on the DMS processor in areas such as:

1) Archival data storage;

2) Fault diagnosis;

3) Health monitoring;

4) Operational state of commandable functions.

A normal function throughout the life of the space station will be software maintenance. The lack of software maintenance may be costly. Potential causes of software maintenance problems are:

1) Inadequate software documentation;

2) Temptation to save money by cutting corners on software documentation;

3) Inadequate test;

4) Inadequate quality control;

5) Inadequate sneak-path analysis;

6) Many potential interface pitfalls;

7) Many individuals will work on software over the life of the space station;

8) Configuration-control deficiencies.

The ability of the EPS to perform its function in the space station will be highly dependent on the software--not only the EPS applications programs, but also the computer executive routines. Maintenance and documentation of the computer applications and operating software will be just as significant as changes to EPS wiring.

4.8.3 EPS/Astronaut Interface

The EPS/Astronaut interface will consist of the information display about the EPS available to the crew via the onboard control-and-display subsystem, and the method for the crew to analyze and control the EPS. Display options range from a CRT to a dedicated meter for each parameter. Command input options range from a computerlike keyboard to a dedicated switch for each command. Other aspects of the EPS/Crew interface are:

1) Crew command authority;

2) Crew override;

3) Automatic validation of commands;

4) Quick-look problem assessment;

5) Crew training.

Design of the EPS/Crew interface has many wide-ranging impacts. The first requirement is that the crew be involved in not only the interface design but also the EPS design. Crew/EPS interface errors can cause loss of EPS functions or underuse. For a long-life space station, crew rotation is an operational necessity and crew training will be a continuing operation. Inadequately trained and certified crews can affect the EPS. Onboard ability to determine the EPS state of health quickly and precisely is, therefore, quite essential--especially on high-power systems.

As for the DMS, the extent to which the EPS is automated significantly affects the design and cost of the control and display subsystem.

4.8.4 EPS/Ground Interface

Almost everything said about the EPS/flight-crew interface applies to the EPS/ground-operations interface. Configuration control is a paramount ground activity that has the potential of getting out of control. Mistakes in configuration control could affect the EPS. During the life of the space station, there will be new flight-operations personnel every few months or years. Training and certification will be activities that can affect the EPS, if there are deficiencies. Both onboard and ground automation has a large effect on the cost of any ground-support equipment, actual mission operations, and documentation.

4.8.5 Modular Buildup

A space station will be built up in a modular fashion over a period of years. This implies adding new structures, modular EPS components, and new loads. As new equipment is brought online, there are many potential problems such as:

1) Interface compatibility;

2) Software growth;

3) Sneak paths (software and hardware);

4) Updating of performance capability.

4.8.6 Thermal-Dissipation Management

The amount of heat dissipated by the EPS components--in particular, power converters, inverters, and batteries--can exceed the design capability of the thermal-control subsystem. Inadequate temperature control or thermal-dissipation capability result in the following forms of EPS degradation:

1) Reduced bus-power capability;

2) Reduced battery life;

3) Reduced power-handling capability.

An ability to quickly and precisely assess thermal-control problems, determine solution approaches, and implement them is mandatory. Because user load and housekeeping-subsystem load control is involved, thermal-dissipation management and power management must be integrated. This is a system-level automation function that should be implemented by the space station's central computer.

5.0 TASK 3 - DEFINITION OF AUTOMATION TASKS

OBJECTIVE AND SCOPE

The first objective of this task is to develop a candidate list of automation activities that could minimize or eliminate the impact identified in Task 2 as well as from other activities that affect EPS performance. The second objective of this task is to create a generic-benefits list and identify the range of benefits available from each automation activity.

SUMMARY

It should be noted that there are basically two ways of automating any function or operation. One is to use hardwired logic and circuits containing discrete devices. The other is via a digital computer. This study is oriented toward automation of the second kind, and therefore, unless otherwise stated, this report generally implies use of a computer where automation is discussed.

Tasks that are generally suitable for automation are:

- Routine Tasks
- Precision Tasks
- Sequential and Timed Tasks
- Tasks That Must be Done on Compressed or Expanded Timeline
- Monitoring
- Memorization
- Complex Math or Logical Tasks

Table 5-1 presents the definition of the above general tasks.

Table 5-1 Definition of General Automation Tasks

Routine Tasks - Routine tasks by their nature are performed frequently in the same manner. As such, they are prone to generate errors by the astronauts or ground crew. By reducing astronaut and ground-crew interaction with the TPS by automating routine tasks, there is the potential to reduce workload and errors. Examples of routine tasks are battery-charge and -discharge control.
Precision Tasks - The benefits from automating precision tasks is to improve performance. An example of a precision task is solar-array pointing.
Sequential and Timed Tasks - A potential benefit of automating sequential and timed tasks is to eliminate errors. Common errors are to eliminate steps, perform steps out of sequence, or perform multiple steps. An example of sequential and timed tasks are load sequencing.
Tasks That Must Be Done on a Compressed Timeline - Tasks that must be done on a compressed timeline may cause an excessive workload for the astronaut or ground crew. The benefit from automating this class of task is to reduce workload. An example of a compressed timeline function is correction of a bus undervoltage.
Monitoring - A space station will have a large number of monitoring tasks. Routine monitoring may be considered a boring task that humans perform poorly. The benefits from automating monitoring tasks are a reduction in errors and crew boredom. Examples of monitoring tasks range from accounting for relay position, battery state of charge, and user load-status to doing limit checks such as for caution, warning, and alarm.
Memorization - A benefit from automating tasks requiring both short and long-term memory is task simplification. An example of a memorization task involving detailed knowledge of a component is checkout of an assembly.
Complex Math or Logical Tasks - Consider automating complex mathematical tasks to improve mission performance. An example of such a task is prediction of the time when a battery will become fully charged under varying load scenarios.

To standardize the definition of automation tasks, six categories of generic functions were identified as listed in Table 5-2.

Table 5-2 Automation Task Categories

1.0 Data Handling	3.0 Fault Handling
1.1 Acquisition	3.1 Fault Detection
1.2 Processing	3.2 Fault Isolation
1.3 Storage	3.3 Fault Correction
2.0 Monitoring	4.0 Control
2.1 Operational State	5.0 Planning and Operations
2.2 State of Health	6.0 Anomaly Handling
2.3 Performance Analysis	
2.4 Trend Analysis	

Data Handling - Data handling is required in all other automation tasks because they are dependent on input data. Data handling involves acquisition, processing, and storage of engineering data and commands. Data acquisition includes collection of measurements via multiplexing and analog-to-digital conversion to digitize the data to put it in a form acceptable for processing by digital computers. Processing involves all of the computational tasks. One of the processing tasks involved with data acquisition is conversion of the raw-ADC outputs to engineering units useful to the human users. Storage refers to storing of basic operating and application software as well as the storage of raw data and processed data.

Monitoring - Monitoring is defined to include operational state and state of health determination, performance analysis, and trend analysis. Operational state means the position of all switches, the good/bad status of all components, and the active/inactive status of all EPS components. State of health determination deals with determining if a particular EPS component is operating within its normal envelope. Thus, limit checking and built-in test and checkout are inherent subfunctions. If it is operating within its normal envelope, it is healthy. If it is operating outside its normal envelope, it may be impaired, unhealthy, or it may be in danger of an incipient failure.

Performance analysis deals with measurable indexes of performance, such as solar-array temperature or battery state of charge. Trend analysis involves the analysis of a variable as a function of time. Trend analysis may involve the analysis of one or many variables as functions of time.

Fault Handling - Fault handling includes the automation of fault detection, isolation, and correction. Faults may be true, false, or transient. An important goal of fault-detection automation is to minimize the number of false faults declared. The strategy for minimizing false or transient faults is to require a fault condition to exist for a time greater than a limit time. With hope, the limit time will be greater than the transient time. Fault isolation or safing consists of actions to remove the faulty component or isolate it from the EPS after a fault is declared. Fault correction requires analysis and action to correct the fault (switch in a standby redundant unit) or manage it if redundancy is not available, such as priority-load scheduling to reduce battery drain.

Control - This function is intended to include all routine housekeeping and maintenance tasks. Automation of control means mechanization of processes to effect the required results. An example of a frequent routine control task is the control of battery charge and discharge. An example of an infrequent control task is the determination of when to recondition a battery.

Planning and Operations - The planning and operations function involves all mission-operations activities. As a result, this is a space-station-level task. Automation of operations management will involve computer software to close the loop by monitoring the plans as they are implemented, evaluating performance, and taking corrective actions.

Anomaly Handling - Automation of anomaly handling is one of the more difficult and challenging tasks. An anomaly can be defined as an unforeseen situation or condition, a situation that is not understood, or a condition that can not be resolved by the existing measurements, hardware, or computer programs. One characteristic symptom is an occurrence of what appears to be a fault, but the fault is not repetitive and has no trend. Anomaly handling appears to be a candidate area for implementation via expert-system approach.

A general statement of benefits from EPS automation, which was developed for use in Task 3, is listed in Table 5-3.

Table 5-3 Benefits from EPS Automation

No.	Description
1	Increased Life
2	Increased Reliability, Maintainability, and Safety
3	Improved Performance
4	Reduce Cost
	4.1 Subassembly (Black Box)
	4.2 Subsystem
	4.3 Spacecraft
	4.4 Launch Operations
	4.5 Flight Operations
	4.6 Inflight Fault Detection, Maintenance, and Servicing
	4.7 Design, Development, Test, Evaluation (DDTE)
	4.8 Ground-Support Personnel Labor
	4.9 Ground-Support Equipment (Prelaunch & Flight Operations)
	4.10 C&DH Subsystem
	4.11 Thermal-Control Subsystem
	4.12 Life-Support Subsystem
	4.13 Crew Training Simulator/C&D Subsystem
5	Reduced Maintenance
6	Able to Overcome Technology Limitations
7	Reduced Astronaut/Power Subsystem Interaction
8	Reduced Number of Ground-Support Personnel
9	Reduced New-Subsystem Familiarization/Training Time
10	Reduced PV-Array Size and Weight
11	Reduced Battery Size and Weight
12	Reduced Power-Conditioning Size and Weight
13	Minimized Human Error
14	Allows Space Operation without Crew
15	Provides Real-Time Short-Response Control
16	Reduced Software and Hardware Interfaces to C&DH Subsystem
17	Improved Security and Survivability
18	Enables a Given Task, Operation, or Mission

This benefits list is a compilation of all automation-benefits lists from present and previous studies involving autonomy and automation. Note that the benefits can be grouped into one of the following action categories:

- Increase
- Improve
- Reduce
- Overcome
- Minimize
- Allow
- Provide
- Enable

Analysis of the benefits list in Table 5-3 shows that this list consists of a benefit category and a space station parameters column that is affected by automation. To provide more weight into the range of benefits potentially available from EPS automation, the space-station EPS parameter benefiting from automation is given as a function of the benefit action in Table 5-4.

Table 5-4
Benefit Action and Space Station Parameter Impacted by Automation

Action	Benefits
- Increase - Improve - Reduce	- Life, Reliability, Maintainability Safety - Performance, Security, Survivability - Cost - Maintenance - Astronaut/EPS Interaction - Number of Ground-Support Personnel - New Subsystem Training Time - PV Array Size and Weight - Battery Size and Weight - Power Conditioning Size and Weight
- Minimize	- Human Error
- Allow	- Operation without Crew
- Provide	- Real-Time Short Response Control

An inspection of Table 5-4 shows that the first result of automation is to increase, improve, allow, or provide for that which is desirable. Such space-station attributes as enhanced life and performance, ability to operate without a crew, and real-time short-response control capability are all needed. The effect of EPS automation is to enable these needs.

The second benefit of automation is to reduce or minimize undesirable characteristics. It is desirable to reduce or minimize cost, astronaut EPS interaction, size, weight, and human error. The effect of EPS automation is to reduce and minimize these undesirable EPS characteristics.

A matrix of benefits for each generic automation task is given in Table 5-5, and a brief summary of general approach to satisfy each automation goal is presented in Table 5-6.

Table 5-5 List of Benefits for Generic Automation Task

Automation Task	Benefits*																												
	1	2	3	4.1	4.2	4.3	4.4	4.5	4.6	4.7	4.8	4.9	4.10	4.11	4.12	4.13	5	6	7	8	9	10	11	12	13	14	15	16	17
1) Data Handling	x	x	x		x	x	x	x	x	x	x	x	x	x	x	x			x	x	x				x	x	x	x	x
2) Monitoring	x	x	x	x					x	x	x	x				x	x	x	x	x	x	x	x	x	x	x	x	x	x
3) Fault Handling	x	x	x					x	x		x	x					x	x	x	x	x	x	x	x	x	x	x	x	x
4) Control	x	x	x					x			x	x						x	x			x	x	x	x	x			
5) Planning and Operations	x	x	x																										
6) Anomaly Handling	x	x	x					x	x		x							x	x						x	x	x		x

*See Table 5 -3

Table 5-6 Benefits from EPS Automation

Automation Goal	General Approach
1.0 Increase life.	- Minimize stress on EPS during a normal operation and allow continuous operation in degraded mode.
2.0 Increase reliability, maintainability and safety.	- Detect, isolate, and correct faults quickly.
3.0 Improve performance.	- Operate EPS close to its limits, especially during degraded modes.
4.0 Reduce cost.	
4.1 Subassembly (black box).	- Replace number of discrete parts.
4.2 Subsystem.	- Do via software rather than hardware, wherever possible.
4.3 Spacecraft.	- Automate EPS to reduce other subsystem costs; automated test and checkout.
4.4 Launch operations.	- Automated test and checkout.
4.5 Flight operations.	- Reduce astronaut involvement in EPS monitoring and control, astronaut freed for other activities.
4.6 Inflight fault detection, maintenance, and servicing.	- Reduce astronaut, ground/EPS interaction.
4.7 DDTE (Design, Development, Test, Evaluation).	- Minimize design freeze via use of software.
4.8 Ground-support personnel labor.	- Automate EPS monitoring and control.
4.9 Ground-support equipment (prelaunch & flight operations).	- Onboard test and checkout, and fault handling reduce ground-support equipment.
4.10 Data-management subsystem.	- Reduce data and command interfaces due to EPS.
4.11 Thermal control subsystem	- Minimize thermal-dissipation management via EPS automation.
4.12 Life-support subsystem.	- Do integrated load control
4.13 Crew-training simulator and C&D subsystem.	
5.0 Reduce maintenance.	- Fault-handling automation will allow maintenance to be done on convenient schedule. Automatic monitoring functions and redundancy management.

Table 5-6 (cont)

6.0 Overcome technology limitations.	- Overcome limited component lifetimes by fault handling and redundancy.
7.0 Reduce astronaut/power subsystem interaction.	- Hardware and software automate fault handling, reducing astronaut-EPS interaction.
8.0 Reduce number of ground support personnel.	- Hardware and software automate fault handling, reducing need for ground support.
9.0 Reduce new subsystem familiarization/training time.	- Reduces penalty associated with operator mistake.
10.0 Reduce PV-array size and weight.	- Via automation, optimize use of available power, and road management.
11.0 Reduce Battery size and weight.	- (Same as above.)
12.0 Reduce power-conditioning size and weight.	- (Same as above.)
13.0 Minimize human error.	- Automate sequential, routine, boring tasks.
14.0 Allow space operation without crew.	- Automate EPS monitoring control, and fault handling functions.
15.0 Provide real-time short response control.	- Onboard hardware and software available in real time.
16.0 Reduce software/hardware interfaces to command and data management subsystems.	- Use digital-data interface and minimize analog-data interface; transmit processed engineering parameters (pwr, energy) and average quantities to minimize raw-data flow.
17.0 Improve security and survivability.	- Automation of fault handling provides continuous fault handling not interrupted by communications problems, operator error, or operator distracted to higher-priority task.

Table 5-6 (concl)

18.0 Enable: a) Mission.	- Reduction of array and battery weight through EPS and load management, enables certain missions to use photovoltaic system.
b) Autonomous operation.	- Automation of all critical monitoring and control tasks previously done on ground.

5.1 FAULT-HANDLING TASKS

The automation tasks identified in Table 5-2 and the benefits list identified in Table 5-3 were used to analyze the faults and activities identified in Task 2. The approach taken was to identify the automation function required to resolve or permit a workaround solution for each of the failure modes identified for each selected EPS component. The results of this analysis are shown in Tables 5.1-1 thru 5.1-19 at the end of this chapter.

5.2 MONITORING TASKS

Monitoring tasks consists of (1) operational state determination, (2) state-of-health determination, and (3) performance and trend analysis. Self-test and checkout are included under state of health. Table 5.2-1 (at the end of this chapter) is a list of specific subtasks identified for the photovoltaic/battery power subsystem.

5.3 CONTROL TASKS

All routine control functions are included in this category. Table 5.3-1 (at the end of this chapter) lists specific examples for several subsystem components.

5.4 PLANNING AND OPERATIONS TASKS

Planning and operations tasks involve all activities required by the space station, flight crew, and/or the ground crew to satisfy the mission-operations requirements. The principal task identified is that of electrical-consumables management or simply energy management. This is a system-level task because it affects not only various housekeeping subsystem functions but also the operational sequence of experiments.

The energy management goals are to:

1) Provide the required power under normal and degraded mission modes;

2) Maintain a positive average bus power margin;

3) Extend battery life and minimize battery maintenance.

It is further intended that the above goals should be fully automated with lesser autonomy initially, but growing into a fully autonomous onorbit capability. Achieving these goals will provide benefits such as reducing ground labor and equipment costs, improving flight crew and ground-crew productivity, and allowing complex, concurrent operations with minimal human error.

Table 5.1-1
Solar Array Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Task*	Method	Benefits**
Lose Power from Part of Array Fail to Track Sun Degraded Ability to Track Sun Plasma Interaction Long Term Degradation Excessive Charged Particle Degradation.	1, 2, 3 1, 2, 3 1, 2, 3 1, 2 1, 2 1, 2	1) Determine status of all subarrays via limit checks, and identify failed or degraded subarrays. 2) Determine total array power available. 3) Calculate total array power degradation. 4) Determine impact on bus load-handling capability. 5) Maintain state-of-health and performance trend data. 6) Isolate failed subarrays.	4.5, 4.6, 4.8, 7, 8, 10

*See Table 5.1-2.
**See Table 5.1-3.

Table 5.1-2 Gimbals Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Slip Ring Short, or Roll Ring-Twist Flex Open, or Degradation	1, 2, 3	Periodically calculate P(IN) & P(OUT). Archive data, trend-analysis projections. Pinpoint failure.	4.6, 4.8, 7, 8, 15
Rotary Transformer Fail or Degrade	1, 2, 3	Same as above plus under-voltage management, redundancy switching.	

