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MULTI-KILOWATT MODULARIZED SPACECRAFT POWER PROCESSING SYSTEM DEVELOPMENT

FINAL REPORT

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GENERAL ELECTRIC COMPANY
SPACE DIVISION

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16. Abstract

A review of existing information pertaining to spacecraft power processing systems and equipment was accomplished with a view towards applicability to the modularization of multi-kilowatt power processors.

Power requirements for future spacecraft were determined from the NASA Mission Model-Shuttle Systems Payload Data Study which provided the limits for modular power equipment capabilities.

Three power processing systems were compared to evaluation criteria to select the system best suited for modularity. The shunt regulated direct energy transfer system was selected by this analysis for a conceptual design effort which produced equipment specifications, schematics, envelope drawings, and power module configurations.

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FOREWORD

This study program was sponsored under Contract NAS 3-17779 from the National Aeronautics and Space Administration.

This is a report of work performed solely and completely by Space Systems, Space Division, General Electric Company. The following individuals were the principal contributors:

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SUMMARY

This report describes the results of the study to develop a conceptual design for a multi-kilowatt modularized spacecraft power processing system that can be used on a majority of space vehicles to be launched during the shuttle era.

During the Task I effort, electrical power requirements of orbital spacecraft of the shuttle era were identified by mission type, power level, number of missions, and number of spacecraft in each mission. The data indicated that electrical power requirements can be met by solar array/battery systems, fuel cells for the short duration missions, and radioisotope thermoelectric generators for planetary missions.

A survey of power levels indicates that modularization on a multihundred watt level could economically satisfy most missions, considering 228 satellites of 41 different systems from 1973 through 1991.

The power processor functions were investigated, and it was determined that the shuttle power system influenced user equipment, that source power processing was best modularized with the source, and that load processors could best be integrated when located with the load. This concept of remote power processors for users is similar to the remote decoders and remote multiplexers placed within using subsystems.

Based on the results of Task I, Task II emphasis was placed on modularizing power systems which satisfy a greater total number of spacecraft than any other type. Specific attention was placed on the direct energy transfer approach which makes the solar array power available to the load at a regulated voltage level without an in-line power processing penalty. Total, partial, and sequenced source shunt regulation concepts were considered, and advantages and disadvantages of each were examined to provide data for a specific recommendation. The sequenced shunt regulator appeared most attractive, but has the drawback of a more complicated interface with the solar array.

Since energy storage is required with most solar array systems, configurations to accommodate batteries were investigated, both to provide the battery energy to the bus at a regulated level, and to replenish the battery energy. It appears that battery charging is mission dependent, since near-earth missions require relatively controlled, high rate charging. Geosynchronous missions can accommodate variable, low rate approaches that include separate battery charge solar arrays and simple current limiting resistors.

In parallel with these system studies, mechanical arrangements were reviewed to permit a modular approach. The lack of a standardized vehicle interface directed attention inward rather than outward. Earlier concepts considered for standardized repair and retrieval were not being pursued with vigor, since it appears that the modular power system should provide a simple, clean interface with the vehicle.

Since the majority of planned spacecraft will be shuttle launched, methods of accepting fuel cell power during the launch and ascent mode were studied, and at least one approach was

found acceptable without a significant penalty on the modular power system. This concept isolates the solar array and battery charger portion of the bus from the load side by a holding relay energized from the shuttle. Fuel cell power can now be applied to the isolated source bus, and the flow of power can be controlled by the series combination of battery charger and battery discharger. Since the shuttle is capable of near-earth orbits only, its vehicle payloads will have high-rate chargers fully capable of processing the fuel cell power without the penalty of additional hardware.

To simplify selection of an optimum bus configuration based on a numerical rating, all candidate systems were categorized as unregulated with remote regulation, central series regulated, or central shunt regulated. Thirteen considerations were identified as affecting system selection, and arbitrary weighting factors were developed based on past experience to permit optimization.

Based on this approach for selection of the optimum system, a comparative matrix was developed showing the relative ranking, weighting factors, and product of the two. The cumulative values indicated that the central shunt regulation concept was the preferred system configuration.

The first effort of Task III was to fully define requirements for functional blocks of the selected power system. The most complex interface is the battery charge regulator, and its requirements were developed in detail. This effort indicated that three different battery charge rates are required to satisfy variations in mission mode and orbital altitide.

The specific charge regulator approach is also used to interface with the shuttle fuel cell power source, and the battery charge regulator also conditions this unregulated power source so as to maintain bus voltage quality during orbit acquisition. Variations in orbital altitude also require two additional normal battery charge rates--a high rate for near earth, and a low rate for geostationary orbits.

The constant voltage system with central shunt regulation was explored (1) to provide protection against single point failures with the Electric Power System, (2) to provide excess peak power capability to clear a downstream fault without failure propagation, and (3) to detect lack of power margin and provide a signal to shed non-essential loads in the event of degraded power availability or excess user demand.

This implementation of autonomous power control results in a high piece part cound; an indicator of higher cost, lower reliability, and complexity. To circumvent these apparent detrimental attributes, designs utilizing bybrid microcircuits were explored for common logic functions. These circuits can be used in a power system at any power level, even though power stages may have to be custom designed. This results in a standard central control and logic circuit in microcircuit form with low parts count and low recurring cost. However, the inherent redundancy and autonomous control of power is reliably maintained.

CONCLUSIONS OF THE STUDY

The review of requirements for free flying spacecraft in the next 15 years has shown that with the exception of a small percentage of broadcast missions, power requirements are below 2 kilo watts, with 65% of the spacecraft requiring less than 600 watts. Therefore, a modular power processing system need not be multi kilowatt, but growth to 2 kilowatts is needed.

Developing a modularized universal spacecraft power processing system will result in economic savings as non-recurring design and development costs are eliminated from each spacecraft system. Standardization of any design will be beneficial in this respect, but this study showed that a shunt regulated direct energy transfer power system design will provide the greatest advantages compared to other designs using unregulated power buses.

The basic modular units of the power system are shown in Figure 1. The equipment consists of:

- A partial shunt regulated solar array that is modularized in 500 watt segments.
- A highly redundant central control that governs the operation of the power equipment to maintain bus regulation.
- A 16 cell nickel cadmium battery, using 20 ampere-hour capacity standard cells.
- A dedicated battery charger that 'bucks' the regulated bus voltage down to the battery terminal voltage while limiting the maximum charge current to 5 amperes (c/4).
- A pulse width modulated boost discharge regulator with a 600 watt output capability that can operate in parallel with like units as required to satisfy eclipse load demands.

When selecting a power system design, and comparing it with others, all the power processing functions must be considered whether they are within the boundaries of the power system or are provided by the user loads. That is to say, when analyzing an unregulated bus system, the regulators located at the loads must be considered a power system item even though historically their penalty has been allocated to the load.

The design effort of this study has produced modular concepts at the circuit level, equipment level, and system level. Design rationale developed during the study and failure modes analysis has provided preliminary design specifications for the power processing equipment. These can be used for the next logical phase of development which is breadboard hardware. Thermal analysis has verified that the power module layouts can be thermally controlled by passive radiator designs.

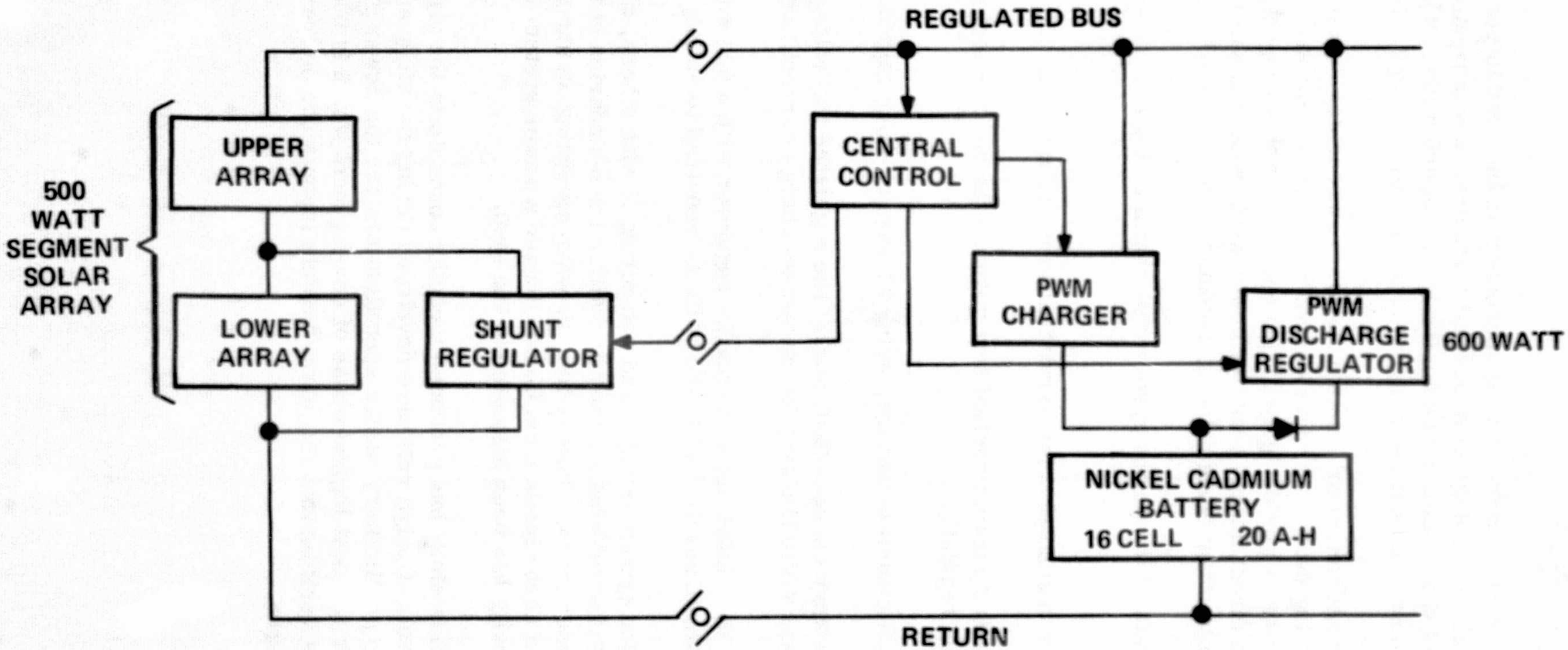

Figure 1. Simplified Shunt Regulation – Direct Energy Transfer Power System

INTRODUCTION

This contract covers a Phase I program to develop a multi-kilowatt modularized spacecraft power processing system. The program, initiated on 19 October 1973, was organized into the following tasks:

1. Review present and anticipated electrical loads of earth orbital and solar orbital spacecraft, and existing applicable information on modularization of multi-kilowatt power processors.

2. Select system concepts and sub-element standardization concepts which demonstrate definite inherent advantages, and evaluate ac and dc distribution systems, central and remote power processor modules, and standard trade study parameters.

3. Create a conceptual modularized design and provide engineering documentation.

TASK I

REVIEW OF INFORMATION

APPRAISAL OF POWER SYSTEM TECHNOLOGY

A review of historical documents was performed at the onset of this project to determine what previous efforts might be applicable to power processor modularity. This data has been collected and summarized in the format shown on Table 1. The source of information is identified, a brief description of the material is included, and particular subjects addressed in the literature are tabulated as shown. The complete listing of the literature search has been included as Appendix I-1.