Table 5.1-3
Dc/Dc Converter, P^3 Type Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Shorted Series Pass Transistor	1, 2, 3	Detect overvoltage and close shunt switch.	1, 2, 4.5, 4.6, 6, 7, 15
Low V_{OUT}	1, 2, 3	Sense V_{OUT}. When valid undervoltage, prior and load sheet and bus test. Determine P3 good/bad. Determine V_{IN} good/bad. If P3 bad, switch-in backup, priority load connect. If P3 good, source overloaded, limit loads reconnected.	
Efficiency Below Acceptable	1, 2, 3	Switch backup online, use low-efficiency one as standby.	3, 5, 7, 8
V_{IN} High	1, 2, 3	Monitor V_{IN}. P3 shut-down on V_{IN} H1. Shift loads to another P3, or add loads to one with H2 V_{IN}.	2, 6, 7, 15
I_{IN} High	1, 2, 3	Priority load shed, then if still failed, switch off and bring on backup.	
High Internal Temp	1, 2, 3	Monitor temps, shut down on overtemp. Bring back up online. Priority load add.	
I_{OUT} Overload	1, 2, 3	Monitor I_{OUT}, compare to limit, support for programmed time, turn off pause, restart.	

Table 5.1-4
Battery Charger (P^3) Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Failure Mode, Batt V, I, or T Overlimit	1, 2, 3	Monitor, limit check, reduce charge V&I; if still over limit, turn off.	2, 4.6, 4.8, 7, 8, 13
Battery-Charger Mode, Solar-Array Voltage Collapse	1, 2, 3	Sense V across series-pass transistor & when less than limit, turn P3 off, pause until solar array recovers, then restart.	2, 6, 7, 15
Piece-Part Failure in Stabilization Circuit, or Output Filter Cap Open-Useable, But Increased Ripple Voltage	1, 2, 3	Onboard computer analysis of time response, compare spectrum to nominal, detect failure, use this one as standby.	4, 6, 4.8, 7

Table 5.1-5
Transformer Coupled Converter Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Output Over/ Under Voltage	1, 2, 3	Output V sense, limit check for undervoltage, hardware over V detect & shunt trip, priority load removal, reapply, switch backup on line.	6, 7, 15
Low Efficiency	1, 2, 3	Periodically calculate efficiency, switch low unit to backup status.	6, 7, 15
Input V, I, T Out of Limit	1, 2, 3	Monitor, limit check, turn off for out of limit, bring back up online.	6, 7, 15

Table 5.1-6
Series Resonant Inverter (Dc/Ac) Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Input Cap Overvoltage	1, 2, 3	Input cap over V detect & shutdown. Bring back up online & priority connect loads.	2, 4, 6, 4.8, 7
Output Over Undervoltage	1, 2, 3	Monitor & limit check output voltage, turn off on over V, on under V priority remove loads, find failure in SRI or source, start backup and priority load connect.	2, 4, 4.6, 7
Input Fuse Open	1, 2, 3	Monitor fuse status; if bad,start back up, alert higher levels that this SRI is bad.	

Table 5.1-7
Solar Array Voltage Controller Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Discrete Switch Failure to Operate	1, 2, 3	Direct Monitor, extra set of contracts, indirect monitor, I & V.	2, 3, 4.6, 4.8, 7, 15
Solar Array Battery Share Mode	1, 2, 3	Monitor solar array V & Bat I during sun. If Bat is discharging when it should be charging, remove loads on priority basis to allow array to recover, or use boost conv to raise array V.	
Control Electronics Failure Causes Solar Oscillations	1, 2, 3	Compare measured to theoretical solar bus power, or use spectrum of bus V, unwanted harmonics mean a failure.	
Closed Loop Controller Failure	1, 2, 3	Monitor error signal, saturated error signal means failure, switch-on backup unit.	

Table 5.1-8
NiCd and NiH_2 Batteries Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Low Discharge Voltage (DV) - Cell or Module	1, 2, 3	1) Compare the EODV with average EODV all other cells or modules (EODV) within one battery string. 2) Reestablish EODV caution, warning, and alarm limits based on trend data. 3) When alarm limit is reached, and EODV limit, try load shedding during each successive discharge period, increasing the amount of load power removed as the EODV decreases.	1, 2, 4.8, 7, 8, 11
- Battery	1, 2, 3	1) Compare the EODV with those of other batteries (EODV). 2) Same as 2 above. 3) Same as 3 above.	
Cell Short or Open	1, 2, 3	1) Monitor individual cell voltages and verify shorted cell (check charge, discharge, and open-circuit voltages of cells and battery). 2) Bypass shorted cell; replace with spare cell following charge equalization procedure.	6
Cell Voltage Reversal during Discharge.	1, 2, 3	1) If reverse voltage alarm limit: - Bypass that cell, and/or - Reduce load on battery or - Remove battery until DV is positive.	1, 2, 4.8, 7, 8, 11

Table 5.1-8 (cont)

Failure Mode	Automation Candidate	Method	Benefits
Cell Under-pressure during Charge or Discharge, or Low Battery Capacity	1, 2, 3	1) Determine if cell(s) has partial short; compare with other cells for excessive unbalance in pressure. 2) Determine if battery was excessively discharged or undercharged in previous cycle. 3) If sufficient recharge power is available, increase the RF by 0.03 in subsequent cycles; monitor battery EODV and average end of discharge pressure (EODP). 5) If EODV and/or EODP do not increase in each cycle, reduce battery load and/or remove battery during each eclipse period, and continue until EODV and EODP have attained normal values. 2) During subsequent charge/discharge cycles: - Increase recharge fraction (RF), - Reduce load on battery or - Remove battery during eclipse periods. 3) Determine Goodness/Badness of cell by comparison with other cell performance.	1, 2, 11
Cell Overpressure during Charge	1, 2, 3	1) Determine if cell is being severely overcharged (check RF, cell temperature, charge-voltage limits).	1, 2, 4, 8, 7, 11, 15

Table 5.1-8 (concl)

Failure Mode	Automation Candidate	Method	Benefits
		2) Reduce charge current or charge voltage, or remove battery. 3) Check for excessive unbalance in pressure relative to other cells in battery.	
Excessive Battery Temperature	1, 2, 3	1) Determine cause(s) of excessive temperature. - Excessive overcharging - Excessive discharge rate or DOD - Thermal-control failure - Spacecraft orientation so the battery is exposed to sunlight. 2) If it is due to excessive overcharging, reduce RF or charge rate; if caused by excessive discharge rate, thermal-control failure, or spacecraft orientation, reduce battery load; continue until it attains normal temperature.	1, 2, 4, 8, 7, 11, 15
High Charge Voltage (CV)		1) Determine Cause(s) of High CV: - Charge controller failure (to clamp voltage) - Temperature sensor failure 2) Reduce battery current by array section switching.	1, 2, 11, 15

Table 5.1-9
Housekeeping Supplies Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Voltage Current Hi/Lo Out of Limit	1, 2, 3	Direct monitor, limit check, switch to back up if available, report status.	2

Table 5.1-10
Magnetic Latching Relay Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Failure to Transfer, Spurious Transfer, Relay Driver Fails, or Contacts Open or Welded Shut	1, 2, 3	Verify command executed by direct and indirect determination of relay position. Automatic reentry of a failed command. Periodically compare relay commands to position, and report differences.	2, 4, 6, 4.8, 8
Relay Oscillates	1, 2, 3	Look for measure of output, amplitude harmonics.	2, 4, 6, 4.8, 8

Table 5.1-11
Motor Driven Switch Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Fail to Transfer, or Spurious Transfer	1, 2, 3	Command verification & periodic position monitoring (see Mag Latch Relays).	2, 4.8, 4

Table 5.1-12
Remote Power Controller Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Fail to Transfer, Spurious Transfer, Relay Driver Fails, Contacts Open or Welded Shut. Thermal Failure Causes RPC Cold Plate Temp to Increase	1, 2, 3	Verify command executed by direct and indirect method. Automatic reentry of a failed command, report a failed command. Periodically compare relay commands to position and report differences.	2, 4.6, 4.8, 7, 8
RPC Oscillates or Fails to Limit Rise of Current	1, 2, 3	Measure spectrum of outlook,look for high-amplitude harmonics.	
Fail to Limit Current Fall (-di/dt)		Same as above. This can work for small inductance. For large inductance, RPC destroyed after failure.	
RPC 3 Second Timer Fails. RPC Carries Fault Current until RPC Internal Fuse Opens		Computer timer monitors fault current and trip indicator on RPC. When fault clear-time exceeds RPC carry time and no trip indicator, report as failed or anomalous RPC.	

Table 5.1-13 Fuses Failure Modes, Automation Candidates, and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Open	1, 2, 3	Determine fuse state good/bad direct or indirect. Direct determination by blown fuse indicator, indirect by input, output current & voltage sensors. Periodically monitor and report status. Store time when failure first detected.	2, 4.6, 4.8, 7, 8

Table 5.1-14
Circuit Breakers Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Assumed Manual Breaker. Open When Should Be Closed, or Closed When It Should Open	1, 2, 3	Direct or indirect position measurement. Periodically compare manual command table to measured position, store time of change and report status.	2, 4.6, 4.8, 7, 8

Table 5.1-15 Cabling Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Cable Opens, Insulation Shorts Wire-to-Wire or Wire-to-Return	1, 2, 3	Monitor source loads, load switching	2, 4.6, 4.8, 7, 8, 13, 14, 15, 17
Insulation Degrades due to Overtemperature in Cable	1, 2, 3	Monitor cable temp sensors & limit check. Report status to next computer. Higher level to shed loads on priority basis to decrease cable temps, or decide to tolerate on a limited, monitored basis. Higher-level decision required.	1, 2, 4.8, 7, 8, 15, 17
Thermal Subsystem Failure	1, 2, 3	Same as above.	(Same as above)
Modular buildup or Attitude-Control Mode	1, 2, 3	Same as above. Resource protection automated. System fault may require human involvement for correction.	(Same as above)

Table 5.1-16
Sensors and Signal Conversion Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Catastrophic Failure	1, 2, 3	Limit checks, compare to redundant unit, check state of user, periodically report status.	1, 2, 4.6, 4.8, 7, 8
Drift	1, 2	Compare redundant units, sum V, I, P & check deltas from zero, trend analysis, periodically report status.	
Out of Calibration	1, 2	No practical method now (desirable to develop).	1, 2, 6
Antialiasing Filter or ADC Ground Open	1, 2, 3	Inject reference signal with harmonics into filter and ADC. Observe several samples, if good, all ADC outputs within limits. Report status.	1, 2, 4.6, 4.8, 7, 8

Table 5.1-17
$LiSOCl_2$ Battery Failure Modes, Automation Candidates, and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Fail While Not Operating, Open-Shorted	1, 2	Monitor bat. & cell V, periodic short-term loading to verify operational & prove backup capability exists, trend analysis.	1, 2, 4.6, 4.8, 7, 8
System Failure or False Emergency Causes Battery To Be Put Online	1, 2, 3	Monitor AH out & report probably time battery will last at present date of discharge. Also, output time bat. would last at other rates of discharge, store all removed because when emergency over, bat. fault-management capacity will be lowered.	1, 2, 4.6, 4.8, 7, 8

Table 5.1-18
Chemical Turbomachinery Failure Modes, Automation Candidates, and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Fail While Not Operating	1, 2	Monitor reactant pressure, amount remaining, critical temps, periodic short-term operation to verify backup to capability trend analysis, report status.	1, 2, 4.6, 4.8, 7, 8
Some Failure Causes Component to Turn on & Supply Power	1, 2, 3	Monitor rate of reactant use & printout of time remaining at several different rates. Store consumables data because when use over, fault management capability will be lowered.	1, 2, 4.6, 4.8, 7, 8

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Table 5.1-19
Otehr Subsystems and Activities Failure Modes, Automation Candidates, and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Flexible Structures and Control, Oscillations	Not a Candidate		
Data System Degraded; Data Rates; CPU	1, 2, 3,	Automatic scaledown of EPS computation, shift high-level automation to ground.	1, 2, 4.6, 4.8, 7, 8
EPS/Crew/Gnd Interface - Invalid Commands	1, 2, 3, 6	Real-time validation of all commands, prompting of crew on consequences overriding auto function.	1, 2, 4.6, 4.8, 7, 8
- Inadequate Training	1, 2	Computerized training, configuration update, prompting by computer.	4, 13, 9
Activity, Software Maintenance	1, 2	Specialized software tools.	9, 13
Thermal Control Can Not Maintain EPS Temperatures	1, 2, 3	Integrated design of high-level control of thermal & EPS required.	1, 2, 4.6, 4.8, 7, 8
User Loads, Open, Short or Changed Impedance	1, 2, 3	Periodically calculates Z, limit check, output status, & trend. For additional SOH information, take time-response of V&I. Extract spectrum. Compare spectrum and time response to nominals stored in computer.	1, 2, 4.6, 4.8, 7, 8

Table 5.2-1 Monitoring Task Examples

Operational State Determination
- Number and Identity of Components Online, Offline, or Failed Relay Position and Command State
 	
State of Health
- Solar Array, Batteries, Power Conditioning, Bias (Housekeeping) Power Supplies
- Built-in Test and Checkout (Limit Checks)
 	
Performance and Trend Analyses
- Solar Array
  - Normalized Peak Power (NPP); Available Average Power/Daytime vs Orbit Number
  - NPP and I_{SC} Degradation
  - Minimum, Average, and Maximum Temperature
- Batteries
  - SOC, DOD, EODV, and EOCV Limit vs Orbit Number
  - Average Temperature during Charge and Discharge vs Orbit Number
  - Total Number of Cycles above X% DOD, Y% DOD
  - Number of Cycles Since Last Reconditioning
  - Battery Recharge Fraction vs Orbit Number
- Bus Power Capability (Orbital Average, Average Power Margin)
- Bus Load (Day, Night, and Orbit Average)
 	
- Converters and Inverters
  - Efficiency
  - Output Impedance
 	
- Load Equipment
  - Input Impedance

Table 5.3-1 Control Task Examples

Solar Array - Orientation Control - Voltage Regulation Batteries - Charge and Discharge Control - Spare Module or Cell Management - Reconditioning - Redundancy Converters - Loadsharing Control - Redundancy Management Imbedded Controller (e.g., P^3 Converter): - Mode Control (Voltage Regulator or Battery Charger) - Internal Fault Detection and Isolation - Overload Handling - Output-Voltage Programming

6.0 TASK 4 - PARTITIONING OF AUTOMATION FUNCTIONS

OBJECTIVES AND SCOPE

The objectives of Task 4 were to develop a method for partitioning the automation activities between the EPS, Space Station System, and the ground, and to partition all EPS-automation candidates developed in Task 3.

SUMMARY

The partitioning method used was as follows. First, the time criticality of the function is determined. From this analysis, functions can be separated into (1) time-critical functions that require dedicated hardware, such as bus overvoltage, and (2) functions that do not require the fast response time and are candidates to be performed by a computer. Next, the location where the task is to be performed and the resources to do the task are identified. A determination is then made of the external interface impacts--Are the impacts totally within the EPS? Or are these impacts outside the EPS? General criteria is established for partitioning the automation functions are as follows:

- Dedicated hardware are to be located in the EPS component;

- Fault detection, isolation, and correction can be partitioned to different levels;

- To be partitioned to the EPS, the fault must originate in the EPS; the correction resources should be in the EPS; and there should be no impacts outside the EPS.

Finally, the last step consists of considering each function partitioned to the EPS, the space station system, and the ground, and providing rationale for or against each partitioning. Partitioning can be facilitated in terms of where sensing, analyzing, and acting should best be performed.

6.1 GENERAL METHOD

6.1.1 Fault-Handling Partitioning of Tasks

The methodology for partitioning is firmly grounded in an analysis of the time criticality of the fault, a partitioning of the automation task between hardware or software based on the time criticality, an identification of where the fault is defined and where the correction resources are, and an identification of the external impacts of the fault. One of the study ground rules was that the partitioning would be to the EPS, Space Station System, or to the ground. General partitioning criteria were developed. Each specific fault was considered partitioned to each of the three areas, EPS System, and ground, and recommendations and rationale for each particular partitioning were given. It was considered just as significant to give rationale for not partitioning a function to one area as it was to provide rationale for partitioning a function to the area of optimal benefit.

The following sections present the detail steps in the automation-partitioning method.

Identify Fault - The first step in the partitioning process is to identify the fault being studied. The fault is primarily identified by EPS assembly and the specific fault. A further identification of the fault can be made in terms of its operational impact identified in Task 2.

Time Criticality - Time criticality is defined as the length of time between a fault occurrence and when the fault impact will be experienced by the Space Station if the fault is not safed and corrected. The smaller the time interval between a fault occurrence and the impact, the more time-critical is the fault. The time interval can be identified in units of milliseconds, seconds, minutes, fractions of an orbit, or multiples of the orbit period. The time criticality is specified by the time duration between fault occurrence and impact onset and a gross evaluation of YES/NO for time criticality.

The first use made of time criticality is to identify those faults that are so fast that they require hardware for sensing, safing, and correction as opposed to faults that are slower and could be handled by software. A second use made of time criticality is to aid in partitioning and assigning a priority to fault handling in the event of simultaneous fault.

Hardware/Software Partitioning - Time criticality is used to separate those faults that require dedicated hardware for handling from those slower faults that could be done by software. Additionally, there can be hierarchy of protection levels. For example, say the maximum temperature in an assembly is not to exceed 80°C. Software could be used to monitor a temperature transducer and shut the assembly down if the temperature exceeded 74 $\pm$2°C. Functional redundancy could be provided by a bimetallic switch that would disable and protect the assembly if the temperature were 78 $\pm$2°C. In this case, a hardware backup was provided for a primary software system.

Fault Definition Level - An identification must be made of where in the Space-Station functional architecture the fault can be defined. The lowest identifiable failure level may not be the same as the lowest replaceable level. For example, battery cells will be packaged in modules. The lowest identifiable failure level will be the cell level, but the lowest replaceable level is the module.

Exactly where the lowest identifiable fault-definition level and replacement level will be is not known now because they will be functions of packaging and how much redundancy is built into each black box. If the choice is made for block redundancy at the black-box level, then the lowest identifiable and the replacement levels will be the same. If the decision is made to package standby redundant elements in each black box, then the lowest identifiable fault level will be below the black-box level.

For purposes of this study, faults will be defined at the following levels:

- Lowest Identifiable Level
- Lowest Replaceable Level
- EPS Level
- Space-Station-System Level

As previously stated, it is not known now where the lowest identifiable level will be, but is important to identify this level for input to the fault-correction process. Examples of black-box-level faults are fuse failure, relay failure, RPC failure, nonredundant power converter package, or battery-module failure. Some of the more complex failures will be defined at the EPS level. Examples of EPS level faults are a fail to charge batteries due to a solar-array voltage collapse, or a user bus-undervoltage due to a power-converter or power-source failure. Both of these examples would require EPS-level information to detect, analyze, and correct. Faults defined at the Space Station System are those faults that have systemwide impacts as to require system information to define and correct. Examples of system faults are a thermal-subsystem failure that limits the amount of waste heat that can be removed from the EPS, or oscillations in the flexible structure that affect solar-array pointing. Both of these failures will have systemwide impacts and would require system-level information to detect and correct.

<u>Identify Level-of-Correction Resources</u> - It is important to identify where the correction resources are to help in the partitioning process. For purposes of this study, correction resources are identified at the following levels:

1) Lowest Identifiable Level;

2) EPS Level;

3) Space Station System Level.

The partitioning process is aided by this resource-level identification. If the correction resources are in the EPS, then the decision-making authority may be at the EPS level. If the correction resources are not at the EPS level, but at the Space Station System level, then it is that the decision making authority can not be concentrated at the EPS level. Decisions of the Space Station System level will be required.

Identify External Impacts - The purpose of this step is to classify the faults into two impact categories:

1) No impact outside EPS;

2) Impact outside EPS.

Impacts outside the EPS can, of course, be broken down into various other categories such as operating-schedule changes, safety-margin impacts, spacecraft-operating-mode impacts, or payload impacts. For purposes of this study, it was deemed sufficient to use two categories, (1) no impact outside EPS, and (2) impact outside the EPS.

External-impact assessment will be use as an aid in the partitioning process. Faults that do not have an impact outside the EPS are candidates for handling at the EPS level. If the fault has an impact outside the EPS, then it is likely some decisionmaking authority will have to be assigned to the Space Station System.

Partitioning Ground Rules - The ground rules for partitioning the automation functions were established by MSFC. The automations functions will be partitioned among the following three areas:

1) EPS;

2) Space Station System (Central Computer-assumed);

3) Ground.

The above three areas are the lowest level of detail for functions to be partitioned. For example, if a function is partitioned to the EPS, we will not try to assign it to a distributed- or a central-EPS processor. Further, if a function is partitioned to the ground, we shall not try to assign it to a flight-operations or flight-support center. Also, we will not affect, make any assumptions about, or drive the computer architecture with any of the partitioning activities.

Criteria for Partitioning - The following is a discussion of general criteria for partitioning that were developed. All of the criteria are obtained by application of conservative engineering judgment to the material developed in the previous steps.

Time-critical-hardware functions should be done in the EPS. If a function is time-critical and requires dedicated hardware to perform, then the hardware can not be put on the ground, but must be onboard the spacecraft.

Functions that can be performed by either hardware or software should be analyzed further to point out the advantages and disadvantages of a hardware or software implementation. The overriding reason for partitioning a function to hardware is time criticality. Reasons for assigning functions to software are:

1) Flexibility;

2) Reprogrammable;

3) Fast response to changing or unforeseen mission requirements.

A reason for assigning a protection function to both hardware and software is to achieve functional redundancy. If there were to be a major failure in one area, say computers, then the functionally redundant hardware-implemented protection systems could still function independent of the computer.

Software functions can be partitioned to the EPS, Space Station system, or the ground.

Fault detection, safing, and correction do not all have to be partitioned to the same area. Similarly, the functions of sensing, acting, and analyzing can be partitioned to different areas. The more likely scenario is that the sense and act functions (signal transducers and control effectors) will be in the EPS. The analysis and decisionmaking authority can be shared among the EPS, system, and ground.

For partitioning to the EPS, the following should be true:

1) The fault should be defined in the EPS;

2) The correction resources should be in the EPS;

3) No impacts outside the EPS.

Even though a particular function is partitioned to the EPS, there can be enables or concurrence to proceed from either the Space Station System level, the flight crew and the ground, or combinations of the levels.

For partitioning to the Space Station System, one or more of the following should be true:

1) The fault is not defined in the EPS;

2) The correction resources are not in the EPS;

3) There are impacts outside the EPS.

Again, even though a function is partitioned to the Space Station system level, there can be enables or concurrences to proceed from the flight crew and/or ground.

The following are some criteria for partitioning functions to the ground. Functions that can not or should not be automated on board should be partitioned to the ground. Faults having an expected occurrence so low as to not be cost effective in automating their handling onboard could be partitioned to the ground.

Activities so complex or beyond the state of the art for automation onboard the Space Station are candidates for partitioning to the ground.

6.1.2 Partitioning Other Automation Tasks

Any functional operation can be separated into three activities:

1) Sense: Acquire data or information needed;

2) Analyze:
   - Process raw data to generate desired parameters (e.g., power, energy, etc);
   - Analyze data to determine a problem or failure;
   - If a problem or failure is indicated, determine a solution approach;
   - Direct the electronics that actually implement the task, issue command.

3) Act: Do the function requested, implement the command received (e.g., activation of a switch).

Sensing involves signal transducers, multiplexing, and signal conversion. Analyzing involves converting raw ADC outputs to engineering units, analysis of the data to determine the fault, no fault status, determination of a solution if a failure is indicated, and the issuing of corrective-action commands. Acting involves the effectors such as relays or digital-to-analog converters. The acting activity implements the command received from the analysis function.

For the non-fault-handing functions, the three activities of sense, analyze, and act will be partitioned among the EPS, system, and ground. Rationale for the partitioning will be given.

6.2 RESULTS OF FAULT-HANDLING AUTOMATION PARTITIONING

The results of partitioning the fault-handling automation between the EPS, space station system, and the ground is shown in Tables 6.2-1 thru 6.2-15.

The partitioning of automation functions in this task was performed without reference to the level of autonomy of the Space Station. The object was to identify the characteristics of the fault and to perform the partitioning based on identified fault characteristics.

Faults that require a fast detect-and-safe time (milliseconds) and dedicated hardware (not computers) such as a dc/dc converter output over voltage, must of necessity have the machine-autonomy automation placed in the EPS. The fast reaction time makes it impossible to perform the automation of the space station system or ground level.

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Table 6.2-1 Solar Array Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach		Fault					
Fault	Time Criticality	Hardware	Software	Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Loss of Power from Part of Array; Excessive Power, Degradation	Minutes	No	Yes	EPS	EPS	Yes	(1),(2),(3),(4),(5)	(10),(11)	(10),(12)
Failure of Array to Track Sun	Minutes	No	Yes	EPS	System/ACS	Yes	(1),(2),(3),(4),(5)	(7),(10),(11)	(10),(12)
Arcing on Array from Plasma Interaction or Corona	Minutes	No	Possible	EPS	System	No	(1),(2),(3),(4),(5)	--	(10),(12)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Table 6.2-2
NiCd and NiH_2 Battery Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach		Fault					
Fault	Time Criticality	Hardware	Software	Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Low Discharge Voltage or Low Capacity	Minutes to Hours	No	Yes	EPS, Cell or Module	EPS	Yes	(1),(2),(3),(4),(5)	(10),(11)	(12),(13)
Cell or Battery Open or Short	Seconds to Hours	No	Yes	EPS	EPS	No	(1),(2),(3),(4),(5),(6)	(10),(11)	(12),(13)
Cell Voltage Reversal during Discharge	Seconds	No	Yes	EPS	EPS	Yes	(1),(2),(3)	--	(12)
Excessive Cell Pressure during Charge	Seconds	No	Yes	EPS	EPS	No	(1),(2),(3),(8)	--	(12)
Battery Temp High or Low	Seconds to Minutes	No	Yes	EPS	EPS	Yes	(1),(2),(3),(4),(5),(8)	(10),(11)	(12)
High Charge Voltage	Seconds to Minutes	No	Yes	EPS	EPS	No	(1),(2),(3),(4),(5),(8)		(12)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Table 6.2-3
Regenerative Fuel Cell Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Fuel Cell Module Failure - Low Voltage - High Internal Resistance - Open or Short - Cell Voltage Reversal - Pump - Reactant Leakage	Seconds	No	Yes	EPS	EPS	Yes	(1),(2),(3), (4),(5)	(10),(11)	(10),(12), (13)
Electrolysis Module Failure - Pump - Cell Open or Short	Minutes	No	Yes	EPS	EPS	Yes	(1),(2),(3), (4),(5)	(10),(11)	(10),(12), (13)
Reactant Subsystem - Leakage - Pump	Minutes	No	Yes	System	System	Yes	(1),(2),(3), (4),(5)	(10),(11)	(10)
Electrolysis Regulator	Minutes	No	Yes	EPS	EPS	No	(1),(2),(3), (4),(5),(8)	(10),(11)	(13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Table 6.2-4
Solar Array Voltage Controller Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Partial Loss of Power or Control	Minutes to Hours	No	Yes	EPS	EPS	No	(1),(2),(3), (4)	(10),(11)	(10),(12), (13)
Full Shunt Fail Short (No Power)	Minutes	No	Yes	EPS	EPS	No, If Corrected, Yes, If Not Corrected	(1),(2),(3), (4)	(10),(11)	(10),(12) (13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

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Table 6.2-5

P^3 (dc-dc Converter) Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Output Over Voltage	Milli-second	Yes	No	EPS	EPS	No	(1),(2),(3)	(11)	(12)
Output Under Voltage	Milli-seconds to Seconds	No	Yes	EPS	EPS	No	(1),(2),(3), (4)	(1),(8), (10),(11)	(12),(13)
Efficiency Low	Minutes to Hours	No	Yes	EPS	EPS	No	(1),(2),(3)	(11)	(12),(13)
Out of Limit: V(In), I(In) Temp	Seconds to Minutes	No	Yes	EPS	EPS	No	(1),(2),(3), (8)	(11)	(12),(13)
Thermal Control Failure	Minutes	No	Yes	System	System	Yes	(1),(2)	(8),(10)	(13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Table 6.2-6

Transformer Coupled Converter Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
V(Out) High	Fraction of Sec to Secs	Probably Not. Slower Failure Than Non Trans-former Coupled Con-verter	Yes	EPS	EPS	No, If There Is Block Redun-dancy for Cor-rection	(1),(2),(3)	(11)	(12),(13)
No Output	Seconds to Minutes	No	Yes	EPS	EPS	Yes, If No Redun-dancy	(1),(2),(3), (4)	(10),(11)	(10),(12), (13)
Efficiency Degraded	Hours to Months	No	Yes	EPS	EPS	No	(1),(2),(3), (4)	(11)	(12),(13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Table 6.2-7

Series Resonant Inverter Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach		Fault					
Fault	Time Criticality	Hardware	Software	Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Resonant Capacitor Over Voltage	Milli-seconds	Yes	No	EPS	EPS	No	(1),(2),(3)	(11)	(12),(13)
Output Over Voltage	Milli-seconds	Yes	Back Up to Hard-ware	EPS	EPS	No	(1),(2),(3)	(10),(11)	(12),(13)
Input Fuse Open	Seconds	No	Yes	EPS	EPS	No	(1),(2),(3)	(11)	
No Output	Seconds to Minutes	No	Yes	EPS	EPS	Yes, If No Redun-dancy	(1),(2),(3),	(10),(11)	(12),(13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Table 6.2-8

Magnetic Latching Relay, RPC, and Motor Driven Switch Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach		Fault					
Fault	Time Criticality	Hardware	Software	Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Fail to Transfer, Spurious Transfer (Command Verification)	Seconds to Minutes	No	Yes	EPS	EPS	Yes	(1),(2),(3)	(11)	(12),(13)
Output Oscillates	Minutes to Hours	No	Yes	EPS	EPS	Yes	(1),(2),(3)	(11)	(12),(13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

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Table 6.2-9
Relay Configuration Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Lose Redundancy, Operate Normal	No	No	Yes	EPS	EPS	No	(1),(2),(3)	(10),(11)	(12),(13)
Single Relay Fail Open, 2 Series Relays, One Fail Open (Load Can Not Be Connected)	No	No	Yes	EPS	System	Yes	(1),(2),(3)	(10),(11)	(12),(13)
Single Relay Fail Closed, 2 Parallel Relays One Fail Closed (Load Can Not Be Removed)	No	No	Yes	EPS	System	Yes	(1),(2),(3)	(10),(11)	(12),(13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Table 6.2-10
Remote Power Controller Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Fail to Limit di/dt	Milli-seconds	No	Analysis by Software	EPS	System	Yes	(1),(2),(3)	(10),(11)	(12),(13)
RPC 3-sec Timer Fails; RPC Fails to Clear Fault Current	Seconds	No	Yes	EPS	EPS	Yes	(1),(2),(3)	(10),(11)	(12),(13)
Thermal Control Failure Causes RPC Cold Plate Temp to Approach Limit	Seconds to Minutes	No	Yes	System	System	Yes	(1),(2),(3)	(10),(11)	(12),(13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

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Table 6.2-11
Fuse Configuration Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Single Fuse Open; Series Fuses One Open; No Power Can Be Applied to a Load	No	No	Yes	Fuse	System	Yes	(1),(2),(3)	(10),(11)	(12),(13)
Two Parallel Fuses, One Open	No	No	Yes	Fuse	System	No	(1),(2),(3)	(10),(11)	(12),(13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Table 6.2-12 Cabling Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
High Temp in Cable	Minutes	No	Yes	EPS	System	Yes	(1),(2),(3)	(10),(11)	(12)
Insulation Shorts Wire to Wire or to Return	Seconds to Minutes	No	Yes	EPS	EPS	Yes	(1),(2),(3)	(10),(11)	(12)
Modular Buildup Activity Impacts Cables (Overloads or Overtemps)	Minutes	No	Yes	EPS	System	Yes	(1),(2),(3)	(10),(11)	(12),(13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Table 6.2-13 Gimbal Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Slip Ring Noise	Hours	No	Yes	EPS	System	No	(1),(2),(3)		(12)
Slip Ring Short, Roll Rings Open, Twist Flex Open, Rotary Transformer Open	Minutes	No	Yes	EPS	EPS, If Block Redundant; System, If No Block Redundancy	Yes	(1),(2),(3)	(11)	(12),(13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Table 6.2-14 Sensor Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Catastrophic Failure, Drift, Antialiasing Filter or ADC Ground Open	Minutes	No	Yes	EPS	EPS	No	(1),(2),(3)	(11)	(12),(13)
Out of Calibration	Days	No	Yes	EPS	System	Yes	--	--	(1)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Table 6.2-15
Auxiliary Power Unit Failure Types and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
APU Failure; Reactant Supply Failure	Minutes to Days	No	Yes	APU	EPS	Yes	(1),(2),(3)	(10),(11)	(12),(13)
Emergency Shutdown System False Shutdown Alarm	Yes	Yes	Yes	EPS	EPS	Yes	(1),(2),(3)	(11)	(12),(13)

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

It was found that many of the well-understood faults that had correction times low enough to be compatible with software could technically be done either in the EPS, the Space Station System, or the ground. The discriminators used to pick the best area were:

1) Fault Definition Level;

2) Correction Resources Level;

3) External Impacts.

If the fault could be defined and corrected in the EPS without external impact, then it was recommended that the automation should be done in the EPS. If the fault could not be defined or corrected in the EPS or there were external impacts, then it was generally found there would be reason to require some analysis or executive authority at the Space Station System level. The sense and act functions would be at the EPS, but there would be some analysis at the system level. This executive authority could be at the Space Station System level or on the ground. It was generally not partitioned to the ground because of the following reasons:

1) Not minimum ground involvement;

2) Not minimum communications overhead;

3) Lose communications, lose function.

There were some failures that were classed as not practical to automate onboard early in the program. They included solar-array pointing problems due to oscillations in a large flexible structure and plasma interaction. The above faults are recommended to be done on the ground. It is expected that in the initial stages of the space station program, the above faults would not be automated on the ground, but would be handled by human experts. As the program matures, these problems could become candidates to be automated by expert systems software.

6.3 RESULTS OF PARTITIONING OF OTHER AUTOMATION TASKS

A summary of the partitioning of the non-fault-handling automation candidates is shown in Table 6.3-1. A detailed discussion of several potential automation activities is presented in the following paragraphs.

6.3.1 Battery Reconditioning

Battery reconditioning basically involves deep discharging and recharging at a low current. Reconditioning is not necessary more than once every six months. The autonomy-level requirements for the Space Station will be a major driver in the partitioning of this function. For example, if the requirements were for 7-day operation without ground intervention, then the decision could be placed on the ground. If the requirement were for 8-month operation without ground intervention, then the decisionmaking would have to be placed onboard the Space Station.

EPS Partitioning - If the authority to make the decision to recondition the batteries were placed in the EPS, it is likely the decision to permit reconditioning is still required by the system computer or ground.

Space Station System Partitioning - It is functionally acceptable for the decisionmaking to recondition a battery to be placed at the Space Station System level. Because this is an EPS decision, it could logically be assigned to the EPS. The decision as to an exact time to perform the battery reconditioning appears to reside logically at the space station system level because there may be system-level impact in taking a battery offline for reconditioning.

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Table 6.3-1
Other Subsystems and Activities That Can Impact EPS and Partitioning of Correction Tasks

	Analysis						Task Partitioning		
		Correction Approach							
Fault	Time Criticality	Hardware	Software	Fault Definition Level	Correction Resources	External Impacts	EPS	System	Ground
Flexible Structure Oscillations; Degraded Solar Array Pointing	Minutes to Hours	No	Yes	System	System	Yes	--	(1),(8)	(12)
Command and Data Subsystem Degraded Data Rates	Minutes	No	Yes	System	System	Yes	--	(1),(8)	(12)
Command and Data Subsystem, Loss of CPU Power	None	No	Yes	System	System	Yes	--	(1),(8)	(12)
EPS, Crew, and Ground Command Interface	None	No	Yes	N/A	N/A	N/A	--	(1),(8)	(12)
Thermal Control Degradation or Failure	No, Minutes to Hours Because of Thermal Masses	No	Yes	System	System	Yes			
User Load Short or Overload	Shorts, Yes Fractions of a Second. Overloads No, Seconds	Shorts, Yes; Overloads, No	For	EPS	EPS	Yes			

Notes:

(1) Sense Fault
(2) Effect Load Control As Required
(3) Monitor State of Health
(4) Calculate Total Bus Power Capability
(5) Calculate Energy Capability
(6) Isolate Fault
(7) Correct Fault
(8) Isolate and Correct Fault
(9) Enable Automatic Fault Correction by EPS
(10) Generate New Load Sequence Commands
(11) Store Failure Diagnostic Data
(12) Do Trend and/or Failure Analysis
(13) Bus Power Capability and Demand Analysis & Enable Power Management

Ground Partitioning - Due to the slow response time for this decision, it is completely acceptable for this decision to be made on the ground. The range of authority that can be assigned to the ground ranges from none to the authority to decide when to perform the reconditioning. For the early Space Station, ground should decide the time for battery reconditioning.