TABLE 1. LITERATURE REVIEW

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/ SOURCES	BATTERIES CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT COST
2.	SOLAR ARRAY COST REDUCTIONS D. T. BERNATOWICZ, NASA LEWIS 1972 1973	HISTORICAL SOLAR ARRAY COSTS ARE PRESENTED FOR A BROAD CROSS SECTION OF FLIGHT PROJECTS OVER THE PAST 10 YEARS COVERING A RANGE OF AVERAGE OUTPUT FROM LESS THAN 100 WATTS TO 4 KW.	x					x					x
3.	ANALYSIS AND PERFORMANCE OF PARALLELING CIRCUITS FOR MODULAR INVERTER-CONVERTER SYSTEMS ARTHUR G. BIRCHENOUGH FRANCIS GOURASH NASA LEWIS MARCH 1972 NASA-TN-D-6713	PARALLELING CIRCUITS FOR ANY NUMBER OF INVERTERS (OR CONVERTERS) ARE ANALYZED AND TEST RESULTS PRESENTED. GOOD CORRESPONDANCE BETWEEN THEORETICAL AND ACTUAL PERFORMANCE IS SHOWN. TRANSIENT LOAD SHARING IS NOT CONSIDERED, BUT EFFECTS OF OUTPUT FILTER PARAMETERS IS NOTED.			x								
4.	SOLID STATE POWER CONTROLLERS (PROCEEDINGS OF THE SPACE SHUTTLE INTEGRATED ELECTRONICS CONFERENCE, VOLUME 2 JACK C. BOYKIN AND WILLIAM C. STAGG NASA-MSC AND DONALD E. WILLIAMS, NASA-MSFC 1971 N71-35039 (NASA-TM-X-58061)	ADVANTAGES OF SOLID STATE POWER CONTROLLERS ARE DISCUSSED, ESPECIALLY AS APPLIED TO DISTRIBUTED BUS REMOTE CONTROLLED SYSTEMS. THE NEED FOR FIRM DEFINITION OF REQUIREMENTS FOR THESE CONTROLLERS IS IDENTIFIED, SO THAT QUALIFIED HARDWARE CAN BE DEVELOPED.				x		x					x

Review of documentation on past power systems, and various projections and claims for future power systems yields some areas of agreement, and yet several areas of disagreement or doubt.

SOURCES

It is generally agreed that spacecraft prime power sources will continue to be photovoltaic solar arrays or radio-isotope thermoelectric generators (RTG's) for missions in excess of 30 days. For short term missions, fuel cells will still have great usage. Developmental sources, such as the various Brayton cycle engines, are far from the maturity required of long-life spacecraft. It sould be noted that solar arrays, RTG's, and fuel cells are all sources compatible with simple shunt regulators as well as series (in-line) switching regulators, both techniques having a high state of development and flight history. By comparison, the rotating machinery sources, is exemplified by the Brayton and organic-Rankine engines, have not yet reached the total "hands-off" status of operation.

REGULATORS

Each of the non-rotating sources have wide output variations which must be handled by a voltage regulator, but many regulation refinements have been built and satisfactorily flown. The regulators can be classed in two broad catagories: Dissipative & non-dissipative. Most viable series (in-line) regulators employ non-dissipative techniques, wherein on-off switching duty-cycle is varied to provide a constant integrated output voltage. To overcome response problems inherent in the large output filters required, two loop designs have been employed, where pulse current is also sensed as a "lead" function. Whether the regulator is configured as a "boost", "buck", or "boost-buck" type, the control techniques are common to all.

In the case of shunt regulators, non-dissipative are in the minority. Most are designed as linear-modulated analog loads, although some "hard-switching" (digital) shunt regulators have been flown. Bus noise appears to be the primary problem with digital shunt regulators, especially in cases where the magnitude of the bus load change is sufficient to cause the digital regulator to switch many shunt elements. Digital shunt regulators do, however, tend to equalize thermal dissipation requirements over wide load and source variations.

DISTRIBUTION

Two main areas of power distribution are being given considerable attention: Voltage level and switching techniques.

Voltage

The most mature voltage level, in terms of available hardware and flight experience is at 28 volts DC. Other levels have been studied in the interests of conserving wire (weight) and losses (prime power source). The most-often proposed DC voltage level for future large-scale spacecraft is 100 to 130vdc. Unfortunately, there is a lack of qualified components for this level, which range from corona-free connectors and other wiring devices to semiconductors with reasonable characteristics and margins.

AC distribution also has its proponents, who base their decision on distribution of high-frequency power, and requiring each user-load to provide the transformer-rectifier-regulator components to meet his needs. This concept is analogous to commercial power systems, wherein each user employs "power supplies" to meet his needs, except that weight savings are realized in magnetic and filter components by use of high frequencies. Offsetting this "advantage" are the EMC problems of interaction and containment of the harmonics originating from the non-sinusoidal waveforms used. DC distribution proponents are quick to point out that the high frequency-related "advantages" are realized anyway with DC, since each user load contains DC/DC converters running at these same frequencies and power levels!

Until qualified high-voltage components are available, with usage-inspired confidence, 28 to 35 volt DC distribution appears to be the choice for some time to come.

Control

Use of solid-state switches is advocated by many, for the purpose of greater control versatility and increased reliability. Inherently more reliable than electromechanical switching devices solid-state permits control of switching speeds, and times thereby effectively reducing sources of EMI. Performance of control logic (overload detection, load balance, etc) is best implemented by solid-state means, and the intent is to further the expansion into the load switching as well. Full utilization of these concepts results in distributed-bus systems, with remote control of all loads and bus/feeder switching by means of a "data bus" under computer control.

The need for back-up operation in the event of component failure can be covered to an extent by computer control, but "manual" operation (as well as by comm. link) is a necessary requirement. The advantages of remote control/computer control are an overriding consideration only as spacecraft power system size and complexity increases. Below some break-even point, a cost/weight penality accrues for these methods.

TOTAL POWER SYSTEM

The choice and design of a particular power system for missions in excess of one month life is governed very strongly by total power, size and complexity considerations. In the foreseeable future, all long mission spacecraft will probably continue to have "limp" sources, which can be properly controlled by shunt regulators. Battery discharge controllers appear to be well developed in their present non-dissipative form. Regulated DC bus distribution is to be most commonly employed, with optional computer/remote control, when warranted by increased size and complexity, if failure modes and their effects can be fully predicted.

REVIEW OF FUTURE SPACECRAFT POWER REQUIREMENTS

To determine the needs of future spacecraft and to provide limitations to this study, the NASA Mission Model - Shuttle Systems Payload Data (SSPD) was used as a reference for automated spacecraft into the 1990's. The term "automated" implies that the spacecraft is self-sufficient after the shuttle launch, i.e., it has its own power system as opposed to the "sortie" spacecraft that remains with the shuttle and uses shuttle power throughout its mission.

A sample of the SSPD data sheets which provided the power information is shown on Table 2. Included is the total power, type of power source, and energy storage if required. The complete summary of all automated spacecraft power needs is provided as Appendix I-2. A significant requirement used in the subsequent study was that the most desired voltage for power input was specified as +28 VDC.

The SSPD identified 228 spacecraft of which 90% required solar arrays and batteries as the power source. About 5% were nuclear powered and the remaining 5% identified other power sources. Table 3 shows the solar array powered missions which constitute the largest percentage of all spacecraft.

TABLE 2. SSPD SHEET A-4

AUTOMATED PAYLOAD
SUPPORTING SUBSYSTEMS

1-6

Data Sheet No. A-4 b
Payload No. AS-01-A
Date 16 Apr. 1973 Rev. 29 Jun 73, 16 Oct 73

Payload Name Large Space Telescope

Subsystem: 1. Item No.	Subsystem: 2. Name	3. Type (e.g., active or passive environmental control, star trackers or horizon sensors, hydrazine or cold gas, solar cell and battery or RTG)	4. Performance (e.g., accuracy of pointing & stabilization, propulsion and RCS thruster levels, electrical power avg and peak levels)	5. Weight, kg (lb)	6. Remarks
SM-107	Electrical	Solar arrays, batteries, electrical distribution	Capability 1.5kw avg. at 28 vdc ±2%, 1.9 kW peak including provisions for redundancy and 0.22 kw growth margin; 5.4 kWh batteries.	686 (1511)	335w avg + 146 Wp in orbit + 911 W for mission equip = 1245W
SM-108	Other (Specify)	Pressure, contamination control and monitoring.	10,000/100,000 zoned system	29 (64)	Protective pressurization and control of OTA volume during prelaunch, ascent, on orbit preparation, reentry, and post landing with inert clean gas.
			Supporting Subsystems module total	2686 (5922)	
			LST = $W_{OTA} + W_{SI_{struct}} + W_{SI} + W_{SSM}$ + 20% conting. = 4450 (9812) - 150 (331) + 1003 (2[illegible]) + 2686 (5922) + 1657 (3655) *	9946 (21931)*	V 180.5 m² (6430 ft²); See Note 3
SM-109	Protective cover for LST	Non contaminating plastic cover or shield	Minimizes entry of contaminants during ascent or reentry	200 (441)	
SM-110	SSM remote checkout unit	Checkout and test of SSM elec power, comm/data handling (TTC), caution and warning, guidance/navigation/stabilization, environmental control, attitude control, contamination control functions and performance	Enables check, activation, test of supporting subsystems functions and performance by subsystem as well as checking integrity of command, monitor, and data output as well as power circuits used in conjunction with standard equipment at payload monitoring stations.	45.4 (100)	0.61 m (2 ft) x 0.915 m (3 ft) x 0.75 m (2.5 ft); interface, support, test & control circuits unit in shuttle orbiter cabin at payload monitor station (requires 100w to support checkout, control & test)(1/2 of Item on page A-11)
SM-111	Contamination control support equip.	Filters, fans, controls 0.65 x 0.67 x 1.17m/ 870w	0.189 m^3/sec (400 cfm) to SIP, 0.283 m^3/sec (600 cfm) to SSM Payload	164 (362)	Not required on ground maintained version except for depressurization or pressurizing LST during ascent and reentry.
AS-190	Experiment Equipment	Checkout, test monitor, control, for OTA and scientific instruments		45.4 (100)	0.61m (2 ft) x 0.915 (3 ft) x 0.75m (2.5 ft), interface, support, test and control optical telescope and instrument circuits (1/2 of Item on page A-11)

Auxiliary Equipment + 454.8 (1,003)
Basic Telescope + Instrs = 9,946 (21,931)
Grand Total = 10,401.0 (22,934)

NOTES:

1. Does design have capability for space docking? Yes X, No ____.
2. Does TTC subsystem assume use of TDRS? Yes X, No ____.
3. LST from TMX-64726 = 21567 lbs; LST + mirror weight increase is anticipated to change weight up 25087 lb maximum in order to reduce cost and development risk.

SSPD (A-4) 3-12-73

Prepared by: E. Siari

TABLE 3. AUTOMATED PAYLOAD SOLAR POWER REQUIREMENTS

DISCIPLINE	NUMBER OF FLIGHTS *	SOLAR ARRAYS NO./TYPE	POWER RANGE (WATTS)	ORBIT LIFE (YEARS)
ASTRONOMY & SOLAR PHYSICS	11	11/ROTATING (2 PANELS)	150-1500	2-3
HIGH ENERGY ASTROPHYSICS	8	8/ROTATING (2 PANELS)	150-1200	2-5
ATMOSPHERIC & SPACE PHYSICS	24	2/ROTATING (4 PANELS) 22 BODY MOUNTED	200-500	1-3
EARTH OBSERVATIONS	87	69/ROTATING (2 PANELS) 18 BODY MOUNTED	140-960 140	2-5
EARTH & OCEAN PHYSICS	15	13 BODY MOUNTED 2 NO ELECT. S/S	90-620	0.5-5
COMMUNICATIONS/NAVIGATION	69	34 ROLL-OUT 7 BODY MOUNTED 28 ROTATING (2 PANELS)	300-5000	5-10
TOTALS:		116 ROTATING (2 PANELS) 2 ROTATING (4 PANELS) 60 BODY MOUNTED 34 ROLL-OUT ARRAYS		

* NUMBER OF FLIGHTS BASED ON JUNE 1973 MISSION MODEL

The data of Appendix I-2 was used to develop Figure 2 which shows the distribution of the numbers of spacecraft at various power levels. This shows that with the exception of "poles" at 4.5 and 6 kilowatts, all power requirements are below 2 kilowatts with 65 percent requiring less than 600 watts. The "poles" were considered to be unique applications and therefore not a significant factor in the subsequent study.

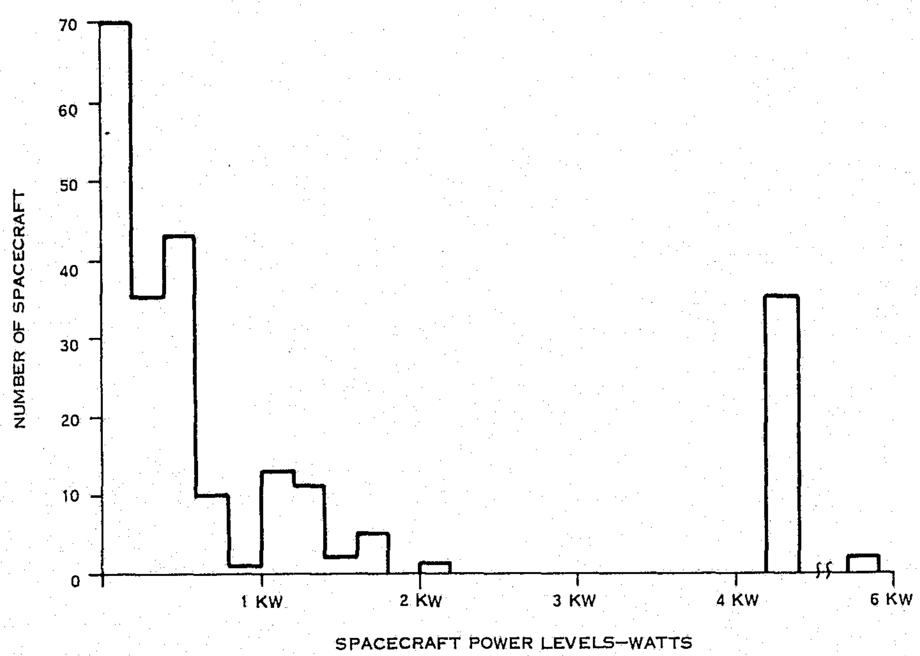

Figure 2. Spacecraft Power Requirements

TASK II

SELECTION OF SYSTEM CONCEPT

MATRIX PRIORITY RATING ANALYSIS TECHNIQUE

To select the power system best suited for modularity, a matrix priority rating system was used. This technique allows comparison of candidate systems by numerical analysis. First, a list of criteria is developed and these criteria are evaluated as to their relative importance. A total of 100 points is divided among the criteria as desired by the evaluator to give them a Weighting Factor.

Next, the candidate systems are measured with respect to each criterion and ranked on a scale of 1 to 10.

The final step in the evaluation is to develop the selection matrix where the rank of each candidate system is multiplied by the Weighting Factor of each criterion. The sum of these products determines the best candidate approach.

This technique is shown in diagram form in Figure 3. Table 4 presents an example of the technique. In the example, two alternative systems (A & B) are to be evaluated with respect to three criteria. The 100 criteria importance points are allocated as shown. Next, the two systems are ranked with respect to each criterion. In the example, system B is considered

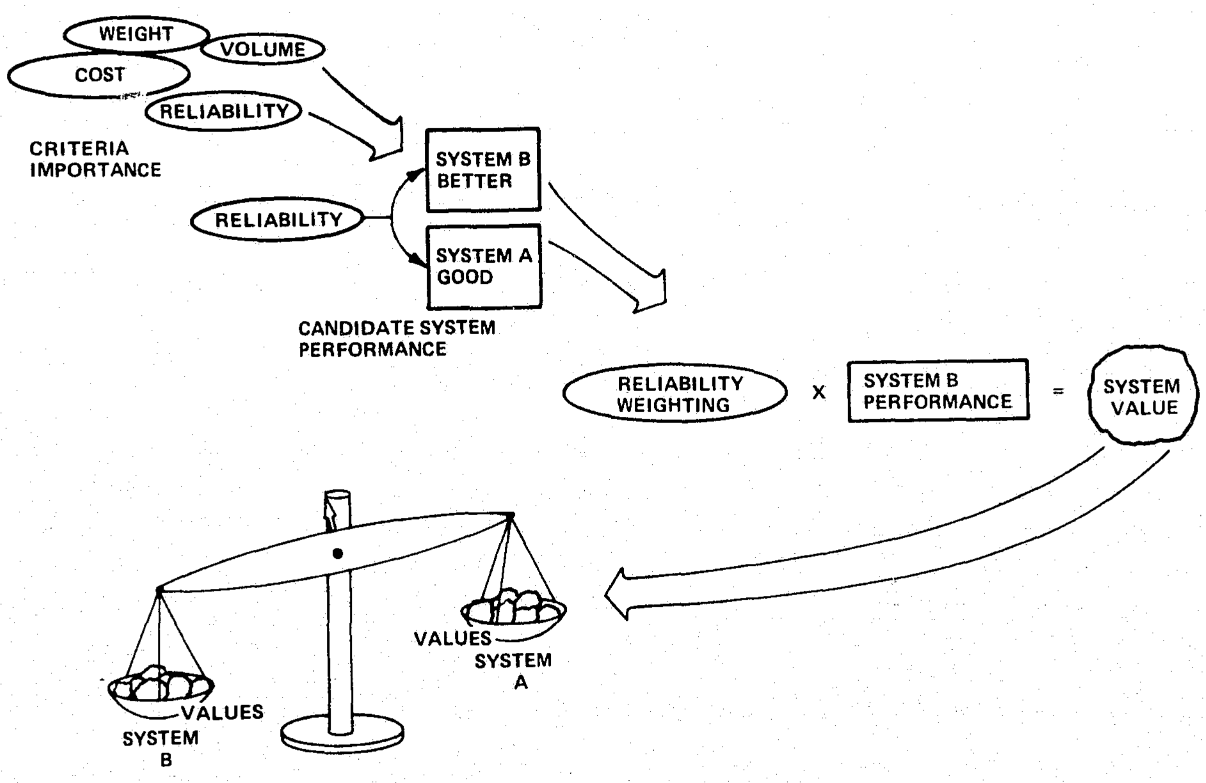

Figure 3. Analysis Technique

TABLE 4. ANALYSIS TECHNIQUES MATRIX PRIORITY RATING SYSTEM

EXAMPLE

- TWO ALTERNATIVES ARE EVALUATED WITH RESPECT TO THREE CRITERIA.
- CRITERION IMPORTANCE WEIGHT (100 POINTS TOTAL)

PERFORMANCE	50 POINTS
WEIGHT	25 POINTS
POWER	25 POINTS

- RANK OF TWO ALTERNATES (SCALE FROM 1 TO 10)

	RANK	
CRITERION	SYSTEM A	SYSTEM B
PERFORMANCE	2	7
WEIGHT	8	2
POWER	8	2

- SELECTION MATRIX

CRITERION	WEIGHTING FACTOR	SYSTEM A RANK	SYSTEM A RXWF	SYSTEM B RANK	SYSTEM B RXWF
PERFORMANCE	50	2	100	7	350
WEIGHT	25	8	200	2	50
POWER	25	8	200	2	50
TOTAL VALUE			500		450

better than A with respect to performance, but not as good in the other two criteria of weight and power. In the last step, the criterion weighting factor is multiplied by the rank, the products summed, and the system A which has the highest total value is the choice for this example.

DEVELOPING THE EVALUATION CRITERIA

Thirteen criteria, as shown in Figure 4 were selected as applicable to a modularized multi-kilowatt power system.

After identifying the criteria, the next step was to provide definitions in order to remove any ambiguities. Table 5 provides the description of each criterion.

To provide a weighting or a measure of importance to each criterion, the procedure of Figure 5 was used. The descriptions and some basic assumptions pertaining to the use of the power systems as shown in Table 6 were given to six power subsystem engineers and they were asked to distribute 100 points among the 13 criteria. Table 7 contains the results of this voting. As can be seen, there were some disagreements as to relative importance, but generally 5 out of 6 agreed to each relative value placed on the criteria. The average weighting of Table 7 was used in the subsequent analysis.

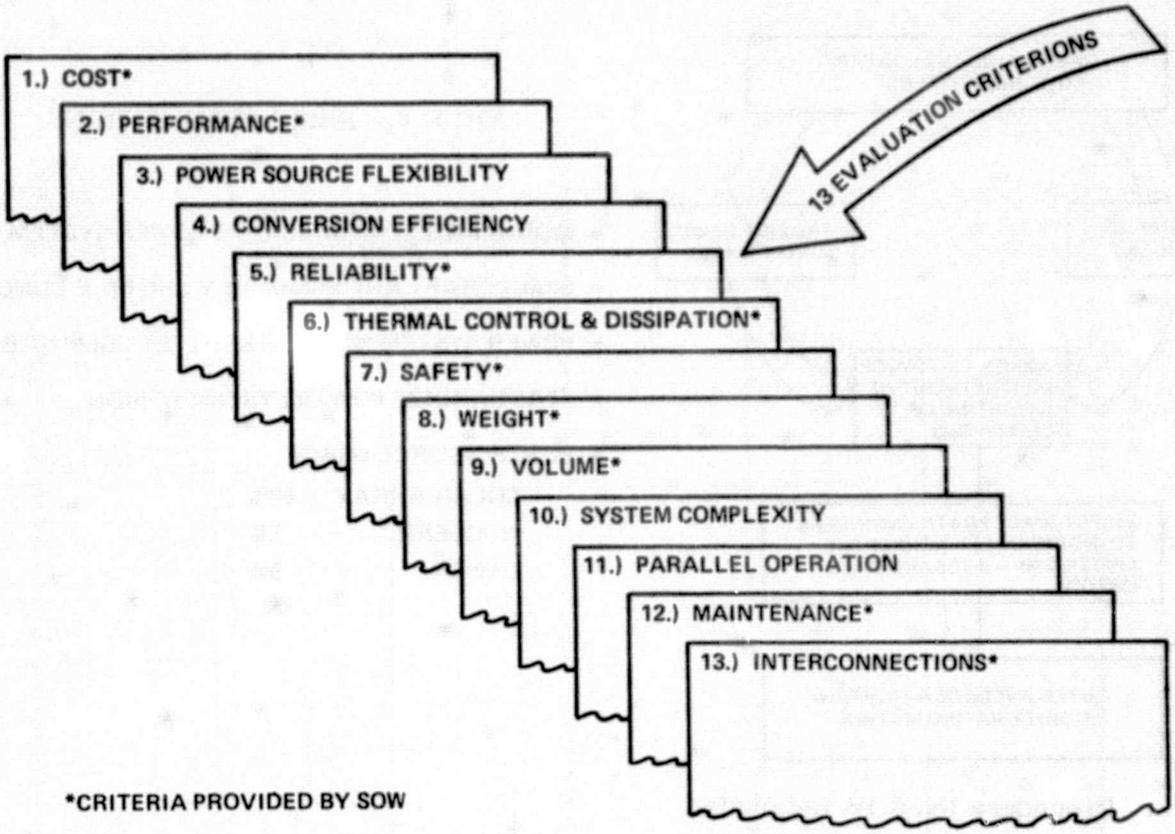

Figure 4. Evaluation Criteria

TABLE 5. CRITERIA DESCRIPTION

1. COST RECURRING DOLLARS PER SYSTEM. 2. PERFORMANCE RIPPLE, DYNAMIC IMPEDANCE, TRANSIENT RESPONSE & LOAD REGULATION 3. POWER SOURCE FLEXIBILITY THE CAPABILITY OF SELECTED POWER PROCESSING SYSTEMS TO INTERFACE & CONTROL POWER SOURCES (EG. SOLAR ARRAYS, RTG'S, FUEL CELLS & SHUTTLE POWER. 4. CONVERSION EFFICIENCY $\text{EFFICIENCY} = \frac{\text{(LOAD POWER REQUIRED)}}{\text{(LOAD POWER REQUIRED + SYSTEM LOSSES)}}$ 5. RELIABILITY THE PROBABILITY OF MISSION SUCCESS. 6. THERMAL DISSIPATION & CONTROL THE CAPABILITY OF THE POWER PROCESSING SYSTEM TO MINIMIZE DISSIPATED POWER AND CONTROL SAME.	7. SAFETY THE DEGREE TO WHICH HAZZARDS ARE MINIMIZED. 8. WEIGHT 9. VOLUME 10. SYSTEM COMPLEXITY THE QUANTITY OF BASIC FUNCTIONAL ELEMENTS OR CIRCUIT PIECE PARTS REQUIRED TO ASSEMBLY A POWER PROCESSING SYSTEM. 11. PARALLEL OPERATION THE ABILITY TO INCREASE SYSTEM CAPACITY WITH STANDARD MODULES. 12. MAINTENANCE THE FREQUENCY OF REPAIR OR REPLACEMENT. 13. INTERCONNECTIONS THE QUANTITY OF FUNCTIONAL MODULE INTERFACING WIRES.