6.3.2 Battery Charge/Discharge Control

Battery charge/discharge control is a routine function that is performed continuously, 24 hours a day. It is a function that is logically an EPS function. It is a function that is technically acceptable to perform either at the Space Station System level or on the ground. Performing the routine function on the ground would not be consistent with the goal of reducing ground involvement.

6.3.3 Trend Analysis

The principal driver in considering onboard trend analysis is the cost of nonvolatile, mass storage. As an example, 1000 eight-bit words sampled every five minutes will require 104 megabytes per year. Once the decision is made to do onboard trend analysis, there will be a requirement for onboard data-base management, retrieval software, and graphics software for display.

Another decision is how to use the trend data onboard. If use of the trend data is to be automated, then software is required. If the trend data are to be used only manually by the flight crew, there will be a training impact to assure that the crew is at a certified level of competence to interpret and use the data. Another possibility is automated analysis of the trend data but concurrence by the crew or ground before action is taken by the onboard software.

6.3.4 Caution & Warning

It is assumed that a computer will determine the caution, warning, and shutdown status and make it available to the astronauts and ground personnel.

The critical issue is the autonomy level of interpreting the computer-generated status, planning corrective action, and implementing the corrective action. If there is no autonomy, this would mean that a man (astronaut or ground) would be required to interpret the status, plan the corrective action, and input corrective-action sequences to the Space Station.

The next higher level of autonomy would have a computer interpret the status, and plan corrective action. The computer would then advise the man (astronaut or ground) of its analysis and corrective-action plan. The computer would not take any corrective action. The man would be required to input corrective-action sequences to the space station to implement correction. Different degrees of autonomy can be described by the language the astronaut or ground controller uses to command the space station. The least autonomy would occur if a low-level language similar to assembly language were used. The next higher level would occur if a high-level language were used.

Partition to EPS

Detection can be performed at the EPS level because the measurements are available at the EPS level. To place the analysis and corrective-action planning and implementation in the EPS would require sophisticated computer programs. There would be an increase in front-end program costs and a reduction in downstream operating costs. There would be an increase in software development and validation costs. There would be an impact on computer speed, random-access memory, and nonvolatile mass memory.

Partition to Space Station System

The detection function could be done at the system level, but it would result in a higher communications overhead then performing detection at the EPS level.

If caution and warning is put at the space station system level, there are several options as to how to do it. The options are:

1) Astronauts interpret outputs and initiate corrective action;

2) Computer analyzes outputs, advises astronaut, astronauts initiate corrective action;

3) Computer analyzes outputs, initiates corrective action with astronauts' concurrence or initiate corrective action without astronaut concurrence, and then inform the astronaut of the results of the corrective action.

An advantage of completely autonomous operation is that the Space Station can be operated unmanned.

Partition of the Ground - The detection function could be done on the ground, but it would have a higher communications overhead than performing detection onboard. The different levels of ground autonomy are the same as for the onboard system level, with astronaut replaced by ground controller. A disadvantage of performing any of these functions on the ground is that if communications are lost, the function is lost. An advantage is that the Space Station can be operated unmanned.

6.3.5 Space Station Modular Buildup

The growth philosophy entails a complex operation that is not understood in detail at present. With respect to partitioning, the following scenario is postulated for the migration of authority and autonomy over the life of the program. In the first stages of the program, the onboard systems can do automated checkout, but the authority to proceed is received from the ground. The ground would be responsible for the decision to proceed during the validation and early program stages. As the program matures, it is expected the authority to proceed could migrate from the ground-operations crew to ground automated systems, then to the onboard crew, and ultimately, to the onboard automated systems.

It is expected that detail checkout of the EPS assemblies will be partitioned to the EPS even on the initial station, but responsibility for verifying the checkout and authority to proceed to the next step will migrate from the ground crew, to the flight crew, and ultimately, to the onboard automated system.

7.0 TASK 5 - METHOD FOR AUTOMATION TASK ASSESSMENT AND IMPLEMENTATION

OBJECTIVE AND SCOPE

The objective of this task is to develop a system to use all of the information resulting from the first four tasks to provide a logical ordering of automation activities and derived benefits. The system should serve as a logic flow for determining (1) what activities should be considered for automation, (2) what is required to implement the automation, (3) how the options compare, (4) availability of technology, and (5) impact on system performance.

SUMMARY

A study flow plan for automation assessment is shown in Figure 7-1.

The first step is to define a specific study area such as how to automate the correction of overtemperature faults in batteries. Three basic inputs required for the study are:

1) System-level criteria,

   a) Space station autonomy/automation requirements, including autonomy level,

   b) Reliability, maintenance and safety requirements,

2) Subsystem-level criteria,

   a) Functional requirements and description,

   b) Subsystem interfaces,

   c) Component functional requirements,

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3) Mission operations,

a) Man-machine interface,

b) Flight-controller functions (i.e., ground crew),

c) Astronaut/subsystem operational criteria and constraints.

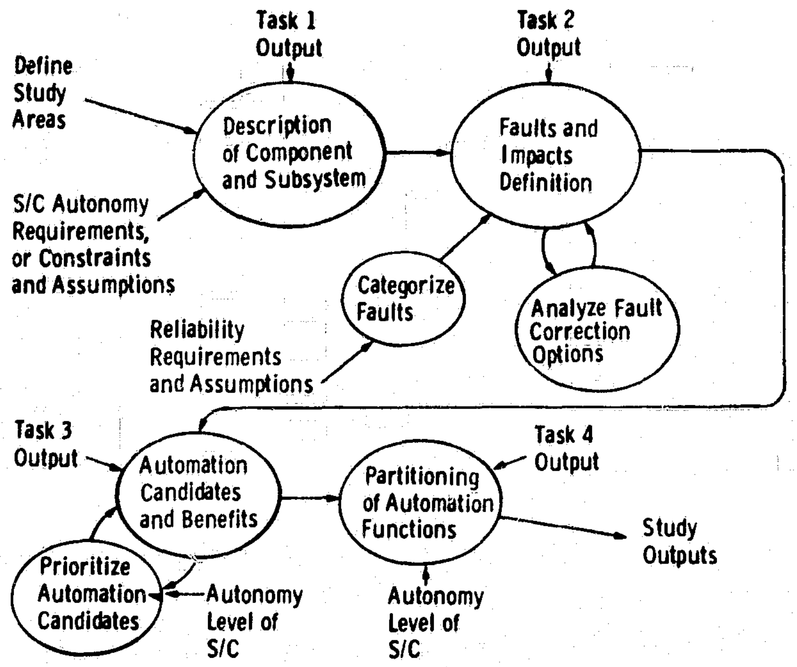

Figure 7-1 Study Flow Plan for Automation Assessment

The autonomy level is used to prioritize automation candidates and aid in partitioning automation functions between the ground and the space station. Reliability requirements are used to categorize faults and to aid in selecting a fault-correction option. Mission-operations criteria are used to define specific automation functions needed for orbital operations.

Factors to be analyzed and defined in a detailed assessment of the automation function are:

1) Impact;

2) Fault category;

3) Fault correction options;

4) Benefits;

5) Time-criticality;

6) Basic implementation, hardware or software.

Basic technical elements in NASA's program development usually consist of Phase A (planning, conceptual requirements definition, and design), Phase B (preliminary requirements definition and design), and Phases C and D (detailed design, fabrication, and integration; launch operations; mission operations). It is assumed that Space Station-level autonomy/automation and reliability requirements will be addressed in each of these program phases, and their details will increase the program phases' progress. The method outlined here depends to a large extent on the system-level requirements available. Therefore, the extent to which automation assessment can be done at the subsystem level is a function of level of details available at the station level. It is logical, then, to assume that the designers, especially during Phases B, C, and D, would have access to top-level specifications and design-criteria documents covering not only autonomy/automation requirements, but also other high-level functional criteria.

Other inputs to the automation-assessment study are the outputs of Tasks 1 to 4. The outputs of the automation-assessment study for one specified area are the following:

1) Description of study area;

2) List of faults and activities from Task 2,

    a) Impacts on subsystem and system (i.e., Space Station),

    b) List of fault-correction options,

3) Automation Candidates from Task 3,

    a) Priority list of automation candidates based on spacecraft autonomy level,

    b) Benefits list,

4) Partitioning of automation candidates between ground and space station based on station autonomy level:

    a) Partition onboard automation between EPS and system based on output of Task 4,

    b) Time-criticality of function,

    c) Basic implementation, hardware or software.

7.1 GENERATION METHOD

7.1.1 <u>Step 1 - Define Study Area</u>

The first step is to define the study area. The study area should be defined in terms of the descriptions used in Tasks 1 to 4. Examples of specific study areas are:

1) Cable overtemperature;

2) Power converter failures;

3) Battery charge/discharge control;

4) Battery operations management.

7.1.2 Step 2 - Define Inputs

The basic autonomy/automation requirements identified in Space Station Definition Book 5 (Ref 9) are listed in Table 7.1.2-1.

Table 7.1.2-1
Summary List of Space Station Autonomy/Automation Requirements

- Implement Autonomy and Automation to Ensure Cost-Effective Operation without Compromising Mission Success or Crew Safety
- Space Station Shall Operate Independent from Ground Support for TBD Time
- Near-Term Activity Planning Shall Be Required Onboard the Manned Space Station
- Consumables Management Required on Board under Supervisory Control of Flight Crew
- Eliminate, As Far As Practicable, the Need for Real-Time Monitoring of Control of EPS by Flight or Ground Crew. Maximize Machine Autonomy to Minimize Crew Involvement in Fault Handling
- Autonomous Handling of Low Faults. High-Level Unsafe Conditions Shall Autonomously Initiate Safe State and Hold for Human Involvement
- Machine Autonomy Shall Be Provided for:
  - Periodic Maintenance--Battery Conditioning
  - Resource Management--Power Management, Battery Energy Accounting and Control
  - Load Sequences Shall Be Autonomously Modifiable in Flight
- Load Sequences Shall Be Autonomously Modifiable in Flight
- Fault-Detection Limits Shall Be Reprogrammable
- Machine-Autonomous Functions Shall Have Individual Enable/Inhibit Control
- Fault-Handling Responses Shall Be Reprogrammable in Flight
- General Approach Is to Place Flight Crew in a Supervisory Capacity and to Program Computers and Machines to Do Most of the Work

The primary driver for the partitioning of automation function between ground and the spacecraft and for the priority ranking of automation functions is the level of autonomy of the spacecraft. For this study, we have used the following definitions of autonomy based on the JPL study in the Air Force's Autonomous Spacecraft Project (Ref 10).

Autonomy	-	The ability of a spacecraft to meet mission-performance requirement without human intervention or ground support for a period of time.
Autonomy Level	-	Level of spacecraft autonomy; increasing level signifies an increased number of automation functions.

The level of autonomy from Reference 10 is reproduced in Appendix C. The following observations were made about the ten levels of autonomy defined by JPL. For level 4 and under, ground intervention is required for fault correction. For levels 5 to 10, the spacecraft is autonomously fault tolerant. As the autonomy level of the spacecraft increases, more capability is placed aboard the spacecraft and less dependence on the ground as the level of autonomy of the spacecraft increases. Figure 7.1.2-1 shows automation functions plotted against level of autonomy for levels 4 thru 10. The figure illustrates the migration of automation functions from the ground to the spacecraft and the decreased dependence on the ground as the level of autonomy of the spacecraft increases.

System safety, reliability, and maintainability requirements will be significant drivers in the automation. For the purpose of our method, the basic reliability requirements from the Space Station Systems Definition Book 5 (Ref 9) is cited as an example of the level of details available during Pre-Phase-A and Phase-A periods. The excerpts from this document are given in Table 7.1.2-2.

Define all basic design, performance, and mission-operations requirements, including all functional interfaces with other subsystems and experiments.

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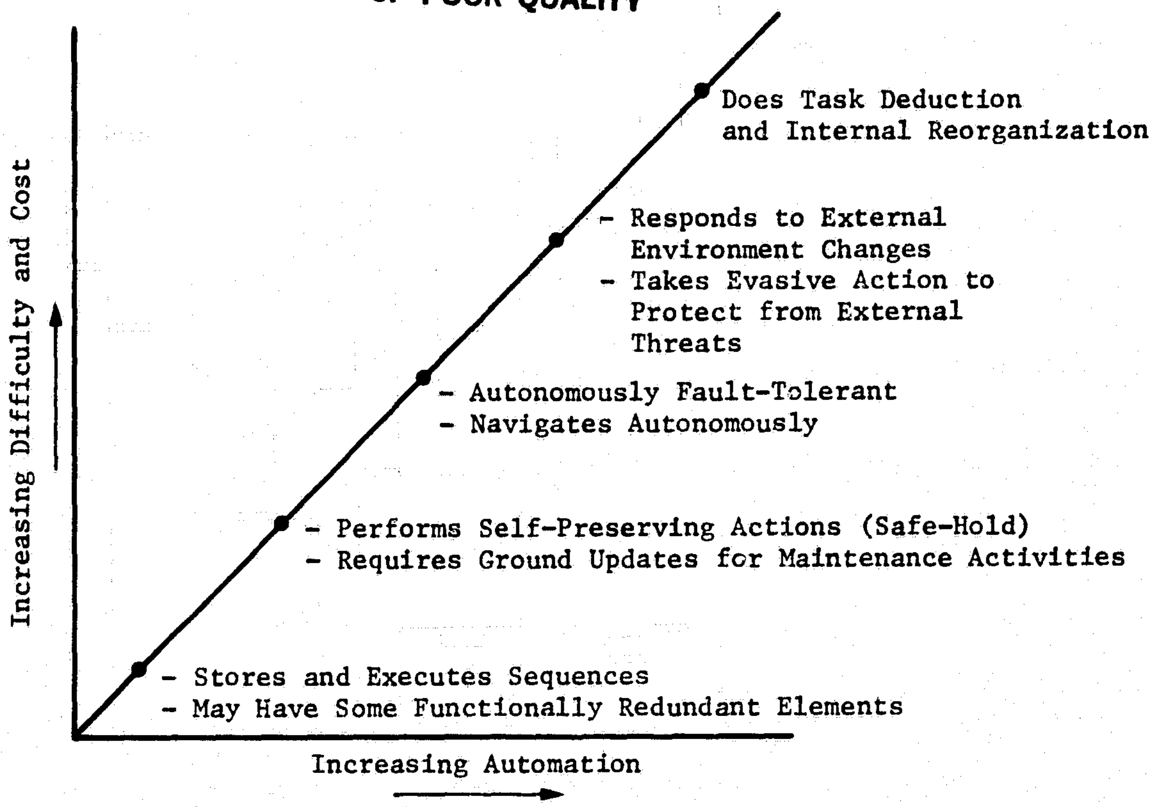

Figure 7.1.2-1
Effects of Increasing Level of Automation on Implementation Difficulty and Cost

Table 7.1.2-2 Excerpts from Space Station Book 5 on Reliability

Requirements

The basic reliability requirement for the EPS is redundancy. The redundancy requirement is that the EPS shall be designed to be fail operational/fail safe as a minimum (except primary structure and pressure vessels) during all operational phases (except assembly and maintenance or repair, all subsystems shall be designed to be fail safe as a minimum.

Applicable Technology/Readiness Assumptions

The intent here is to discuss reliability technology and assumptions applicable to EPS tradeoffs. Assumptions applicable include: (1) Safe operation of Space Station can be assured by an integrated reliability-maintainability approach, (2) Reliability-maintainability must be an integral part of the design, development, test, and operation of each subsystem. Technology applicable includes: (1) hardware redundancy (i.e., replication of subsystem and systems), (2) functional redundancy (i.e., nonidentical subsystems and systems which satisfy common functional requiremental, and (3) higher design margins (i.e., safety factors, high reliability parts). Tradeoff studies of individual subsystems will address reliability-maintainability and safety requirements in arriving at optimum choices between technical options, costs, and performance.

Issues and Trades

A viable reliability-maintainability design approach for Space Station through trade studies will be required early in the program. Limitations on time to restore equipment and on resupply due to failures must be evaluated from the standpoints of reliability, maintainability, safety, and performance.

The basic concept of Space Station long life (10 years to indefinite) with continuous operation has a significant impact on long life technology. Some conclusions can be drawn form the basic reliability requirements from Book 5. Redundancy is a basic requirement. Therefore, redundancy management will be a major automation task. A question about redundancy is, shall the redundant unit be operating continuously or shall it be in the standby mode only. A problem to be faced in redundancy management is accessing the state-of-health of a nonoperating redundant unit.

When autonomy requirements are added to the redundancy requirements a burden is placed on the subsystem designer to assure the system's reliability is increased and not degraded by the addition of redundancy. Redundancy should not be used as an excuse for making the nonredundant element as reliable as possible.

The reliability requirements will drive significant trade studies in the automation assessment area. there are questions of how to implement redundancy. Shall redundancy be at the piece part level, and level within an assembly, the assembly (black box level) or at the subsystem level. The implementation of redundancy will set the level that faults can be detected and corrected.

7.1.3 Step 3 - Define Faults and Impacts

Obtain Information from Task 2 Output - Use the study area defined in Step 1 and obtain the list of faults and impacts from Task 2 results.

Analyze Fault-Correction Options - This is the point where the subsystem designer can introduce the reliability requirements to generate a trade study on the fault-correction options. Table 7.1.3-1 is a list of reliability and redundancy question to be considered by the subsystem designer.

Table 7.1.3-1 Reliability and Redundancy Questions

Hardware Redundancy

What Level?
- Piece-Part
- Board Level in Black Box
- Assembly
- Subsystem

Operating State
- Continuous Operating
- Standby Nonoperating

Block Redundancy Implementation
- Block Size
- Number of Blocks
- Redefine Impact Assessment for Each Successive Block Failure

Functional Redundancy

- Can Nonidentical Assemblies or Subsystems Be Used to Satisfy Common Functional Requirements?
- Increase Design Margins
- Investigate the Possibility of Increasing Reliability by Increasing Design Margins in the Following Ways:
  - Increase Component Derating Factors
  - Decrease Max Allowed Semiconductor Junction Temperatures
  - Move Stringent Piece-Part Screening and Burn-In
  - More Rigorous Worst-Case Analysis

It is likely that there will be Space Station-Level requirements in the above areas. It is also unlikely the subsystem designer will be able to have much impact in the above areas, but he should be aware of them.