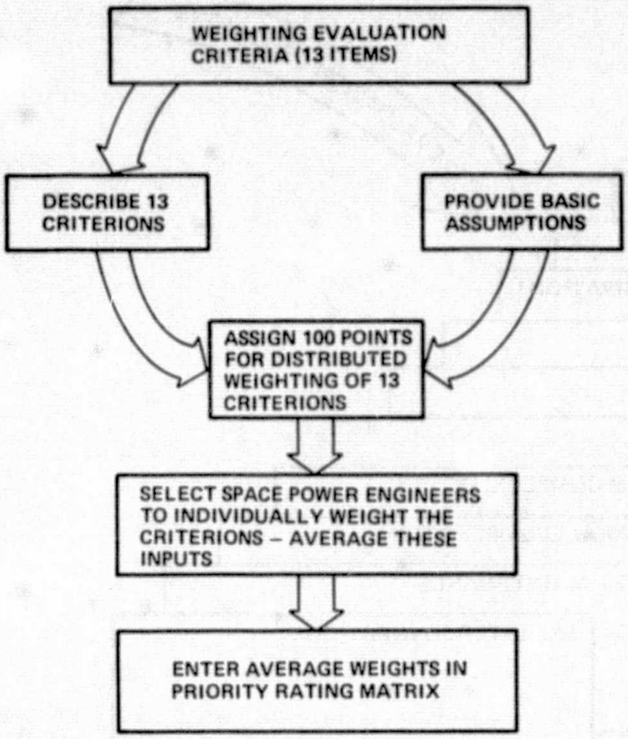

Figure 5. Procedure Used To Determine Criteria Weight

TABLE 6. BASIC ASSUMPTIONS

- MODULARIZED SPACECRAFT POWER SYSTEM
- SPACECRAFT ARE PRIMARILY SHUTTLE LUNCHED
- POWER SYSTEM MUST BE SHUTTLE COMPATIBLE
- STANDARDIZE POWER SYSTEM DESIGN
- POWER SOURCE USAGE

SOLAR ARRAY	90%
NUCLEAR	5%
OTHER	5%

TABLE 7. NORMALIZED VOTING SUMMARY

CRITERIA	WEIGHTING BY SPACE POWER ENGINEERS A	B	C	D	E	F	AVERAGE WEIGHTING
COST	17	16	16	19	19	20	18
PERFORMANCE	15	13	10	13	13	10	12
POWER SOURCE FLEXIBILITY	7	8	4	7	7	10	7
CONVERSION EFFICIENCY	11	4	16	13	16	10	12
RELIABILITY	9	13	11	13	9	10	11
THERMAL CONTROL & DISSIPATION	7	5	4	6	6	8	6
SAFETY	7	9	4	5	3	5	5
WEIGHT	4	4	8	1	3	5	4
VOLUME	3	4	8	3	3	2.5	4
SYSTEM COMPLEXITY	8	6	12	8	10	10	9
PARALLEL OPERATION	3	12	4	6	6	5	6
MAINTENANCE	5	2	5	4	2	2.5	3
INTERCONNECTIONS	4	4	2	3	3	2	3
							100 POINTS

SYSTEM DESIGN CANDIDATES

In developing the system design candidates, several ground rules and assumptions were made which would influence their complexity.

Figure 6 shows pictorially the following:

- The system will contain two batteries to allow for one failure.
- The system must be compatible with the Space Shuttle fuel cell or condition this source for acceptance by the power system.
- The basic load level is 400 watts*, and consists of a redundant Telemetry Tracking and Command Subsystem as well as a redundant Attitude Control Subsystem.
- The system must provide redundant regulation for the redundant subsystem and fail-safe regulation for the payloads.
- A system that contains remote regulators (regulation performed at the user load) would not operate all payloads from a common regulator as the loss of that regulator would cause loss of all payloads. It is assumed that three regulators would be provided and the payloads would be divided among them.

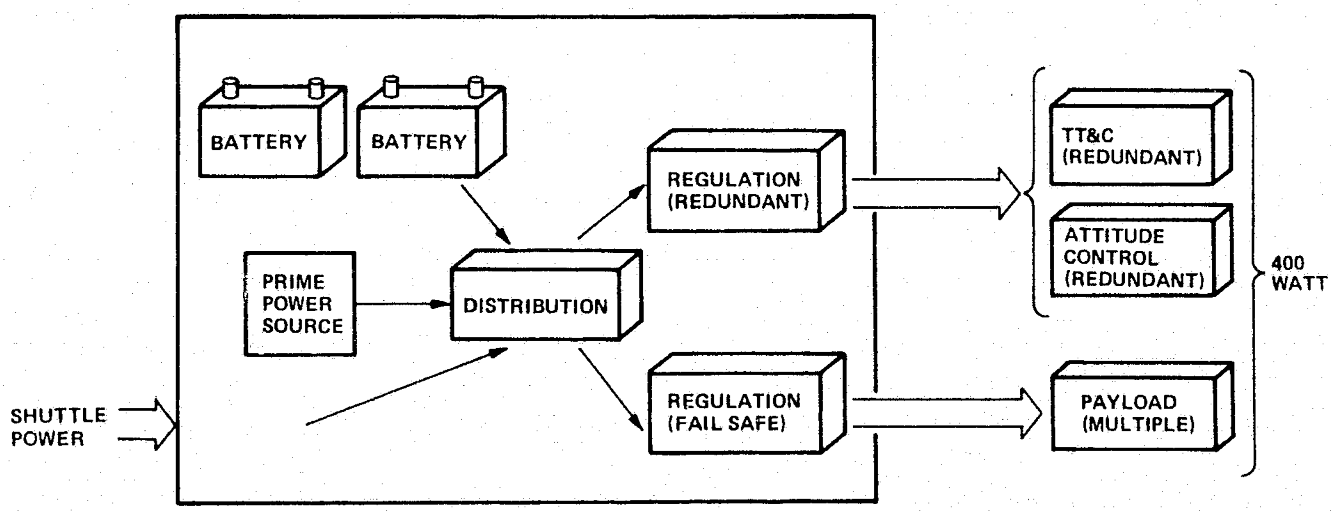

Figure 6. General Assumptions

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*Results of Task I show that 50% of the spacecraft are 400 watts or less.

Using these ground rules and assumptions, three power systems were developed. They are: Remote Regulated, Series Regulated, and Shunt Regulated systems.

The remote regulated system developes and distributes an unregulated bus to the user loads where regulation is performed by the user requipment. However, when comparing this type of power system to others that provide a regulated bus, these load regulators must be considered as part of the power system. The remote regulated power system shown in Figure 7 is used for comparison with the other candidate systems.

The voltage limiter prohibits the voltage excursion of a cold solar array and provides the upper limit of 35 VDC on the power bus. The lower limit of 22 VDC is determined by the minimum discharge voltage of the batteries. The charge regulator is a series regulator with current and battery voltage limits. A power distribution unit provides the switching and distribution of the power bus to the loads. The load regulators are pulse width modulated switching regulators with the redundant units operating in active redundancy (both ON).

The series regulated system of Figure 8 contains the same equipment as the remote system except the remote regulators are replaced by two central series regulators and as one is in standby, a failure detector is provided to cause switch over when required.

Standby redundancy is used in lieu of active redundancy to improve reliability as a dormant unit has about one tenth the failure rate of an active one. Also, the power losses associated with this fairly large series regulator (large with respect to the individual subsystem regulators of the remote regulated system) are eliminated by leaving it off until needed.