Categorize Faults - Two categories can be used for faults:

1) Class I - Mandatory correction;

2) Class II - Correction not mandatory.

From the need to eliminate single-point failures and the requirement for redundancy, one might conclude that it is mandatory to correct all failures and the correction of "not mandatory" to "correct" faults is superfluous; however, there may be low-priority functions that will only be required to fail safe rather than fail operate. Owing to the capability of onorbit maintenance and resupply, some types of faults could assign a fail-safe category, and correction would be by maintenance rather than by redundancy switching. One possible class of fail-safe faults could be low-priority user loads that would be provided by only nonredundant switching and fusing.

This is an area for the subsystem designer to consider--faults where correction is not mandatory--but it is likely the vast majority of faults will require mandatory correction.

For the automation-assessment studies, it is recommended all faults be considered Class I (correction mandatory) unless convincing reasons can be found to classify a fault as Class-II (correction not mandatory).

7.1.4 Step 4 - Determine Automation Candidates, Benefits, and Categories

Automation Candidates and Benefits - Identify the automation candidates and benefits from the output of Task 3.

Prioritize Automation Candidates - At this point, the level of autonomy of the spacecraft can be introduced to prioritize the automation candidates identified from the output of Task 3. A possible set of priority rankings is shown below:

1) Machine autonomy required;

2) Some machine autonomy, but human involvement required;

3) Not practical to automate.

The task for the subsystem designer is now to go through the automation candidates and prioritize them using the level of autonomy from Step 2.

7.1.5 Step 5 - Partition Automation Based on Level of Autonomy

Get Automation Partitioning from Task 4 Output - Use the detail study area defined in Step 1 to obtain the automation partitioning for that area from Task-4 output.

Use Level of Autonomy to Partition - The subsystem designer can use the level of autonomy of the spacecraft to complete the partitioning of automation functions between the spacecraft and the ground. As an example, if level 4 is the level of autonomy being studied, this would require fault detection and safing to be on the spacecraft, but fault correction to be on the ground. If the autonomy level were to be 5, the fault correction function would move from the ground to the spacecraft to satisfy the autonomously fault-tolerant requirements for autonomy level 5.

7.2 METHOD VALIDATION - EXAMPLE 1

7.2.1 Step 1 - Define Study Area

Fault detection, safing, and correction for dc-dc converters (P3 type).

7.2.2 Step 2 - Define Inputs

1) Autonomy level of spacecraft--autonomy Level 5, the spacecraft is to be autonomously fault tolerant;

2) Reliability requirements--as an example, use Section 7.8, "System Safety, Reliability, and Quality Approach," Space Station Systems Definition, Book 5 (Ref 9).

3) Define the basic functional requirements of P^3 and EPS.

7.2.3 Step 3 - Define Faults and Impacts

Identify Faults and Impacts (Table 7.2.3-1) - The subsystem designer can go to Section 4.3 to obtain the list of P^3 failure modes and operational impacts.

Table 7.2.3-1 P^3 (DC/DC Converter) Failure Modes and Impacts

Failure Mode	Cause	Effect	Operational Impact
V_{Out} Hi	Shorted pass transistor, Failed OV Sensor.	Damage loads.	2
(Corrected)	Shorted pass transistor.		5
Low Output Power	Control circuit failure.	Partial Loss of power.	3,4
Efficiency	Filter capacitor leakage, pass transistor switching loss increase, saturation voltage increase.	Assembly overheats.	4
V_{In} Hi	System anomaly.	Assembly may fail.	4
I_{In} Hi	Hi-leak input filter capacitor.	Assembly overheats.	4
Hi Temp	Thermal system failure.	Assembly overheats.	4
I_{Out} Overload	Component degradation, load fault, or overload.	Output overloaded.	4

Analyze Fault Correction Options - In this step, the subsystem designer can use the reliability requirements being used in the study to generate a fault-correction-options list for each of the faults identified in Section 4.3. The obvious fault-correction option is to provide block-redundant dc/dc converters. Block-redundant dc/dc converters will be required. The question that may not be answered in this study is the number of converters required and the amount of redundancy, unless subsystem and component reliability allocations (e.g., 0.965) are available.

A summary of the fault-correction options is shown in Table 7.2.3-2. For each dc/dc-converter failure mode and cause, there is a list of fault-correction options. One option that does not show explicitly in Table 7.2.3-2 is the operational state of the block-redundant converters. A question that must be resolved by the subsystem designer is, Shall the redundant units be nonoperating standby, or shall all the units be operating? Some of the problems involved in operating-versus-nonoperating block redundancy are as follows. It is difficult to determine the state of health of nonoperating units. The control could be made more complex to force a rotation of units from nonoperating standby to primary operating to be able to check the state of health and its performance trend. An advantage of nonoperating standby is that if there were a fault that propagated and failed all operating units, the standby would still be available. The advantage of having an operating redundant unit is minimum response time to correct a failed unit. Disadvantages of operating redundant units are inability to operation them at maximum efficiency and the possibility of a fault propagating and failing all the units connected to a dc bus.

Categorize Faults - There are two fault categories:

1) Mandatory correction (Class I);

2) Correction not mandatory (Class II).

Table 7.2.3-2 Fault Correction Options

Failure Mode	Cause	Fault Correction Options
V_{Out}	Shorted series pass transistor.	- Series, redundant pass transistors consider control complexity increase, decreased efficiency. - Shunt regulator required on bus to detect and blow P^3 input fuse. - Change to transformer-coupled configuration. - N-block-redundant dc/dc converters.
Low P_{Out}	Control-CRT failure.	- Selective piece-part redundancy.
Efficiency out-of spec low	Switching-transistor loss excessive, filter-cap leakage.	- Periodically calculate efficiency and limit check. - Switch to standby and use as backup.
V_{In} High	System anomaly.	- Detect and safe by turning converter off. - Add system-software redundancy to prevent from happening.
I_{In} High	Filter-cap leakage.	- Refer to hardware designers for possible hardware fix. - Periodically calculate and limit check. Remove converter on limit violation.
High Temperature	Thermal subsystem failure.	- Add redundancy to thermal subsystem. - Modularizes thermal subsystem to preclude total failure. - Ensure that there is sufficient thermal mass in converter to make a slow failure (seconds to minutes) to have response time. - Priority load shedding from overtemperature converter. - Priority load transfer to a standby converter.
I_{Out} Overload	Load faults, component degradation.	- Fuse all loads. - Provide active current limiting for each load. - Monitor load Z and remove high-current load. - Periodically monitor loads on bus to ensure that there is adequate margin from converter for fuse clearing. - Make converter overload tolerant.

Inspecting the failure modes from the output of Task 2 for the dc/dc converter, it appears than an efficiency fault could be classified II (correction not mandatory), provided the heating did not exceed shutdown limit. Operation with nonoptimum efficiency would be possible. A possible strategy would be to switch the low-efficiency unit to a nonoperating-standby status and then use it only in the event the main unit failed. Even though the low-efficiency fault could be classified II, it is considered mandatory to periodically access the state of health and check the efficiency.

Except for low efficiency, which can be classified II, all other dc/dc connector faults from the output of Task 2 are classified I (mandatory correction) because if they were not corrected, they would result in loss of power to user loads.

7.2.4 Step 4 - Define Automation Candidates

1) The subsystem design EP can go to Section 5.2 to obtain the list of automation candidates and benefits for dc-dc converters (see Table 7.2.4-1).

2) Prioritize the automation candidates - classify automation candidates in the following three categories:

   a) Machine autonomy;

   b) Some machine autonomy, but human involvement may be required;

   c) Not practical to automate.

Inspecting the dc-dc-converter automation candidates of Table 7.2.4-1, they are all practical to automate; therefore, none are classified III. Further checking of the automation candidate of Table 7.2.4-1 leads to the conclusion that human involvement is not required; therefore, none are classified II. Because categories II and III have been ruled out, then all of the dc-dc converter automation tasks are classified I. What this means is that machines can be used to perform the automation tasks of detecting, safing, and correcting the faults associated with a dc-dc converter.

Table 7.2.4-1
Dc/Dc Converter, P^3 Type Failure Modes, Automation Candidates and Benefits

Failure Mode	Automation Candidate	Method	Benefits
Shorted Series-Pass Transistor	1, 2, 3	Detect overvoltage and close shunt switch.	1, 2, 4.5, 4.6, 6, 7, 15
Low V_{Out}	1, 2, 3	Sense V_{Out}. When valid undervoltage, prior and load sheet and bus test. Determine P^3 good/bad. Determine V_{In} good/bad. If P^3 bad, switch in backup, priority load connect. If P^3 good, source overloaded, limit loads reconnected.	
Efficiency Below Acceptable	1, 2, 3	Switch backup on line, use low-efficiency one as standby.	3, 5, 7, 8
V_{In} High	1, 2, 3	Monitor V_{In}. P^3 shutdown on V_{In} Hi. Shift loads to another P^3, or add loads to one with H^2 V_{In}.	2, 6, 7, 15
I_{In} High	1, 2, 3	Priority load-shed, then if still failed, switch off and bring on backup.	
High Internal Temp	1, 2, 3	Monitor temps, shutdown on overtemp. Bring backup online. Priority load add.	
I_{Out} Overload	1, 2, 3	Monitor I_{Out}, compare to limit, support for programmed time, turn off pause, restart.	

7.2.5 Step 5 - Partition Automation Task

The subsystem designer can go to Section 6.1 and obtain the list of automation partitioning done without regard to spacecraft level of autonomy. The dc-dc converter automation partitioning is given in Table 7.2.5-1.

If a level of autonomy is defined at the Space-Station level, the partition would be driven by it. For this demonstration, a level of autonomy of 5 for the spacecraft was chosen in step 1. The primary meaning of a level 5 is that the spacecraft shall be autonomously fault tolerant and shall do fault correction without ground involvement. To satisfy the requirement for autonomous fault tolerance and fault correction without ground intervention, all of the converter activities must be performed onboard the spacecraft.

7.2.6 Summary of Dc/Dc Converter Automation Assessment

Correction options in addition to block-redundant converters were considered. The low-efficiency fault may not be mandatory to correct if shutdown temperatures are not exceeded. All other converter faults are classified "mandatory correction."

All of the converter automation candidates identified must not require human intervention and should be done by machine. If station-autonomy-level 5 is used, it can be concluded that all functions should be done onboard the space station and not on the ground. It is still the responsibility of the subsystem designer to decide which fault-correction options to implement and to justify the final partitioning between the EPS and system onboard the space station.

The partitioning of automation functions between the ground or the spacecraft is a basic system-design decision. This method illustrates a method of partitioning if a level of autonomy of the spacecraft is given. Use of the high-level autonomy requirement provides a means of tracing the automation partitioning as well as the function-automated space station system requirements.

7.3 METHOD VALIDATION - EXAMPLE 2

7.3.1 Step 1 - Define Study Area

Determine a method of extending cable life on a space station through the application of automation.

7.3.2 Step 2 - Define Inputs

1) Autonomy Level of Spacecraft: Level 5;

2) Reliability Requirements:

   Use Section 7, 8, System Safety, Reliability, and Quality Approach of the Space Station Systems Definition, Book 5, First Edition, November 1982;

3) Define the basic functional requirements of the cable bundle in question.

7.3.3 Step 3 - Define Faults and Impacts, and Analyze Corrective Actions

Faults and Impacts - The faults and impacts for cable are summarized in Table 7.3.3-1. Note that insulation can be degraded by overtemperature condition; cable overtemperature can have numerous causes such as too many wires in a bundle, excessive power transfer, or insufficient heat-sinking. A contributing cause to not detecting and correcting the overtemperature problem can be a lack of temperature monitoring internal to a cable-bundle assembly. The impact is a loss of fault-management capability. In the context of this study, there would be other impacts, namely:

1) Decreased operating power margins;

2) STS resupply mission;

3) Onorbit maintenance;

4) Crew safety.

Table 7.3.3-1 Cabling Failures/Activities and Impacts

Failure or Activity	Effect	Cause	Operational Impact
Cable Opens	Loss of power or signal to user equipment.	Connector Fault	3,4
Wire-to-Return Shorts	Loss of power or signal to user equipment.	Insulation fault.	3,4
Wire-to-Wire Shorts	Loss of power or signal to user equipment.	Insulation fault.	3,4
Insulation Degradation Due to Overtemp	None.	Lack of monitoring.	5,6
Thermal Subsystem Failure	Increase cable temp, decrease allowable power through a cable.	Failure in another subsystem.	4,6
Modular Buildup	Miswiring; open wires.	Inadequate interface design or assembly procedure.	4,6

Analyze Fault-Correction Options - At this stage, the subsystem designer can study the fault-correction options. The first option is to increase the reliability so that monitoring is not required. Ways to increase reliability are to develop higher-temperature insulation; put fewer cables in a bundle to limit cable-temperature increases; heat-sink the cables; match the sources, loads, and cables to make it physically impossible to drive a cable overtemperature in the worst case; and increase the reliability of the power-dispatch software to reduce the probability of a cable going overtemperature. This option study provides a formal way for the subsystem designer to perform trade studies to increase cable reliability.

The next option the subsystem designer can study is the use of block redundancy by adding cables. Questions to be considered are: How much redundancy? Should it be operating or nonoperating? Other questions relate to redundancy level. Should the cables, including connectors, be redundant? Should the insulation be made doubly redundant, or should the wires in a cable be made redundant? Are there different routes for redundant cables? It is necessary to ensure that there is no mechanism that could damage both the primary and the redundant cable?

The above fault-correction options were included to focus on some of the reliability studies that the subsystem designer could perform to lay a foundation for meeting reliability requirements. The subsystem designer could use the above studies to decide if the probability of a cable overtemperature is high enough to warrant installing and monitoring the temperature detectors in the cable assembly.

Categorize Faults - There are two fault categories:

1) Mandatory correction;

2) Correction not mandatory.

Inspection of the faults in Table 7.3.3-1 leads to the conclusion that cable overtemperature may be classified as II for temperatures below immediate failure if decreased cable life is preferred over higher operating temperatures. If cables must not operate above a temperature threshold, all of the faults are classified I (correction mandatory) because failure to correct would violate the no-single-point-failure criteria.

7.3.4 Step 4 - Define Automation Candidates and Benefits

The list of cabling failure modes and automation candidates and benefits is given in Table 7.3.4-1.

The set of automation priority categories is shown below.

1) Machine autonomy required;

2) Some machine autonomy, but human involvement required;

3) Not practical to automate.

It is practical to automate all cable fault correction, except correction of faults due to station modular buildup. Detection and safing of cable faults is possible, but correction of the underlying modular buildup problem is not practical to automate without a definite design.

The thermal subsystem failure could be classified a II (some machine autonomy, but human involvement required). The reason is that a thermal-subsystem failure may be classed as a high-level unsafe condition that will require human involvement. It is expected that machine autonomy would be provided for cable fault detection, safing, and correction, but human involvement would be required in correcting the underlying thermal subsystem failure.

The cable-open, short, or overtemperature failures are classed as I (machine autonomy required), because it is practical to have a computer detect, safe, and correct these faults.

7.3.5 Step 5 - Automation Partitioning

Correction tasks for the cable high-temperature and insulation faults should be partitioned to the spacecraft and not to the ground. A cable fault caused by modular buildup of the space station has dual partitioning. Electrical problems associated with the modular buildup, fault detection, safing, and correction, are partitioned to the spacecraft. For the early stages of the program, it is thought that correction of the underlying problems associated with modular buildup are not routing problems. It appears highly probable that human involvement will be required to resolve modular buildup problems. One study area will be to determine where the expertise should be--with the flight crew or with the ground. A possible conclusion is that the expertise should be on the ground to minimize crew training for nonroutine operations.

7.3.6 Summary of Cable Automation Study

The study area was defined as how to extend cable life through automation. Fault-correction options include:

1) Use of high-temperature insulation;

2) Fewer cables in a bundle;

3) Heat-sink cables;

4) Match sources, cables, and loads to make it impossible to drive a cable overtemperature;

5) Load management;

6) Block-redundant cables;

7) Double insulation;

8) Multiple wires for cable;

9) Different physical routing for redundant cables;

10) Monitor critical cable bundle temperatures and provide appropriate control.

A cable-overtemperature fault may not be mandatory to correct if it is decided to trade cable operating life for cable temperature. Otherwise, all cable faults defined are classified as "correction mandatory."

Problems arising from modular buildup are considered nonroutine, and their correction activities will likely be partitioned to the ground early in the program. All other automation activities were partitioned to the space station.

8.0 ARTIFICIAL INTELLIGENCE (AI) TECHNOLOGY AND ITS ROLES

8.1 AI TECHNOLOGY

Artificial intelligence is that branch of computer science concerned with the design and implementation of programs that make complicated decisions, learn, or become more adept at making decisions, interact with a man in a natural way, and, in general, behave in a manner typically considered the mark of intelligence.

Intelligence is to be understood not as a property that, for example, gifted mathematicians possess, but rather as a property all men and some animals possess. Intelligence, in this sense, is the ability to understand and process large amounts of information. It is the ability to meet and cope with novel situations, to comprehend the interrelationships between facts and concepts, and to generate new concepts and relationships from those already known (i.e., already in the data base). The artificiality of the intelligence means merely that the intelligence is achieved by means of technology.

Scientific research done in AI covers a large area of theoretical topics such as knowledge representation, knowledge acquisition, problem solving and search, vision, theorem proving, and natural language. Though each one of these topics can be researched from the human-ability perspective, i.e., by asking how a man represents knowledge, acquires knowledge, solves problems, sees objects, communicates, etc, researchers in AI are concerned with implementing the given ability in computers. AI is not only a theoretical enterprise, it has definite and robust applications. The primary concern in the applications arena is the design and implementation of expert systems and natural language interfaces.

Aside from the general scientific curiosity of wondering how to design and implement a computer program that learns, what advantages might obtain from the application of AI? Specific examples cited below are some rather broad, obvious ones.