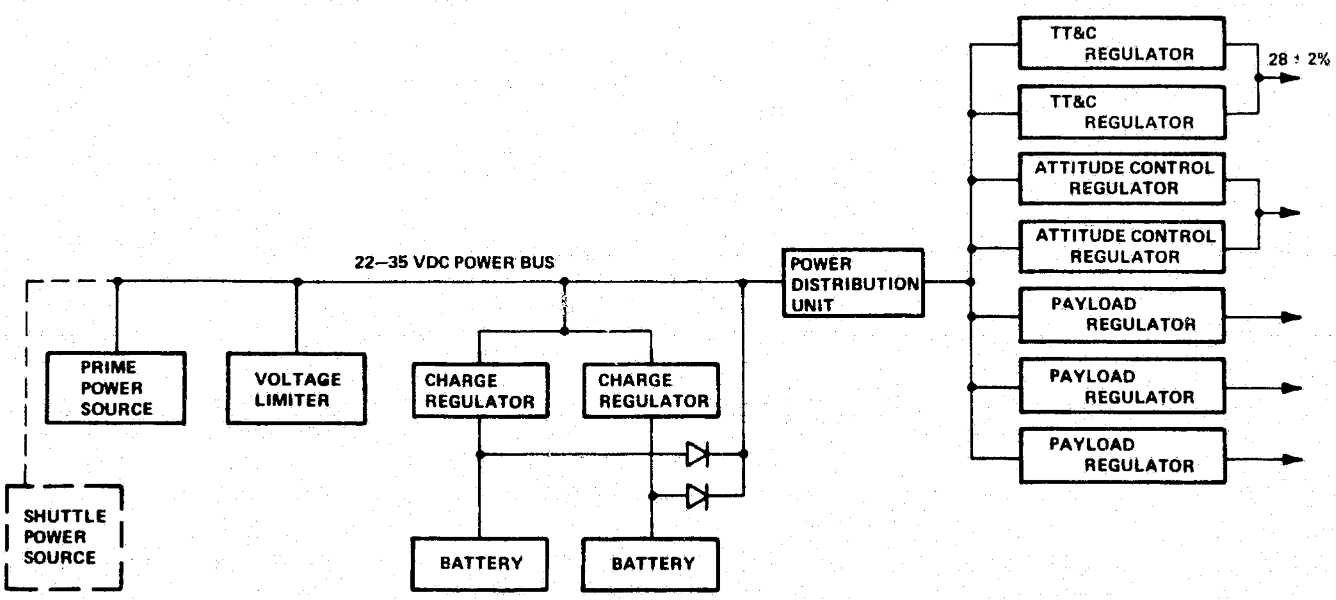


Figure 7. Remote Regulated Power System Block Diagram

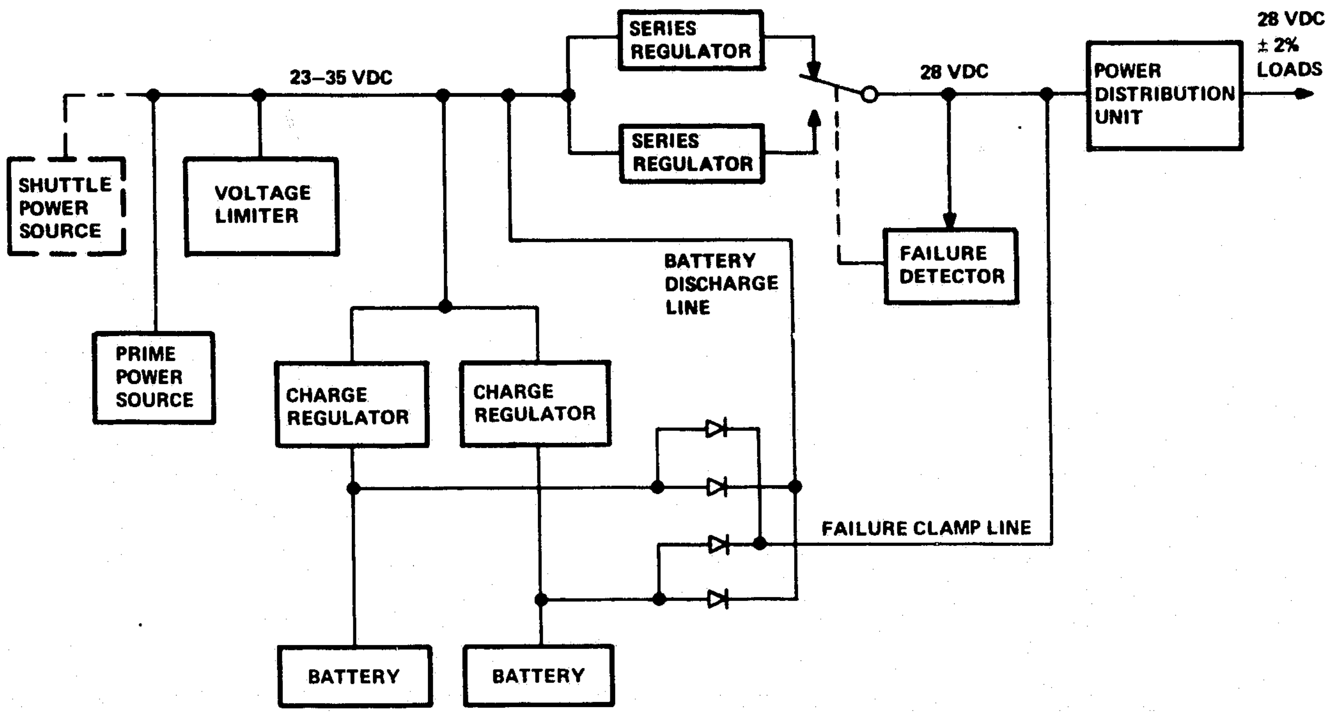


Figure 8. Series Regulated Power System Block Diagram

Figure 9 presents the shunt regulated system approach for comparison with the previous concepts. In this system, the regulating element has been relocated and is no longer between the solar array and the spacecraft loads, but is across the solar array. This system requires battery discharge regulators in addition to the shunt regulator. Recent designs of these discharge regulators have used techniques that allow standby operation and switch-over without the need for the separate failure detector shown on Figure 9. However, to be fair to the series regulated system, these same techniques could be used for switch over of it's standby unit. For comparison purposes, a common approach must be used, and the failure detector concept was selected because of application experience with the series regulator design.

EVALUATION AND RANKING OF CANDIDATE SYSTEMS

The process of ranking the three systems was to grade them on a scale of 0 to 10 with the middle system normalized to a 5. The other two systems grades are based on the percentage difference from the middle system, with the best system receiving a grade higher than 5 and the worst lower than 5.

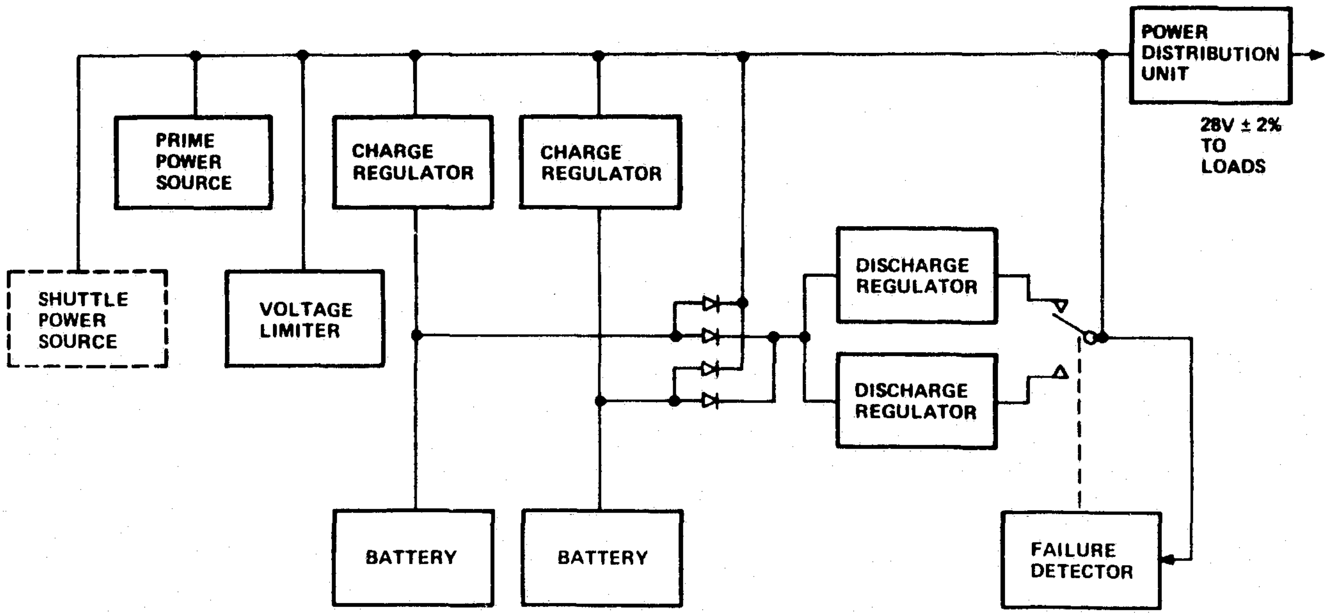


Figure 9. Shunt Regulated Power System Block Diagram

COST EVALUATION

The total cost of each of the candidate systems was developed as shown on Figure 10. The cost of electronics was found by summing the number of piece parts and multiplying by a factor of 100 dollars per part. This recurring cost per electronic part was developed from our Nimbus and Landsat spacecraft experience as well as power regulating circuits developed for the S193 Skylab experiment.

Solar array costs were determined by first calculating the different size arrays required to support a 400 watt load and charge the batteries. The effective array watts per unit area were determined from the Broadcast Satellite Experimental solar array which delivers 10 watts per square foot at approximately 3000 dollars per square foot. The effective array area of a spin stabilized spacecraft (spinner) is found by dividing the effective frontal area by the total circumferential area so that the effective watts per total array area is:

$$\frac{10 \text{ W/FT}^2 \text{ DL}}{\pi \text{DL}} \text{ or } 3.2 \text{ watts per square foot.}$$

Battery costs were based on the unit cost of a 50 ampere hour cell. 50 A-H cells were selected so that one battery could support the 400 watt load during eclipse periods.

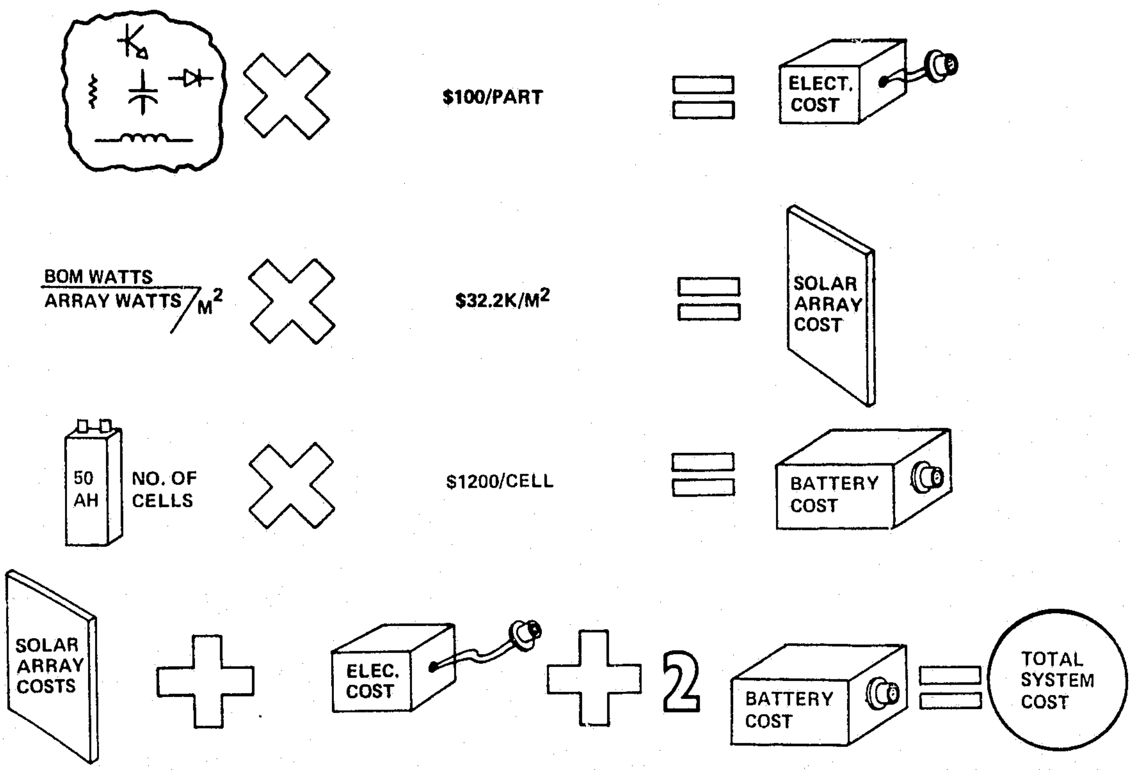


Figure 10. Cost Development

Electronics

Table 8 provides a piecepart count of all the various electronic equipment used in all three candidate systems. These counts are derived from hardware designs of the appropriate power levels and functions which have been built using discrete part packaging (no hybrid microcircuits except standard operational amplifier integrated circuits). The complexity of the power distribution unit was approximated to determine the relay and diode count.

Taking the unit costs from Table 8 and multiplying by the number of such units in each system produces the total electronics cost as shown in Table 9. The series system electronics are the lowest cost, but there is no great difference between any of the system costs.

TABLE 8. POWER SYSTEM PROCESSOR RECURRING COSTS

POWER SYSTEM PROCESSORS → / CIRCUIT COMPONENTS PER PROCESSOR ↓	LOAD REGULATOR	SERIES REGULATOR PWM	FAILURE DETECTOR & SWITCH-OVER	CHARGE REGULATOR	DISCHARGE REGULATOR	SHUNT REGULATOR	ARRAY LIMITER	POWER DISTRIBUTION UNIT
INTEGRATED CIRCUITS	3	–	4	1	2	3	–	–
TRANSISTORS FET'S, SCR'S, UNIJUNCT.	21	19	23	32	27	28	13	–
DIODES & ZENERS	25	41	25	18	33	40	11	160
RESISTORS	32	60	91	95	40	115	60	–
CAPACITORS & SMALL EMI FILTERS	14	42	19	10	25	14	4	–
INDUCTORS	2	6	–	–	3	–	–	–
REACTORS (SATURATING TYPE)	2	5	–	–	2	–	–	–
TRANSFORMERS	1	6	–	–	4	–	–	–
RELAYS	–	–	2	1	–	–	8	40
FUSES	–	–	–	–	2	–	–	50
GRAND TOTALS	100	179	162	157	138	200	96	250
COST/BLACK BOX	10K	18K	16K	16K	14K	20K	10K	25K

TABLE 9. ELECTRONICS COST FOR A 400-WATT SYSTEM

PROCESSOR COST		SYSTEMS ALTERNATIVES		
PROCESSOR TYPE	UNIT PRICE	REMOTE SYSTEM	SERIES SYSTEM	SHUNT SYSTEM
LOAD REGULATOR	10 K	70 K	–	–
SERIES REGULATOR	18 K	–	36 K	–
FAILURE DETECTOR	16 K	–	16 K	16 K
CHARGE REGULATOR	16 K	32 K	32 K	32 K
DISCHARGE REGULATOR	14 K	DIODE ONLY	DIODE ONLY	28 K
SHUNT REGULATOR	20 K	–	–	20 K
ARRAY LIMITER	10 K	10 K	10 K	–
POWER DISTRIBUTION UNIT	25 K	25 K	25 K	25 K
TOTAL COSTS		137 K	119 K	121 K

Solar Array

Solar array costs were determined for low earth application by first calculating the array power required to recharge the batteries, and then that required to support the loads during daylight operation. This data is provided as Appendix II-1.