1) Augmenting our ability as humans to come to grips with the enormous and increasing amounts of information that we are generating;

2) Increasing the efficiency in man/machine interfaces (the ability to communicate with a computer in English) enables humans to get more work done and obviates the need for specialists in those hard-to-use formalisms known as modern computer languages and data-base query languages;

3) Creating systems (such as space vehicles) that can make crucial decisions on their own when they have to;

4) Decreasing the effect of such human problems as forgetfulness, fatigue, and emotional turmoil;

5) More rapid problem solving, and strategic and tactical planning, in a wide variety of domains.

8.1.1 What Is An Expert System?

An expert system is an intelligent computer program that embodies the knowledge of human experts in a particular domain of expertise. Expert systems recognize situations, derive conclusions, make decisions based on what they recognize, and recommend corrective and directive actions. All of this is done with a competence comparable to that of human experts. Figure 8.1.1-1 illustrates the basic components of an expert system. It contains a knowledge base, a rule base, and an inference engine. The knowledge base (sometimes called working memory) stores the information (data) on which the expert system operates. The knowledge base is constantly updated as data are added or deleted. The rule base is the component that gives the expert system its expert competence--that is, the ability to make decisions, recommend actions, etc.

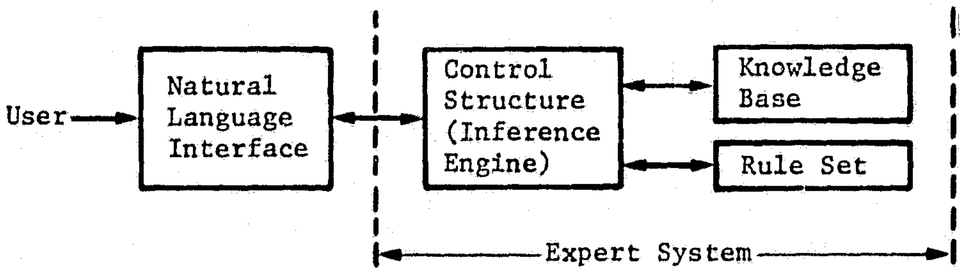

Figure 8.1.1-1 Basic Components of an Expert System

Rules are of the form:

IF conditions A, B, and C are true, THEN perform actions X and Y.

Hence the rules are referred to as condition-action or situation-action pairs.

The inference engine's job is to execute various rules depending on the contents (data elements) of the knowledge base. Conceptually, the inference engine's algorithm is a search and pattern match. It scans the rules, efficiently searching for a rule whose antecedent (the IF part) matches the present state of the world, i.e., the facts in the present knowledge base. If a match is found, the consequent of the rule (the THEN part) is executed. The actions can be anything from querying or advising a human user to performing a real-world action, such as uplinking commands to a satellite or moving a robot arm, to manipulating its knowledge base or rule set and modifying the behavior of the expert system itself.

The rules of the rule set are obtained by interviewing a human expert. This is a tricky and involved process because experts cannot just be debriefed. One could not, for example, walk up to a physician and say, "Tell me how to diagnose and treat a sick person," and hope to produce an expert system. Human experts often are not quite clear about how they do the things they do. Rather, the knowledge of their field must be ferreted out by someone who knows (or discovers) what questions to ask and more importantly, how to ask them. The experts might be given problems and asked how they would solve them, with each step in the solution being fully documented. In fact, a step may require posing another problem in order to explicate it. Interviewing is frequently a lengthy process, but this is what forms the basis of the expert-system technology. Building an expert system is not possible without an expert.

All of the ability of an expert system stems from its matching antecedents and executing consequents. Almost all of an expert system's power derives from the depth of understanding and the cleverness of human experts captured in its rules. It is also important, however, to develop an organizational scheme for the rule set so that efficient searches can be obtained, and it is important to have the knowledge base organized in a way that allows for rapid access, rapid addition and deletion of facts, and, most importantly, the capturing of complex relations between facts that make the knowledge base rich.

The problem of knowledge-base organization is referred to in the artificial intelligence community as knowledge representation. Probably the most favored basic approach to knowledge representation is the directed graph. But the variations on this theme are numerous, and there is some controversy as to which variation is "correct." At stake, it is believed, is not merely an implementational formalism detail, but the deriving of a representation that gives (1) the right facts in the world, and (2) the right relationships between the facts.

Expert systems are designed for, and are most useful in, areas that heretofore relied only on the judgment of human experts--that is, in areas where the problems to be solved are complex, not easy to delimit, and require the use of high-level judgments and evaluations of situations. Thus, expert systems are not designed, or intended, to replace all problem-solving software. Many problems require algorithmic solutions, but many do not; those that do not require experts to evaluate and assess situations and then make judgments based on these assessments. Expert systems exist because such evaluations and judgments can be transformed into rules and then implemented in a programming language.

Another feature that expert systems exhibit that increases their viability is that the rule set can be thought of as data--that is, as part of the knowledge base. This enables the expert system to alter the rules of the rule set in various ways. Under some circumstances, it is possible to view this alteration of the rule set as learning; some expert systems have this feature and do become more adept at decisionmaking. This learning feature is obviously very desirable, and although the technology involved is not yet commensurate with that for deducing and inferencing in expert systems, it is only a matter of time before expert systems incorporate some degree of learning.

8.1.2 Natural Language Interface

It is usual to have a natural language interface to facilitate the use of the expert system. A natural language interface is a computer program that allows an end user to interact with an applications program using a "natural" language such as English rather than special menus or special-purpose languages such as FORTRAN for programming, RAMIS for data-base queries, or JOVIAL for command and control. A key advantage to using a natural language interface rather than a more conventional interface is ease of learning and use. Because English is used, no special languages must be learned. Because its use is an extension of a person's normal communication skills, a natural language interface can often be a highly effective way to interact with a computer program.

The appropriateness of a natural language interface in a given domain is a human factors question; How much will such an interface simplify the activity of the end user? The answer turns on several issues. Foremost is the range of interaction the user will have with the computer program. As noted above, a major difficulty with conventional interfaces is that they often have highly rigid formats and require substantial training. The larger the interactions, the longer the training period and the more difficult it is to remember the specific format required for a particular interaction. When there are only a few interactions (or types of interactions), the more conventional interface might be more appropriate.

The more complex the program the user is working with, however, more likely is the user to want greater interface with the computer program. For many automation activities, the program will be an expert system with a wide variety of capabilities. The wider this variety, the more desirable a natural language interface. Users do not have to learn intricate, easy-to-forget aspects of a special-purpose query or command language. In simply knowing what the system can do, a user can couch a command or query in English and let the system figure out how to respond.

This flexibility is quite important. Menu-driven interfaces have a certain amount of this flexibility also. A sophisticated, well-designed menu system can sometimes be used by individuals who have no training for that menu, especially if they have experience with other menu systems. With no training for a particular menu system, however, "solving" the menu--determining what commands are in which layer of the menu hierarchy--can be tedious and time consuming. Once the menu is known, the layering of menus can become more of an obstacle than a facilitator. Some menu systems attempt to overcome this obstacle by allowing experienced users to type in the commands directly without wading through the menu. Unfortunately, this solution is really just a special-purpose interaction language with many of the same problems as discussed above. It is, however, better than having only a standard special-purpose language because the users can fall back on menus if the special commands are forgotten.

Natural language interfaces resolve the problems of forgetting and having to "solve" the menu. Users never need to learn a menu or a special language; with no special training, users can interact with the system with the same English they use for everyday communication. These "ordinary" language skills can be immediately transported from system to system as special-purpose language skills cannot. Highly sophisticated natural language interfaces are also able to train the user in the capabilities of the end system, eliminating the need for highly detailed knowledge of what the system can do before sitting down to use it.

An occasional argument against natural language interfaces is an alleged loss of efficiency--it takes too long to type complete, grammatical sentences. Once learned, it is claimed, a special--purpose language is much faster and easier. However, the ideal natural language interface would be able to understand English with all the grammatical errors, incompleteness, and inaccuracies found in everyday use. A great deal of work presently is being done in these areas, and significant progress has been made. When continuous speech recognition is perfected--probably sometime in the next few years--the obstacle of needing to type will be eliminated. At that point, the utility of natural language interfaces will far outstrip that of more conventional interfaces for a vast portion of applications.

Expert System Status - Expert systems have existed since 1965 when DENDRAL was introduced. DENDRAL infers the molecular structure of compounds from their spectrogram data. In 1974, MACSYMA was built. MACSYMA is an expert system that does symbolic manipulations of mathematical expressions. Also in 1974, MYCIN was completed. This expert system is perhaps the most famous: it provides diagnoses and prescriptive advice to physicians treating patients with blood-related diseases. All of these expert systems (and there are many more) are being used today either in research tasks designed to test their total capabilities or in narrowly confined aspects of industry. However, an explosion of new applications presently is underway throughout industry and the universities. Within the next decade, expert systems are expected to move out of the laboratories and become increasingly involved in human affairs. In fact, in 1981, R1 was installed for commercial use by Digital Equipment Corporation for configuring their VAX-11 computer systems.

8.2 CRITERIA FOR IDENTIFYING EXPERT-SYSTEM SOFTWARE CANDIDATES

A given candidate for automation warrants considering an expert system approach if:

1) For potential control application, non-real-time processing or very slow response is required;

2) Automating the given activity requires processing large amounts of information that are available in random fashion;

3) The processing involved requires nonalgorithmic and heuristic procedures. In fact, for some activities, there may be no algorithmic procedures, at least not to anyone's knowledge;

4) The automation activity needs, or results in, a high-level decision (e.g., one that affects several spacecraft subsystems);

5) The software responsible for automating the given activity will be frequently modified as a result of the dynamic influences of its environment or as a function of time.

Another discriminator to identify automation tasks for expert systems is complexity and how the tasks have been performed in the past. Simple tasks that are well understood and have algorithmic solutions are not good candidates for expert-system solution. The expert-system solution could be an overkill. If the task is complex enough that in the past it could only be performed by a recognized expert, or group of experts, then the task is a good candidate for automation by expert-system software.

8.3 POTENTIAL ROLES OF THE EXPERT SYSTEM IN POWER SUBSYSTEM AUTOMATION

Several power-subsystem and space station system-related functions appear to be in the domain of the expert system, and thus are good candidates for an indepth evaluation of expert-system software applicability. Table 8.3-1 is a list of these functions. Note the level of complexity in electrical consumables management and battery-operations-management tasks. It is also emphasized that algorithmic and deterministic software modules are involved, along with expert system module, in many of the potential applications. This simply means that even if expert system approach is used, there is a large amount of engineering algorithm development and validation efforts.

Table 8.3-1 List of Potential Expert System Candidates

Function
Electrical Consumables Management
- Power Capability Determination
- Load Profile Determination
- Load Shifting and Shedding Analysis
- Energy Balance Calculation
- Load Sequence Control and Load Command Generation
- Power Subsystems Reconfiguration
- Power Subsystem State Determination
Battery-Operations Management
- Battery Cell/Module State of Health Determination
- Battery SOC Trend Analysis
- Battery Loadsharing Analysis and Control
- Battery Recharge Fraction Adjustment Analysis and Control
- Battery Cycle Life Analysis
Performance Trend Analysis
- All Major Components
Fault Detection and Diagnosis
- All Major Components
Anomaly Analysis

9.0 CONCLUSIONS AND RECOMMENDATIONS

The significant conclusions and recommendations of the study are as follows:

1) To meet basic station objectives and goals presently defined in the NASA Space Station Definition Book, all power subsystem automation candidates defined in this study, except for anomaly handling, must be implemented to a varying degree of automation.

2) Specific functions that have immediate high payoffs for onboard applications are:

   a) Data Acquisition, Processing, and Storage,

   b) State of Health Monitoring,

   c) Built-in Test and Checkout,

   d) Fault Detection, Isolation, and Correction,

   e) Performance and Trend Analysis,

   f) Integrated Array/Battery Controller and Load Management (Space Station Level),

   g) Electrical Consumables Management (Space Station Level).

   Automation of any combination of the above functions (a through g) will have a significant beneficial effect on mission-operations efforts on the ground. A detailed study is recommended to determine the effects of onboard automation of monitoring functions on ground activities such as failure detection, consumables management, and crew and flight-controller training.

3) A key driver in when and what to automate in the subsystem is spacecraft autonomy level, which must be defined at the program level.

4) The best way to partition an automated activity between the EPS, spacecraft system, and ground is to first define each subtask required to be performed, and then assign each subtask to EPS, system, and ground, in terms of:

   a) Sensing,

   b) Analyzing,

   c) Acting.

5) For real-time control consideration, the principal driver in hardwired-versus-software (i.e., using digital computer) trade is the speed requirement for implementing that control function. Therefore, in general, all offline or non-real-time tasks such as monitoring, performance analysis, and fault diagnosis that require slow response and are not in the control loop, can be done with a digital computer.

6) The best onboard-application candidates for expert systems for any of the power automation functions appear to be for electrical-consumables management and battery-operations management. Potential ground applications are in non-real-time fault assessment and mission planning. An indepth research investigation is desirable and highly recommended to determine:

   a) The range and domain of its applicability to power-system control functions;

   b) Adequacy of AI language for onboard use;

c) Computer hardware (speed, memory) required to support expert-system software.

7) A significant effort in engineering-algorithm development and validation is essential in meeting the 1987 technology-readiness date. There are many implementation approaches to each automation function because they are done by software. Thus, future efforts in algorithm development must include optimization processes with simplicity and reliability in mind. It should be emphasized that algorithm development also is necessary to permit a detailed design of any expert-system software such as that for electrical consumables and battery management.

10.0 REFERENCES

1. Autonomous Spacecraft Project: Autonomous Spacecraft Design and Validation Methodology Handbook. Issue 1, SD-TR-82-58, Jet Propulsion Laboratory, April 30, 1982.

2. W. B. Gevarter: "Expert Systems: Limited But Powerful." IEEE Spectrum, August 1983.

3. Patrick Winston: Artificial Intelligence. Addison-Wesley Publishing Co., 1979.

4. Autonomous Spacecraft Program--ARMMS Demonstration Project. Report No. 7030-8, Jet Propulsion Laboratory, October 18, 1982.

5. 250-kW Autonomously Managed Power System (AMPS): Power Management Subsystem Phase II Overall Review. Document No. 34579-075, TRW Defense and Space Systems Group, Redondo Beach, California, October 27, 1981.

6. F. E. Lukens and R. L. Moser: "Programmable Power Processor." 16th IECEC Proceedings, August 1981.

7. M. S. Imamura, R. L. Moser, and L. Skelly: "Design and Development of Automated Power System Management," Final Report. MCR-79-540, Martin Marietta Report, JPL Contract 954924, July 1979.

8. M. S. Imamura and R. Donovan: Development of Single-Cell Protectors for Sealed Silver-Zinc Cells. NASA CR-135054, Contract NAS3-19432, September 1976.

9. Space Station System Defintion, Book 5. NASA Headquarters, November 1982.

10. M. H. Marshall: "Goals for Air Force Autonomous Spacecraft." JPL Internal Document 7030-1, Preliminary Issue, February 1981.

11. <u>Functional Requirements for the Demonstration Systems, Autonomous Spacecraft Program - ARMMS Demonstration Project</u>. 7030-7, Jet Propulsion Laboratory, September 1982.

12. <u>Assessment of Autonomous Options for the DSCSIII Satellite System</u>. SD-TR-81-87, Jet Propulsion Laboratory, August 1981.

13. <u>Military Space Systems Technology Model</u>. Vol III, USAF Space Technology Center, 1983.

14. M. S. Imamura, Dr. B. Khoshaim, and F. Huraib: "470-kW Photovoltaic Power System for Saudi Arabian Villages." 1980 IECEC, Seattle, Washington, August 1980.

15. F. Huraib, et al: "Design, Installation, and Initial Performance of 350-kW Photovoltaic Power System for Saudi Arabian Villages." 4th European Solar Energy Conference, Stresa, Italy, May 1982.

16. <u>Solar Array Technology Development for Solar Electric Propulsion, Fianl Report</u>. Contract NAS8-31357, LMSC, January 1981.

17. <u>Study of Multikilowatt Solar Arrays for Earth Orbit Applications, Final Report</u>. 33295-6001-UT-00, Contract NAS8-32988, TRW, September 1980.

18. R. E. Patterson, "Cassegranian Concentrator Solar Array Exploratory Development Module." 17th IECEC, August 1982.

19. <u>Study of Multi-Kilowatt Solar Arrays for Earth Orbit Applications, Final Report</u>. SSD80-0064, Contract NAS8-32988, Rockwell International, May 1980.

20. W. R. Scott and D. W. Rusta: Sealed-Cell Nickel Cadmium Battery Applications Manual. NASA Reference Publication 1052, 1979.

21. J. K. McDermott: "Status and Evaluation of Air Force/Hughes $Ni-H_2$ Prototype Cells." Interoffice Memorandum, Martin Marietta Corporation, September 21, 1982.

22. Robert L. Cataldo and John J. Smithrick: Design of a 35-Kilowatt Bipolar Nickel-Hydrogen Battery for Low-Earth Orbit Applications. NASA Tech Memorandum 82844, NASA Lewis Research Center, Cleveland, Ohio, 1982.

23. Lawrence H. Thaller: Nickel-Hydrogen Bipolar Battery Systems. NASA Tech Memorandum 82946, NASA Lewis Research Center, Cleveland, Ohio, 1982.

24. Sid Gross: Analysis of Regenerative Fuel Cells, Final Report. D180-27160-1, Boeing Aerospace, Co., November 1982.

25. Julian F. Bedu: "The Role of Fuel Cells in NASA's Space Power Systems." 14th IECEC, August 1979.

26. F. C. Schwartz: "An Improved Method of Resonant Current Pulse Modulation for Power Converters." Power Electronics Specialists Conference Proceedings, 1975, p 194.

27. J. W. Mildice: "Power Management for Multi-100-kW Space Systems." 15th IECEC, August 1980.

28. John R. Sheic, Robert E. Corbett, and Michael C. Glass: "Series vs Shunt Regulators for Power Control in Satellite Power Systems." 16th IECEC, August 1982, p 235.

29. Willie A. Magee, Robert M. Martinelli, and Joseph H. Hayden: "Solar Array Power Management." 17th IECEC, August 1982, p 171.

30. J. Cassinelli: Solar Array Switching Power Management Technology for Space Power Systems, Final Report. Final Report TRW 37243, NASA CR-167890, September 1982.

31. M. C. Glass: "A Six-kW Transformer-Coupled Converter for Space Shuttle Solar Power Systems." 15th IECEC, August 1980, pp 767-772.

32. G. R. Sundberg and W. W. Billings. "The Solid State Remote Power Controller: Its Status, Use, and Perspective." PESC, June 1977, p 244.

33. Engineering-Dependable Protection for an Electrical Distribution System. Part III, Component Protection for Electrical Systems, Bussmann Manufacturing Division, McGraw-Edison Company, 1975.

34. R. E. Corbett: High-Voltage High-Power (HVHP) Solar Power System. AFWAL-TR-81-2103, April 1982.

35. S. M. Weinberger: Preliminary Design Development of 100-kW Rotary Power Transfer Device. NASA CR-GE-B15D4215, General Electric Company, Space Division, Philadelphia, Pennsylvania, Contract NAS3-22266, March 1981.

36. W. D. Cooper: Electronic Instrumentation and Measurement Techniques. Prentice Hall, 1978.

37. D. W. Harris: "The Modular Power Subsystem for the Multimission Modular Spacecraft." 13th IECEC, August 1978.

38. G. Repucci, I. M. Schulman, and W. Wright: "Flight Performance of the High Energy Astronomy Observatory (HEAO 1) Power System." 13th IECEC, August 1978.

39. M. Tasevoli: "OAO-3 End of Mission Power Subsystem Evaluation." 17th IECEC, August 1982.

40. Aerospace Corporation Data, H. Weiner, December 1983.

41. J. Hayden: "An Electrical Power Subsystem for the Defense Satellite Communication System - Phase III." 13th IECEC, August 1978.

42. R. E. Corbett: "High Voltage High Power System." 13th IECEC, August 1978.

43. A. Salim: "A Simplified Minimum Power Dissipation Approach to Regulate the Solar Array Output Power in a Satellite Power Subsystem." 11th IECEC, August 1976.

44. G. Repucci: "FLTSATCOM - A Power Subsystem in Evolution." 18th IECEC, August 1983.

45. Private Communication with H. Weiner, Aerospace Corporation, December 1983.

46. H. N. McKinney and D. C. Briggs: "Electrical Power Subsystem for the Intelsat-V Satellite." 13th IECEC, August 1978.

47. J. Segrest: "High Voltage DC Power Systems for Aircraft." IEEE-NAECOM, May 1971.

APPENDIX A

STATEMENT OF WORK

A. The contractor shall provide the necessary personnel and facilities to conduct the required studies and perform appropriate assessments and trade-offs to define and establish the automation technology required to support a multi-hundred KW electrical power subsystem for a space platform or space station. This study effort will not rely on a specific reference design but will be more generic in nature. Consequently, the study must include a broad characterization of subsystem parameters, functions and operational scenarios.

B. Specifically, the contractor shall perform the following tasks:

Task 1. Characterize and classify a generic electrical power subsystem based on a conceptual system block diagram(s) that includes a definition of the functions, characteristics, voltage types, voltage block diagram. This task shall be done for each phase in a mission profile (i.e., pre-launch, launch, orbital operations, on-orbit service/maintenance/resupply, etc.)

Task 2. Using the results of task #1, develop a comprehensive list of all potential faults and/or activities that could impact the power subsystem and prevent it from performing its intended mission. This will include such parameters as operational environments, single point failures, insufficient redundancy, human error, over-stressed conditions, inadequate protection, inaccurate sensors, etc.

Task 3. Based on tasks 1 and 2 above, generate a candidate list of automation activities that could eliminate and/or minimize the identified impacts as well as those activities not related to impacts that can provide both a short term and a long term benefit to the power subsystem if incorporated. This would include such activities as redundancy, derating, fault management, shifting burden from man to machines, algorithms for management strategies, partitioning of functions between the space station and ground, hierarchy control of functions, etc. Perform an assessment and trade-offs on all automation activities to determine such aspects as range of benefits to be achieved (performance, cost, weight, volume, complexity, etc.), timeline for implementation, system performance improvements, reduced operations burden, relaxed critical measurements (i.e., red line values, limitations, etc.), preprocessing of data, flexibility in scheduling, and other similar activities that will improve performance, reduce costs, reduce dependence on manual involvement, increase operational life and reduce the overall life cycle cost of the power subsystem.

Task 4. Partition the automation activities between the power subsystem, the space station and the ground to maximize the overall configuration in terms of operations management, information flow, controls distribution and system performance. Establish criteria for the partitioning and generate rationale for the resulting configuration. A comparison of the benefits before and after the partitioning shall be done to determine the value of the benefits derived.

Task 5. Develop a system for utilizing all of the information and data resulting from the above tasks to establish a logical ordering of the automation activities vs. derived benefits. Benefits begin such elements as costs, time, reliability, fault isolation, system protection, system recovery, self monitoring and reconfiguration, etc. The end product of this task should be in a format such that the requirements, characteristics, constraints, values, methods, and other parameters that describe introduced and processed to provide a system level engineering approach to the automation of that power subsystem. In essence, the resulting system or plan will serve as a "logic flow" methodology for determining what functions and/or activity should be considered for automation, what is required to implement the automation (options), how do the options compare (cost, complexity, value, etc.) interactions with other elements and/or activities, availability of the technology, impact on system performance, etc. Therefore, the developed system will test the application of automation technology, evaluate it, provide directions and quantify benefits. Specific examples shall be demonstrated to verify the concept.

GUIDELINES, CONSTRAINTS AND INSTRUCTIONS

The following are intended to focus the efforts in conducting the tasks for this study.

A. The space station electrical power subsystem is targeted at 250 KW and probably modular. The space station is large, in low earth orbit, unmanned and manned and has a life of greater than 10 years.

B. Inputs involving automation activities at the space station level will be provided by the COR. JPL is conducting an "Autonomous Spacecraft System Technology" task that will define autonomous system design requirements, develop system architectures (including partioning of functions) and identify enabling and enhancing technology needs. MSFC and JPL will coordinate the respective tasks and all inputs from this effort (specific partitioning of functions, automation criteria, command and control functions, centralized vs. distributed controls, etc.) will be provided only through the COR.

APPENDIX B

SIMPLIFIED BLOCK DIAGRAMS OF VARIOUS SPACECRAFT PHOTOVOLTAIC POWER SYSTEMS

This appendix contains simplified block diagrams of selected photovoltaic power systems on LEO, medium altitude, and GEO spacecraft. Representative terrestrial and aircraft systems are also included.

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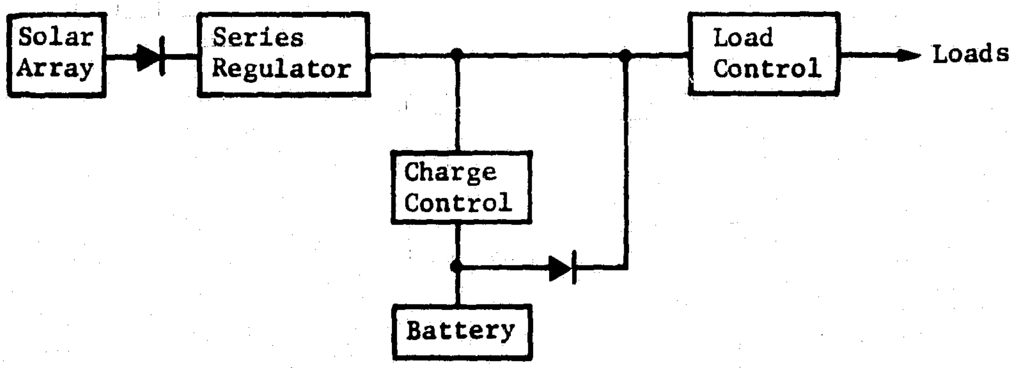

Figure B-1 MMS Power System, LEO Application (Ref 37)

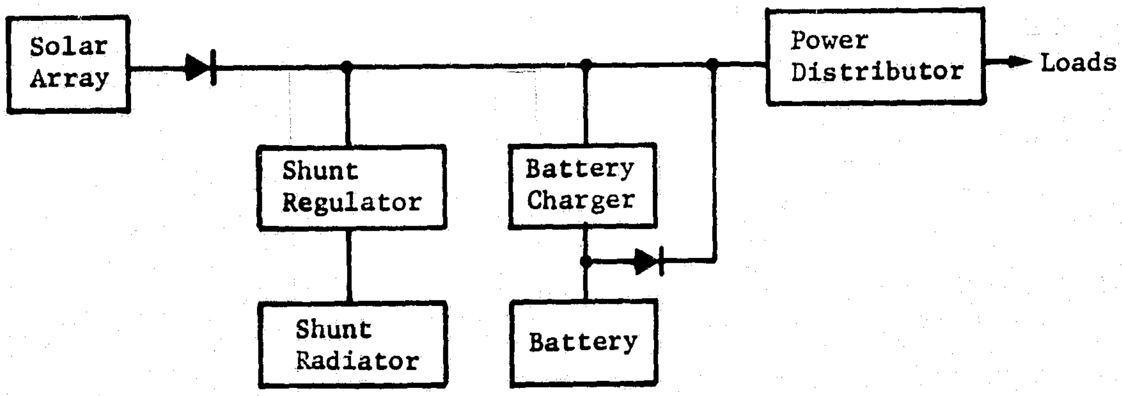

Figure B-2 HEAO Power System, LEO Application (Ref 38)

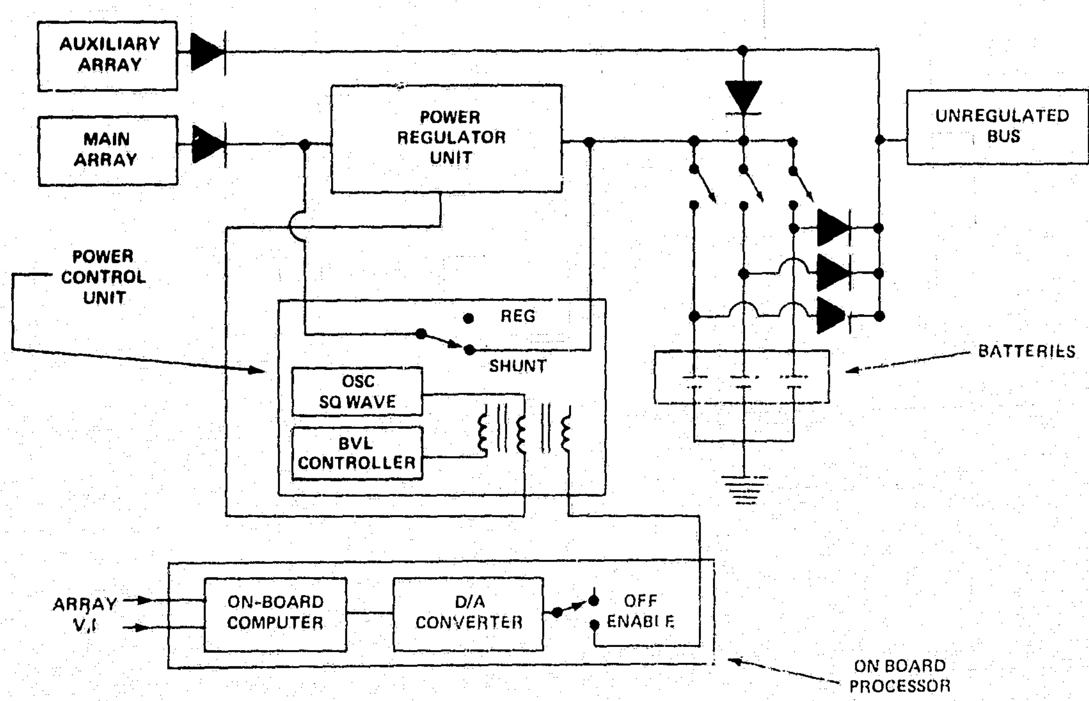

Figure B-3 OAO-3 Power System, LEO Application (Ref 39)

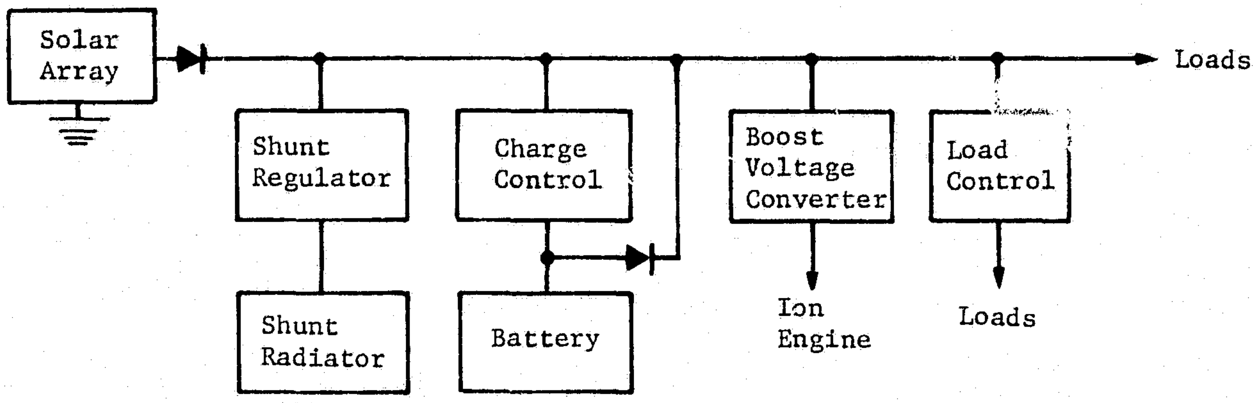

Figure B-4 P80-1 Power System, Medium Orbit Application (Ref 40)

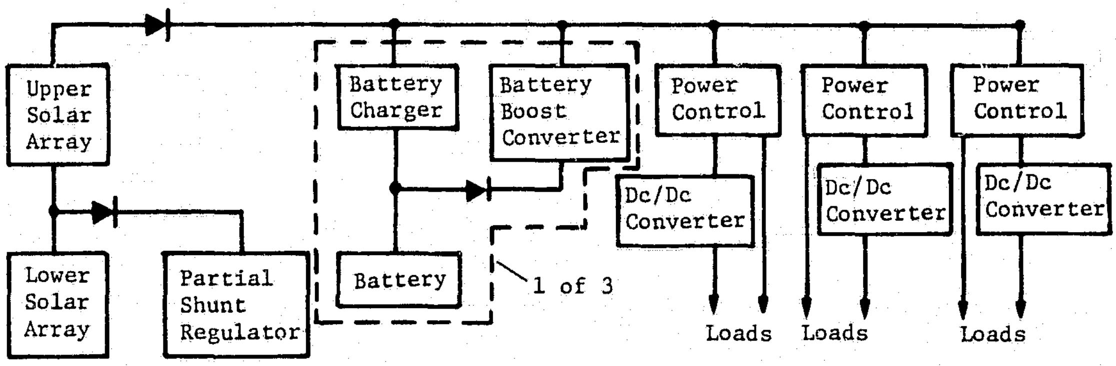

Figure B-5 DSCS-III Power System, Medium Orbit Application (Ref 41)

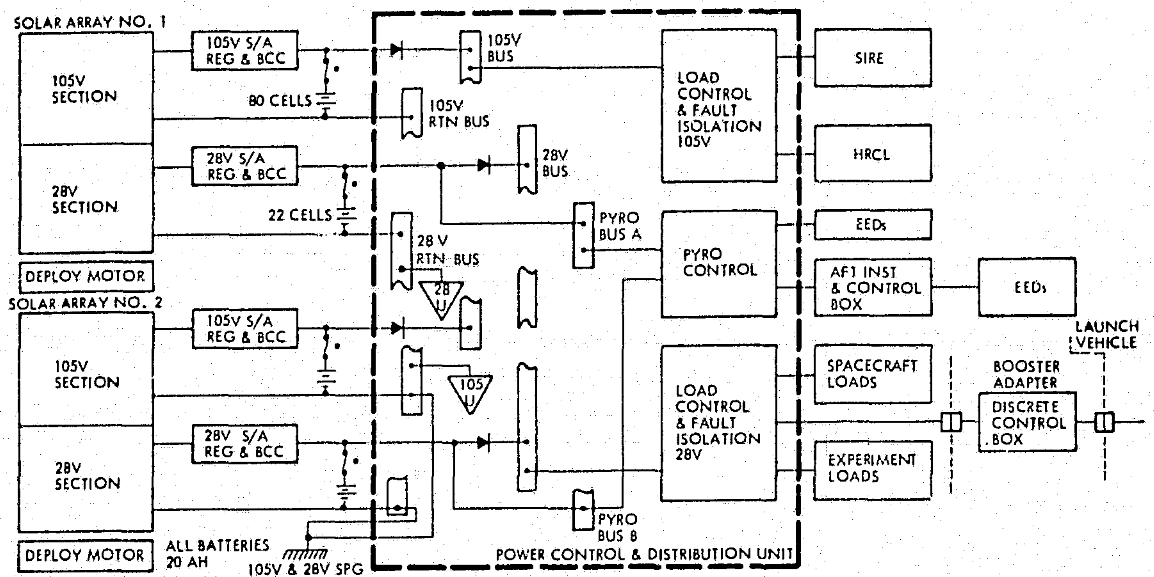

Figure B-6 SIRE Power System, Medium Orbit Application (Ref 43)