After determining the end of mission power levels, a degradation factor of 12 percent per year was used to calculate the beginning of mission array size for a four year mission.

For geosynchronous applications, the array power required during the sunlit operation was divided by the cosine of 23.5 degrees to provide adequate power during solstice periods. A 6 percent per year degradation factor provided the beginning of mission power levels.

Table 10 provides the solar array costs to support a 400 watt spacecraft in both a geosynchronous and low earth orbit using an oriented array or a spinner.

Battery

Development of the battery charge/discharge levels is provided in Appendix II-2, but basically the remote and series regulated systems contains 23 celled batteries while the shunt system contains a 16 celled battery. The battery cost for each of the systems is shown on Table 10.

TABLE 10. SOLAR ARRAY AND BATTERY COSTS FOR A 400 WATT SYSTEM

MISSION	SOLAR ARRAY TYPE	ARRAY COST		
		REMOTE SYSTEM	SERIES SYSTEM	SHUNT SYSTEM
GEOSYNCHRONOUS	ORIENTED	193 K	205 K	181 K
	SPINNER	608 K	643 K	569 K
LOW EARTH	ORIENTED	501 K	531 K	497 K
	SPINNER	1575 K	1667 K	1561 K

BATTERY COSTS

REMOTE SYSTEM	SERIES SYSTEM	SHUNT SYSTEM
55K	55K	38K

System Summary

Adding the costs for a solar array, electronics, and two batteries results in the total cost for each system. As can be seen from Table 11, the shunt system has the lowest cost followed by the remote and then the series systems. (Hardware costs were developed based on current technology. Advances that result in lower costs could effect the ranking to some degree.)

The average system ranking on a cost comparison shows the shunt approach with a 6, the remote with a 5, and the series with a 4.8.

TABLE 11. POWER SYSTEM RECURRING COST FOR A 400 WATT SYSTEM

MISSION	SOLAR ARRAY TYPE	POWER SYSTEM COST REMOTE SYSTEM	RANK	SERIES SYSTEM	RANK	SHUNT SYSTEM	RANK
GEOSYNCHRONOUS ORBIT	ORIENTED	385 K	4.8	379 K	5	340 K	6
	SPINNER	800 K	5	817 K	4.8	728 K	5.9
LOW EARTH ORBIT	ORIENTED	693 K	5	705 K	4.8	656 K	5.5
	SPINNER	1767 K	5	1841 K	4.6	1720 K	5.3
AVERAGE RANK			5		4.8		5.7

PERFORMANCE EVALUATION

Table 12 summarizes the performance characteristics of the three candidate systems. This data is based on experience with switching regulators of the type used for the first two systems, and linear shunt regulators of the third system. The shunt system performance with respect to ripple, dynamic impedance, and transient response cannot be met by the switching regulators of the other two systems. With load regulation of all three systems being equal, the shunt system has the best overall performance rating.

POWER SOURCE FLEXIBILITY

To assess the compatibility of the three power systems with the power sources of Table 13, the source utilization must be factored into the evaluation. This is done by the numbers in parentheses next to the source name in Table 13.

For a solar array source, all three systems rank exceedingly high. However, as both the remote and series systems require an array voltage limiter as well as the regulator and the shunt system performs both functions with one device (shunt regulator), a 10 was given to the shunt while the others received a rank of 9.

Neither the remote nor the series system is compatible with a radioisotope thermoelectric generator (RTG) as the RTG must be operated at a constant power point to maintain internal thermal equilibrium. They therefore received a low rank.

TABLE 12. PERFORMANCE

POWER QUALITY FACTORS	REMOTE SYSTEM	RANK	SERIES SYSTEM	RANK	SHUNT SYSTEM	RANK
RIPPLE	200 MV P-P	5	200 MV P-P	5	50 MV P-P/	8
DYNAMIC IMPEDANCE	0.1 @ 10 KHZ & 0.2 @ 50 KHZ	3	0.1 @ 10 KHZ & 0.2 @ 50 KHZ	3	0.01 @ 10 KHZ & 0.1 @ 50 KHZ	8
TRANSIENT RESPONSE	3-4 MILLI-SECONDS	1	3-4 MILLI-SECONDS	1	30 MICRO-SECONDS	10
LOAD REGULATION	± 2%	5	± 2%	5	± 2%	5
NORMALIZED RANK		5		5		9.2

TABLE 13. POWER SOURCE FLEXIBILITY

		REMOTE		SERIES		SHUNT	
		RANK	WEIGHTED RANK	RANK	WEIGHTED RANK	RANK	WEIGHTED RANK
SOLAR ARRAY	(0.9)	9	8.1	9	8.1	10	9.0
RTG	(0.05)	1	0.05	1	0.05	10	0.5
FUEL CELL	(0.05)	10	0.5	10	0.5	9	0.45
PRIME SOURCE TOTAL			8.65		8.65		9.95
SHUTTLE SOURCE		10		10		9	
TOTAL SOURCE FLEXIBILITY		9.3		9.3		9.5	
NORMALIZED RANK		5		5		5.2	

All three systems are compatible with a fuel cell source, but as a switch must be added to the shunt system to allow the battery charger to drop the fuel cell voltage and the battery discharger to boost it up to the regulated bus, a lower rank resulted for this system.

After determining the weighted rank, the ranking was normalized to provide a value of 5 for the remote and series with the shunt slightly better with a 5.2

CONVERSION EFFICIENCY

Conversion efficiency for the three system candidates was determined by determining the watt hours of solar array required to support a constant load of 400 watts. These efficiencies were determined for both low earth and geosynchronous orbits with the assumptions for the power conditioning electronics as shown on Table 14. In both orbits, the shunt system had the best conversion efficiency, followed by the remote and then the series systems.

RELIABILITY

To determine the reliability of the system candidates, system models were developed showing the probability of success.

The reliability of each of the system components was calculated and these results were factored into the system models from which the numerical results were compared and the system ranking was determined.

TABLE 14. CONVERSION EFFICIENCY

ASSUMPTIONS

- CHARGE REGULATORS & REMOTE REGULATORS ARE 90% EFFICIENT.
- SERIES REGULATORS & DISCHARGE REGULATORS ARE 85% EFFICIENT.
- ISOLATION DIODES ARE 97% EFFICIENT.
- SPACECRAFT HARNESSING IS 96% EFFICIENT.
- LOW EARTH ORBIT 67% DAY & 33% NIGHT
- SYNCHRONOUS ORBIT ECLIPSE = 72 MIN/DAY.
- DAY TIME LOAD = NIGHT TIME LOAD = 400 WATTS.

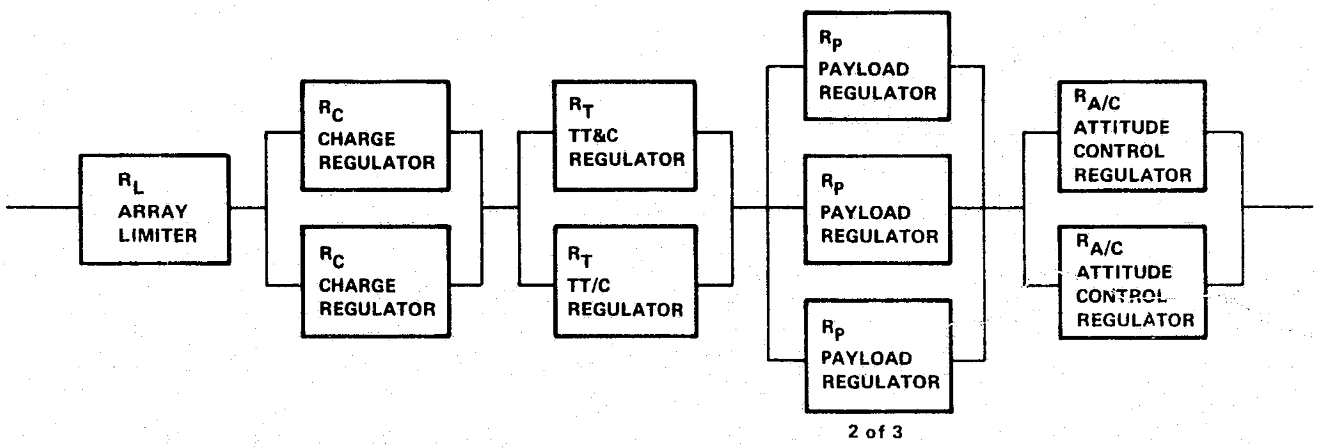
FORM USED

$$\text{EFF.} = \frac{\text{LOAD WATT HRS CONSUMED}}{\text{ARRAY WATT HRS PROVIDED}} \text{ @ EOM}$$

RESULTS

	REMOTE SYSTEM	RANK	SERIES SYSTEM	RANK	SHUNT SYSTEM	RANK
LOW EARTH	69%	5	66%	4.6	70%	5.1
GEOSYNCHRONOUS	84%	5	79%	4.4	90%	5.7
AVERAGE RANK		5		4.5		5.4

Figure 11 shows the reliability model for the remote regulation system. The array limiter must work, one of the two charge regulators, one of the two TT&C regulators, two of the three payload regulators, and one of the two attitude control regulators must operate for success. This is expressed in the equation at the bottom of Figure 11.

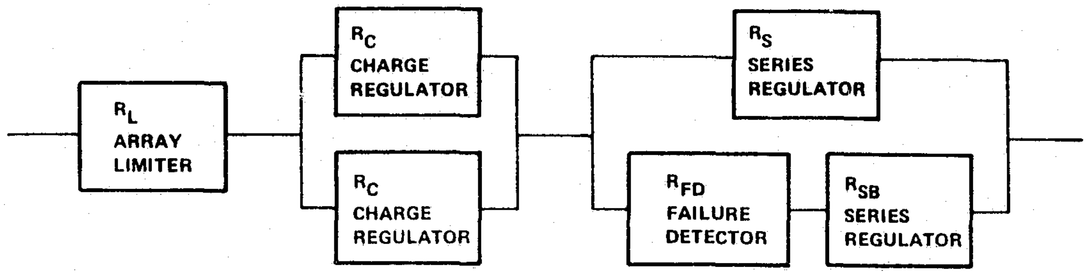


$$R_{SYSTEM} = \left(R\ \text{ARRAY LIMITER}\right)\left(R\ \text{CHARGE REGULATOR}\right)\left(R\ \text{TT\&C REGULATOR}\right)\left(R\ \text{PAYLOAD REGULATOR}\right)\left(R\ \text{ATTITUDE CONTROL REGULATOR}\right)$$

$$R_{SYSTEM} = (R_L)\ (R_C^2 + 2R_CQ_C)\ (R_T^2 + 2R_TQ_T)\ (R_P^3 + 3R_P^2Q_P)\ (R_{A/C}^2 + 2R_{A/C}Q_{A/C})$$

Figure 11. Remote Regulation Reliability Diagram

The series system model of Figure 12 has a different arrangement because of the failure detector used to switch the series regulator. Success can be realized thru the series regulator function if the first series regulator operates, or if it fails, the failure detector operates properly and the second series regulator has survived the standby dormancy period.



$$R_{SYSTEM} = \left(R_{\substack{\text{ARRAY}\\\text{LIMITER}}}\right)\left(R_{\substack{\text{CHARGE}\\\text{REGULATOR}}}\right)\left(R_{\substack{\text{SERIES}\\\text{REGULATOR}}}\right)$$

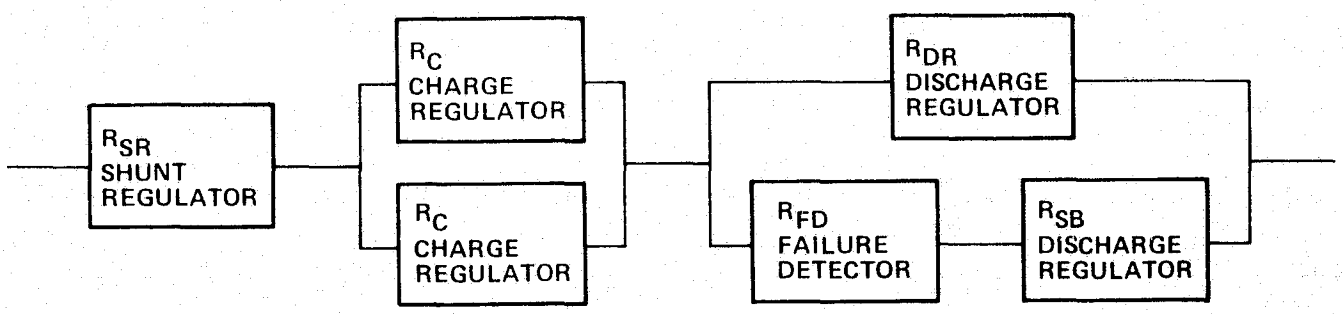
$$R_{SYSTEM} = (R_L)\,(R_C^2 + 2R_CQ_C)\,(R_S + Q_S R_{FD} R_{SB})$$

Figure 12. Series Regulation Reliability Diagram

The shunt system model of Figure 13 is similar to the series, the difference is that the array limiter has been replaced by the shunt regulator.