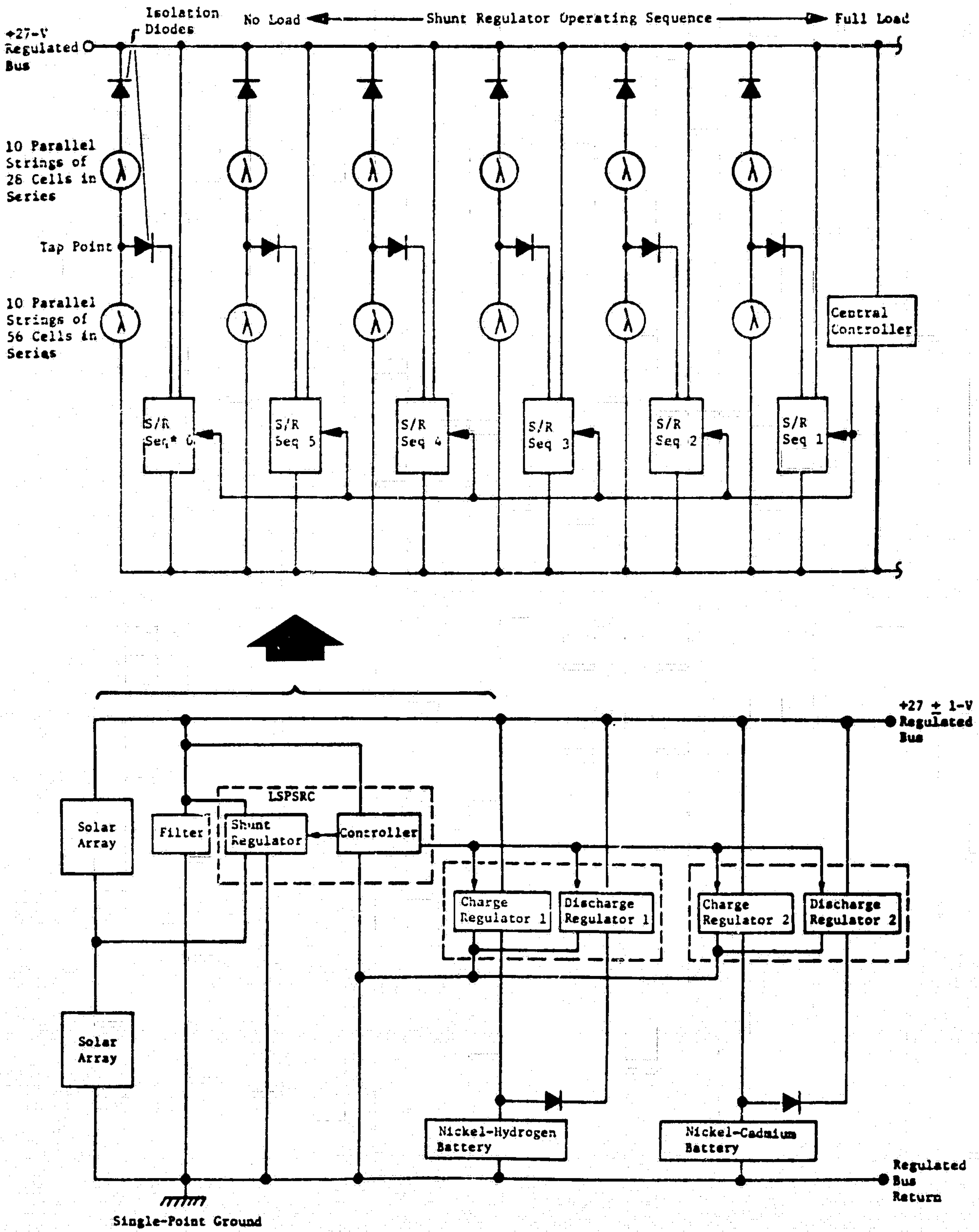

Figure B-7 NTS-2 Power System, Medium Orbit Application (Ref 42)

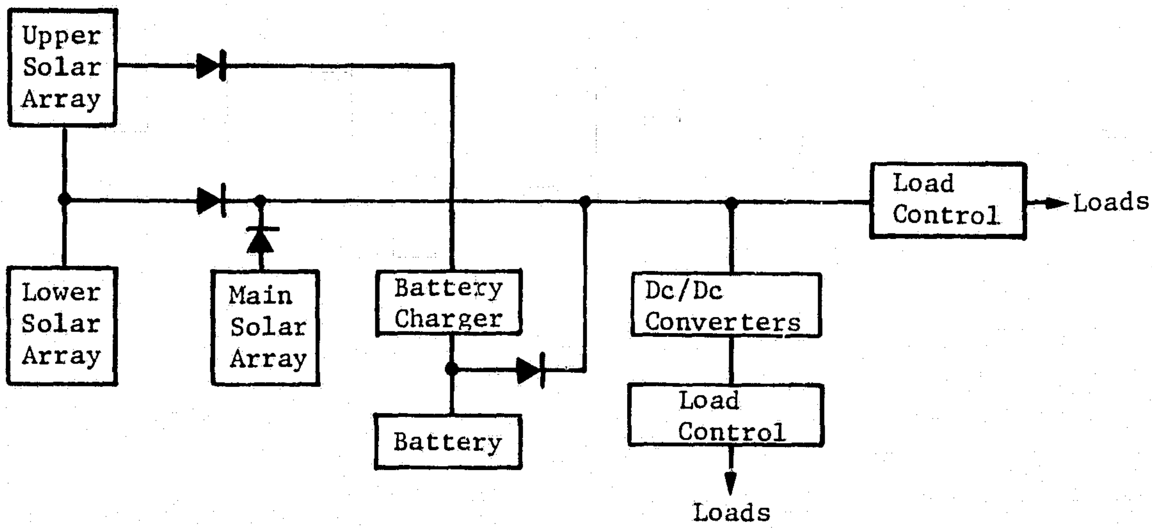

Figure B-8 FLTSATCOM Power Systems, GEO Application (Ref 44)

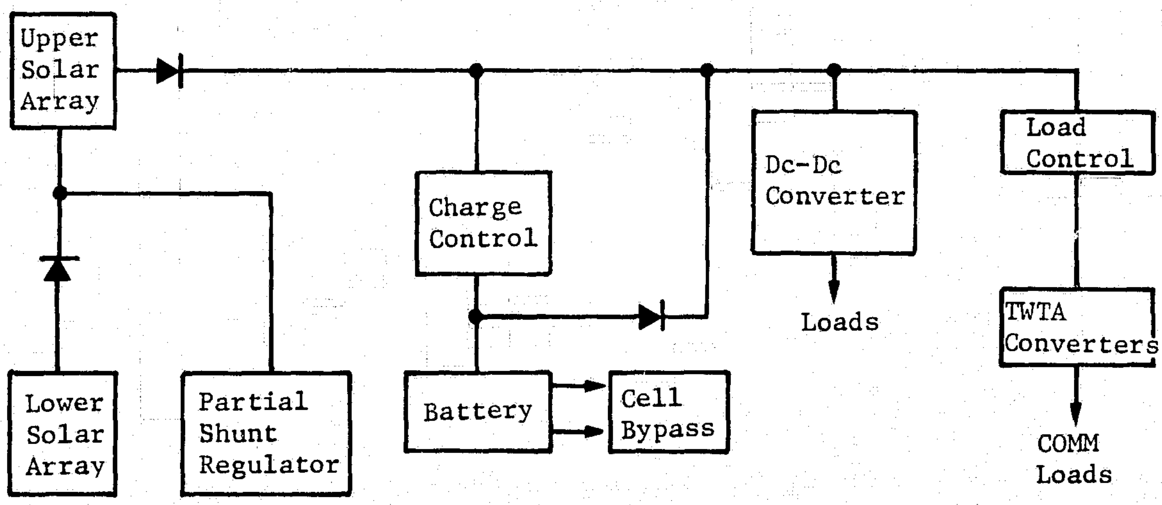

Figure B-9 INTELSAT-3 Power System, GEO Application (Ref 45)

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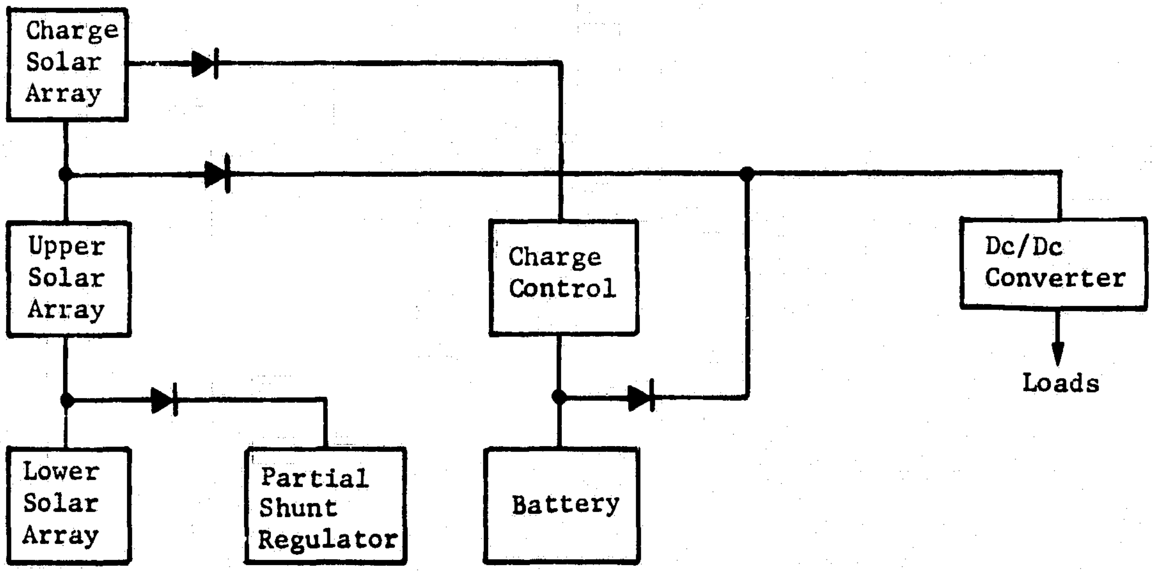
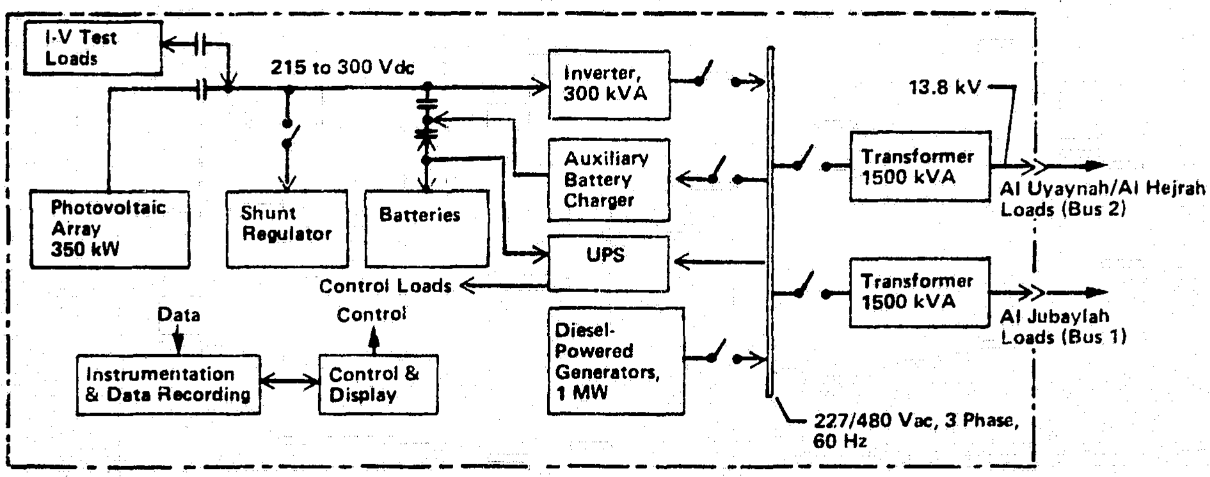


Figure B-10 INTELSAT-V Power System, GEO Application (Ref 46)



Features
- Integrated Array Bus Voltage and Battery Charge Control via Desk-Top Computer (HP9845)
- Four 120-Cell Lead Acid Batteries, 1.6 MW-h Total
- 240-Vdc Bus Tied to a Single 300-kVA Inverter
- 350-kW Point Focus (40X) Silicon Solar Array

Figure B-11
Simplified Block Diagram of 350-kW Photovoltaic Power System for Saudi Arabian Villages (Ref 14, 15)

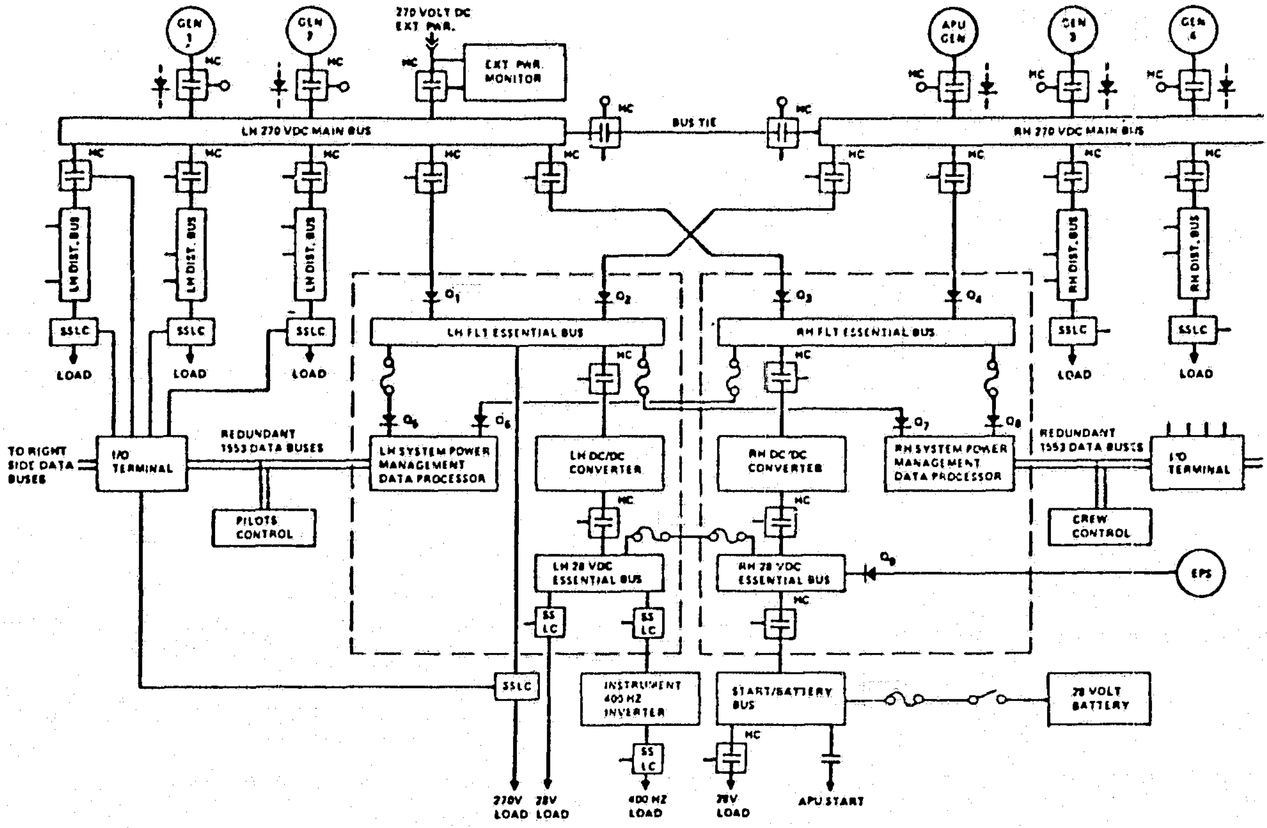


Figure B-12
Advanced Aircraft Electrical System, 270-Vdc Bus, Using Solid State Controllers (Ref 47)

LEVELS OF AUTONOMY

(Reproduced directly from Ref 12, pp 125-127.)

In performance of a space mission, four major policy goal categories have been identified. These are:

(1) Ground interaction reduction.

(2) Spacecraft integrity maintenance.

(3) Autonomous features transparency.

(4) On-board resource management.

The extent to which these goals have been accomplished to date has been through a mix of functions resident in either the space segment or the ground segment. Furthermore, the ground segment, as an integral part of the total system, has been responsible for accomplishing maintenance, navigation mission control, and payload data processing. Thus, only minimal spacecraft autonomy has been needed.

The levels of autonomy described in this appendix are used to define a step-wise increase in spacecraft autonomous capability. By proceeding through the levels, autonomous capability is increased in the space segment and dependency on the ground segment is reduced.

The levels of autonomy are described as follows:

Level 0. A design without redundant elements which meets all mission needs by operating without the on-board control of state parameters (such as rates and position). May respond to a prespecified vocabulary of external commands, but cannot store command sequences for future time-or event-dependent execution or validate external commands. (An open-loop, on-board system controlled from the ground.)

Level 1. Includes Level 0 but uses on-board devices to sense and control state parameters (such as rates and positions) in order to meet performance needs. Is capable of storing and executing a prespecified command sequence based on mission-critical time tags. Will respond to prespecified external commands, but cannot validate external commands. Functionally redundant modes may be available for a degraded-performance mission.

Level 2. Include Level 1 plus the use of block redundancy. Ground-controlled switching of spare resources is required. Uses cross-strapping techniques to minimize effect of critical command link (uplink) failure modes. Significant ground-operator interaction is required to restore operations after most faults if spare spacecraft resources are available.

Requires operator interaction for fault recovery. Is capable of storing and executing mission-critical events which are sensed on-board and may be independent of time.

Level 3. Includes Level 2 and is capable of sensing prespecified mission-critical fault conditions and performing predefined self-preserving (entering a safe-hold state) switching actions. Is capable of storing contingency or redundant software programs and being restored to normal performance (maintaining the command link with a single link fault) in the event of a failure. Timers may be used to protect resources. Requires ground operator interaction for fault recovery. In general, the failure to sense and/or execute the mission-critical event(s) will cause mission failure or loss of a major mission objective.

Level 4. Includes Level 3 but is also capable of executing prespecified and stored command sequences based on timing and/or sensing of mission events. Ground-initiated changes to command sequences may be checked on-board for syntactical errors (parity, sign, logic, time). Uses coding or other self-checking techniques to minimize the effects of internally generated data contamination for prespecified data transfers. Requires ground-operator interaction for fault recovery. In general, failure to sense and/or execute the mission event(s) or state-changes (excluding failure-induced state-changes) will cause mission failure or loss of a major mission objective.

Level 5. Includes Level 4 and is also autonomously fault-tolerant. Is capable of operating in the presence of faults specified a-priori by employing spare system resources, if available, or will maximize mission performance based upon available capability and/or available expendables (i.e., self-loading of contingency programs) without ground intervention.

Level 6. Includes Level 5 and is capable of functional commanding with on-board command-sequence generation and validation prior to execution. Functional commanding may include a high-level, pseudo-English language, spacecraft-system/operator communication and control capability.

Level 7. Includes Level 6 and is capable of autonomously responding to a changing external environment, defined a-priori, so as to preserve mission capability. The capability to change orbit in order to compensate for degradation or to protect the satellite from an external threat is included.

Level 8. Includes Level 7 and is capable of operating successfully within the presence of latent design errors which could cause loss of major mission objectives.

Level 9. Includes Level 8 and is capable of task deduction and internal reorganization based upon anticipated changes in the external environment. This situation is exemplified by multiple satellites operating in a cooperative mode. In the event of a satellite failure, remaining satellites would detect autonomously the condition (task deduction) and may generate and execute orbit-and spacecraft-reconfiguration commands.

Level 10. Includes Level 9 and is capable of internal reorganization and dynamic task deduction based on unspecified and unknown/unanticipated changes in external environment. The system will strive to maximize system utility. Thus, mission objectives should be adaptive and automatically reprogrammable. System resources should be maximized to preserve task adaptiveness.