The failure rates used for the electronic piece parts, the equipment reliability calculations, and the system calculations are provided for reference in Appendix II-3.

A summary of the calculations in Table 15 show the shunt system to be the most reliable with the series system next and the remote system last.



$$R_{SYSTEM} = \left(R_{\substack{\text{SHUNT}\\\text{REGULATOR}}}\right)\left(R_{\substack{\text{CHARGE}\\\text{REGULATOR}}}\right)\left(R_{\substack{\text{DISCHARGE}\\\text{REGULATOR}}}\right)$$

$$R_{SYSTEM} = \left(R_{SR}\right)\left(R_c^2 + 2R_cQ_c\right)\left(R_{DR} + Q_{DR} R_{FD} R_{SB}\right)$$

Figure 13. Shunt Regulation Reliability Diagram

TABLE 15. RELIABILITY SUMMARY

COMPONENT RELIABILITY SUMMARY

REMOTE REG'R	.9848	DISCHARGE REG'R	.9809
SERIES REG'R	.9779	(IN STANDBY)	.9981
(IN STANDBY)	.9978	ARRAY LIMITER	.9914
CHARGE REG'R	.9787	SHUNT REG'R	.9921
		FAILURE DETECTOR	.9671

SYSTEM RELIABILITY SUMMARY

	REMOTE SYSTEM	RANK	SERIES SYSTEM	RANK	SHUNT SYSTEM	RANK
RELIABILITY	.9897		.9902		.9910	
PROBABILITY OF FAILURE	.0103	4.5	.0098	5	.0090	6
SYSTEM RANK		4.5		5		6

THERMAL COTROL AND DISSIPATION

To determine the thermal control and dissipation penalty of each system, the obvious losses in the power processors must be summed, but also the excess solar array power that must be handled by the shunt regulator or the solar array voltage limiter must be included as a thermal consideration.

The three systems with their power losses for battery and solar array operation are shown in Figure 14. Maintaining the same load of 400 watts, the solar array watts required to charge the battery were determined, and those watts for supporting the loads were added to the battery charging watts which provided the total solar array at end of mission (4 years). Using a solar array degradation of 48%, the beginning of mission power levels were determined at the 28 volt level. As the array voltage limiter operates when the voltage reaches 35 VDC, which at the BOM is closer to the array peak power point, a 3% increase in the power provided was factored into the remote and series systems. This data, along with the excess solar array power that must be handled is shown in Table 16.

Table 17 summarizes the power dissipations for all equipment in each system. The orbit average power was determined by adding the day and night watt hours and dividing by the number of hours per orbit. The shunt system has the least thermal burden and rates the highest, followed by the remote and then the series systems.

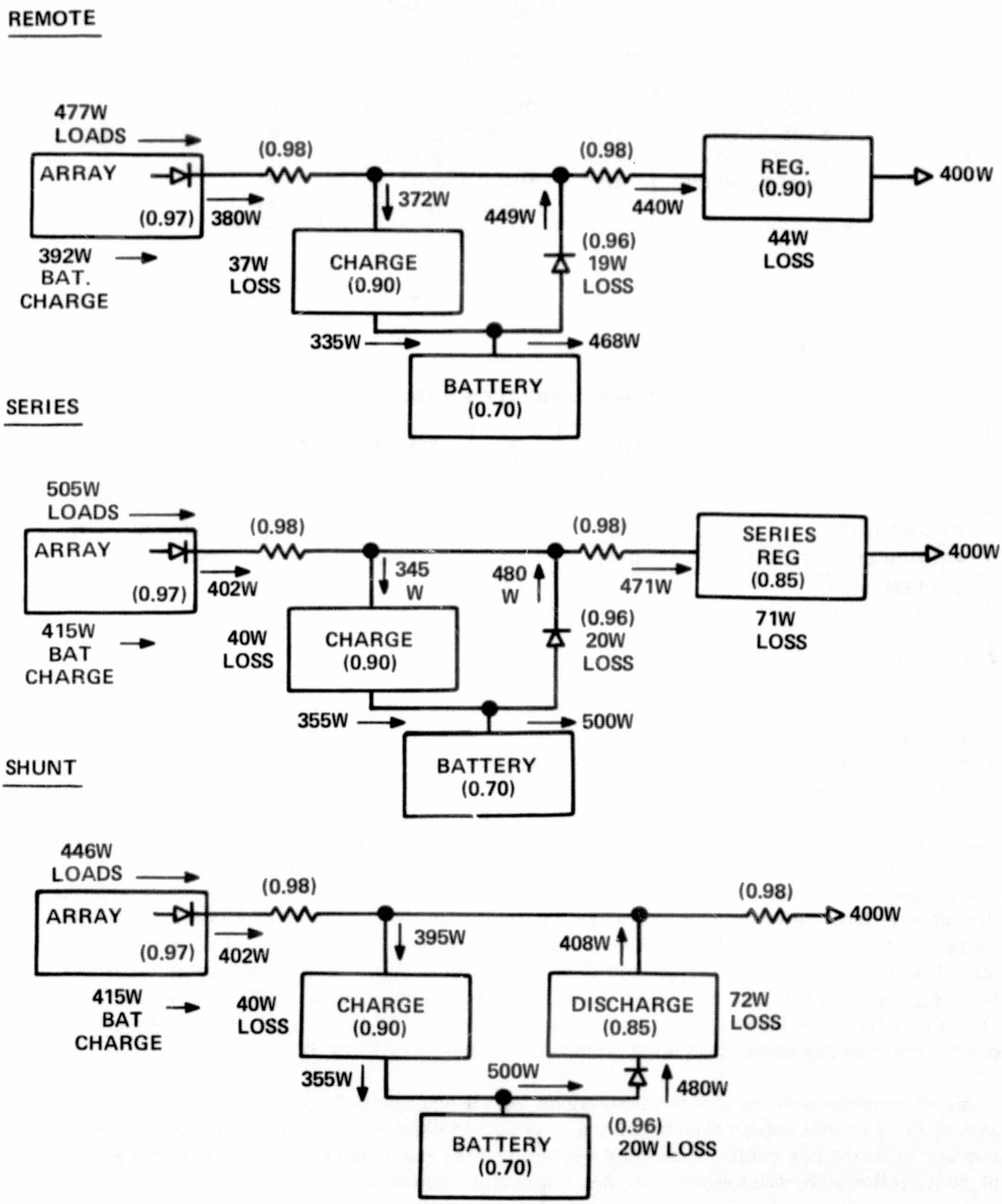


Figure 14. Thermal Dissipation Diagrams

TABLE 16. SOLAR ARRAY POWER AND EXCESS POWER LEVELS

	REMOTE	SERIES	SHUNT
EOM POWER @ 28V	869 W	920 W	861 W
BOM POWER @ 28V	1,671 W	1,769 W	1,656 W
BOM POWER @ 35V	1,721 W	1,822 W	---
EXCESS POWER FOR ARRAY LIMITER	852 W	902 W	---
EXCESS POWER FOR SHUNT REGULATOR	---	---	795 W

TABLE 17. BEGINNING OF MISSION POWER DISSIPATION (LOW EARTH ORBIT)

SYSTEM ELEMENTS	REMOTE		SERIES		SHUNT	
	NIGHT	DAY	NIGHT	DAY	NIGHT	DAY
SERIES REG.	–	–	71W	71W	–	–
DISCHARGE DIODE	19W	–	20W	–	20W	–
CHARGE REGULATOR	–	37W	–	40W	–	40W
REMOTE REGULATOR	44W	44W	–	–	–	-
ARRAY LIMITER	–	852W	–	902W	–	–
SHUNT REGULATOR	–	–	–	–	–	795W
DISCHARGE REGULATOR	--	–	–	–	72W	–
TOTAL POWER	63W	933W	91W	1013W	92W	835W
ORBIT AVERAGE	638W		699W		582W	
RANK	5		4		6	

SAFETY

The three systems were reviewed with respect to the following criteria where personal safety might be affected:

- High Voltage
- High Temperature
- High Pressure
- Toxicity
- Flammability
- Pyrotechnics
- Radioactivity

None of the systems have any of the above potential safety hazards, so they were all rated as a 5.

WEIGHT

To determine the weight of each system, the components such as solar array, battery, and electronics were sized for a 400 watt spacecraft load.

The battery weight was determined by using a 2.05 kg cell weight, a battery packing factor of 1.4 and the fact that there are two batteries per system. This provides a battery weight for each system as follows:

	Remote System	Series System	Shunt System
Weight (kg)	132	132	92

The solar array weight for each system was determined by sizing to the beginning of mission power requirements. Table 18 provides the array sizing assumptions and also the cell, substrate, and deployment hardware weights.

The unit weight of the various system electronic elements was developed from existing hardware of the particular type with an extrapolation to the proper power level. These unit weights, the totals for the three systems, the solar array, and batteries are all summed in Table 19 which shows the shunt system to be the lightest, hence the best, and the other two systems about equal in weight.

TABLE 18. SOLAR ARRAY WEIGHT (ORIENTED ARRAY)

ASSUMPTIONS

- **DEPLOYMENT HARDWARE & YOKE** = 21 Kg
- **CELLS & SUBSTRATE** = 3.252 Kg/M^2
- **ORIENTED ARRAY POWER DENSITY** = 107.6 W/M^2
- **CELL & SUBSTRATE WT/WATT** = 0.0302 Kg/Watt

SOLAR ARRAY WEIGHT/AREA (LOW EARTH-ORIENTED)

	REMOTE SYSTEM	SERIES SYSTEM	SHUNT SYSTEM
ARRAY POWER (BOM) (WATTS)	1671	1769	1656
WEIGHT OF CELLS AND SUBSTRATE (Kg)	50.5	53.5	50
DEPLOYMENT HARDWARE (Kg)	21	21	21
TOTAL WT. (Kg)	71.5	74.5	71
TOTAL AREA (M^2)	15.5	16.4	15.4

TABLE 19. WEIGHT

SYSTEM ELEMENT	UNIT WT Kg (LB)	REMOTE REGULATION		SERIES REGULATION		SHUNT REGULATION	
		QTY	WT	QTY	WT	QTY	WT
REMOTE REGULATOR	1.36 (3.0)	7	9.52 (21)	0	–	0	–
SERIES REGULATOR	4.08 (9.0)	0	–	2	8.16 (18)	0	–
SHUNT REGULATOR	4.76 (10.5)	0	–	0	–	1	4.76 (10.5)
CHARGE REGULATOR	1.36 (3.0)	2	2.72 (6)	2	2.72 (6)	2	2.72 (6)
FAILURE DETECTOR	1.81 (4.0)	0	–	1	1.81 (4)	1	1.81 (4)
DISCHARGE REGULATOR	3.62 (8.0)	0	–	0	–	2	7.24 (16)
ARRAY LIMITER	3.85 (8.5)	1	3.85 (8.5)	1	3.85 (8.5)	0	–
POWER DIST UNIT	4.54 (10.0)	1	4.54 (10)	1	4.54 (10)	1	4.54 (10)
TOTAL ELECTRONICS WEIGHT			20.63 (45.5)		21.08 (46.5)		21.07 (46.5)
SOLAR ARRAY			71.5 (158)		74.5 (164)		71.0 (157)
BATTERIES			132.0 (291)		132.0 (291)		92.0 (203)
TOTAL SYSTEM			224 (494)		228 (503)		184 (406)
RANK			5		5		6.8

VOLUME

In determining the system volumes, the resistor load bank associated with the array limiter and the shunt regulator was not included nor was the solar array itself as these items are not internal to the spacecraft bus and as such do not present a volume penalty to the bus design.

The battery volume was calculated by using a cell volume of 705 cm^3 and a packing volume factor of 1.1. The electronics were determined in the same manner as the weight factors mentioned above. Table 20 shows that the shunt system is the smallest, hence the best and the other two systems are about the same size.

TABLE 20. VOLUME

SYSTEM ELEMENT	UNIT VOLUME cm^3 (IN^3)	REMOTE REGULATION		SERIES REGULATION		SHUNT REGULATION	
		QTY	VOL	QTY	VOL	QTY	VOL
REMOTE REGULATOR	1639 (100)	7	11473 (700)	0	–	0	–
SERIES REGULATOR	4507 (275)	0	–	2	9014 (550)	0	–
SHUNT REGULATOR	2295 (140)	0	–	0	–	1	2295 (140)
CHARGE REGULATOR	1229 (75)	2	2458 (150)	2	2458 (150)	2	2458 (150)
FAILURE DETECTOR	1803 (110)	0	–	1	1803 (110)	1	1803 (110)
DISCHARGE REGULATOR	4360 (266)	0	–	0	–	2	8720 (532)
ARRAY LIMITER	1950 (119)	1	1950 (119)	1	1950 (119)	0	–
POWER DIST UNIT	6884 (420)	1	6884 (420)	1	6884 (420)	1	6884 (420)
TOTAL ELECTRONICS VOLUME			22765 (1389)		22109 (1349)		22160 (1352)
BATTERIES			35.7 K		35.7 K		24.8 K
TOTAL			58.5 K		57.8 K		47.0 K
RANK			4.9		5		6.9

SYSTEM COMPLEXITY

Two approaches were used to evaluate system complexity. For the first approach, block diagrams to the lowest level functions (oscillators, comparators, filters, etc.) necessary to perform a given sub-function (charge regulator, boost regulator, etc.) were developed. Then the number of functions within a system were totaled and compared. These block diagrams are provided in Appendix II-4.

The second evaluation approach was to perform a piece part count of each of the system elements using existing spacecraft schematics. The parts were totaled and systems compared. The results of both techniques shown in Table 21 provide an almost identical ranking of

TABLE 21. SYSTEM COMPLEXITY

COMPLEXITY SUMMARY & RANKING

SYSTEMS ELEMENTS (E)	ELEMENT		REMOTE SYSTEM			SERIES SYSTEM			SHUNT SYSTEM		
	PARTS (P)	FUNCTIONS (F)	(E) QTY	(E · P)	(E · F)	(E) QTY	(E · P)	(E · F)	(E) QTY	(E · P)	(E · F)
REMOTE REG'R	100	8	7	700	56	–	–	–	–	–	–
SERIES REG'R	179	13	–	–	–	2	358	26	–	–	·
SHUNT REG'R	200	15	–	–	–	–	–	–	1	200	15
CHARGE REG'R	157	11	2	314	22	2	314	22	2	314	22
FAILURE DET'R	162	15	–	–	–	1	162	15	1	162	15
DISCHARGE REG'R	138	11	–	–	–	–	–	–	2	276	22
ARRAY LIMITER	96	9	1	96	9	1	96	9	–	–	–
POWER DIST. UNIT	250	40	1	250	40	1	250	40	1	250	40
PART COMPLEXITY TOTAL				1360			1180			1202	
PART COMPLEXITY RANKING				3.7			6.2			5	
FUNCTION COMPLEXITY TOTAL					127			112			114
FUNCTION COMPLEXITY RANKING					3.9			5.2		5	
PART & FUNCTION COMPLEXITY AVE. RANK					3.8			6.2		5	

systems. The series system is the system with the fewest parts and functions and is the best followed by the shunt and then by the remote systems.

PARALLELING

To determine the effects of paralleling, it was assumed that the basic 400 watt systems were doubled to 800 watts. This required an expansion of the 3 system diagrams as shown on Figures 15 thru 17. Next, the functions of regulation, fault protection, power sources, and power distribution were examined for each system to determine paralleling effects.

Discussion of each function is provided on Table 22 which shows that the remote system has the advantage over the others for paralleling effects and ranks higher.

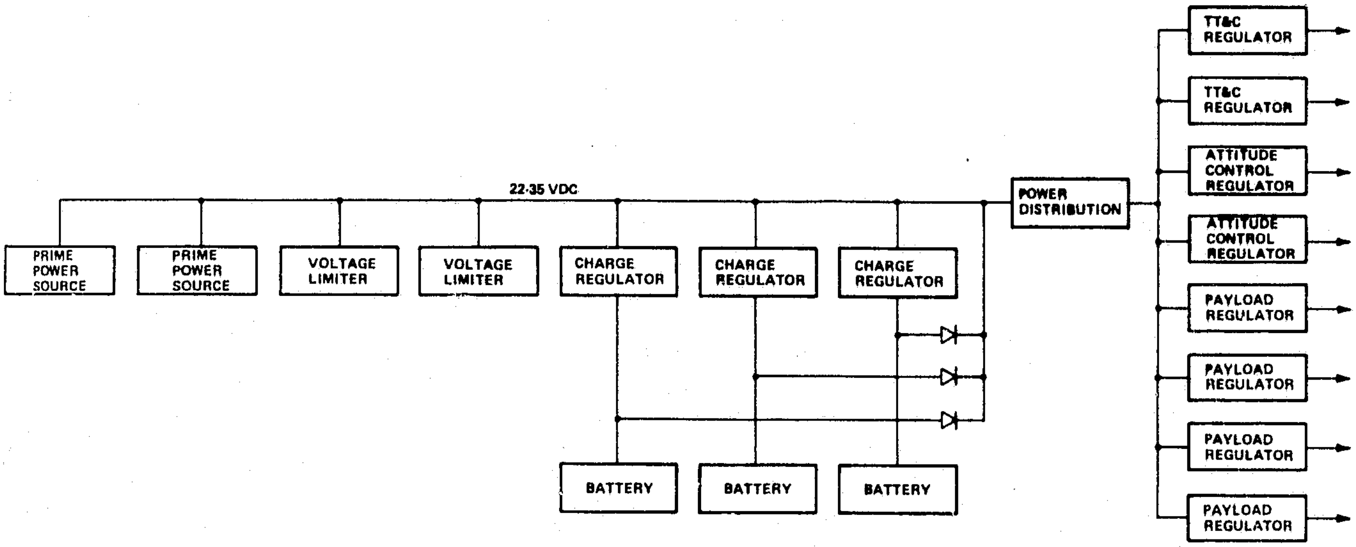


Figure 15. Remote Regulated Power Subsystem Block Diagram

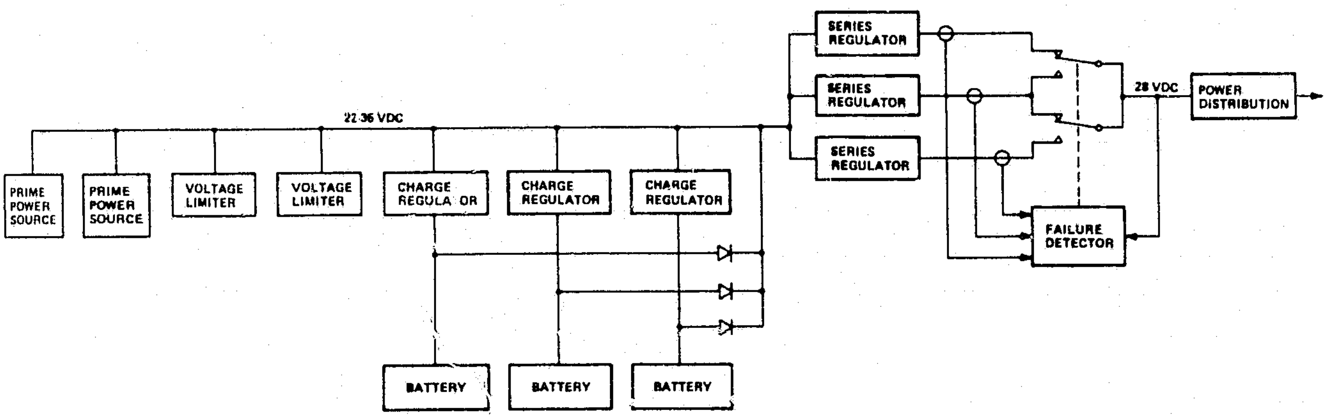


Figure 16. Series Regulated Power Subsystem Block Diagram

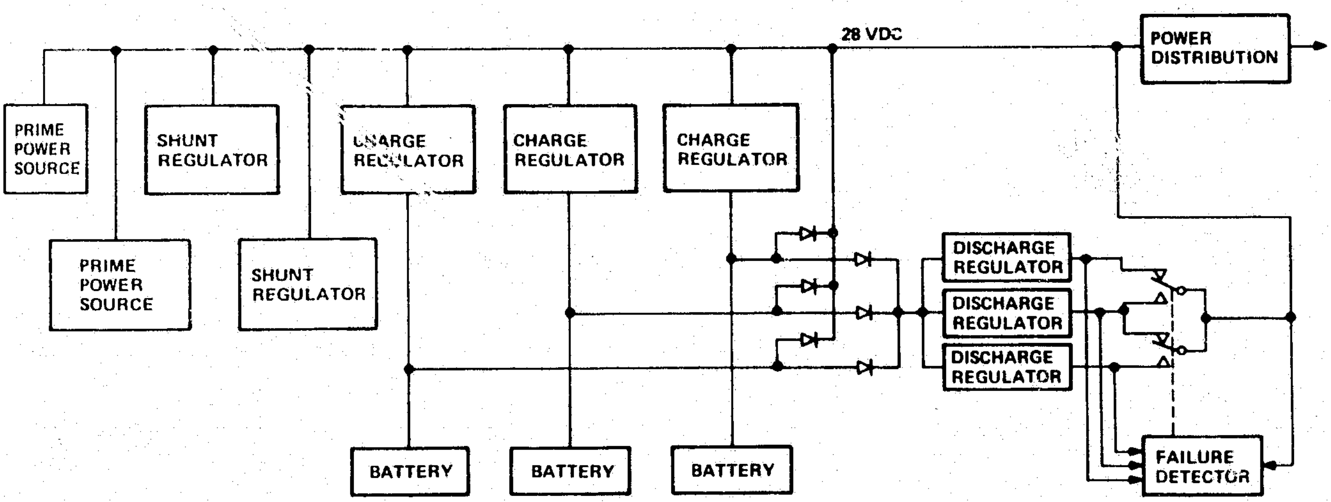


Figure 17. Shunt Regulated Power Subsystem Block Diagram

TABLE 22. PARALLELING EFFECTS

SYSTEM ELEMENTS	REMOTE SYSTEM	RANK	SERIES SYSTEM	RANK	SHUNT SYSTEM	RANK
REGULATORS						
• POWER QUALITY (RIPPLE) (TRANSIENT) (IMPED.)	ISOLATED REGULATORS CHARACTERISTICS NOT COMPLICATED BY PARALLELING.	2	SKYLAB ATM INDICATED NO MAJOR PROBLEM.	2	CHARACTERISTICS EASILY MAINTAINED VIA ADDITIONAL SHUNT ELEMENTS – FRONT END OF REGULATOR CAN BE DESIGNED COMMON TO ALL POWER RANGE	2
• LOAD POWER SHARING	REMOTE REGULATOR IS POWER LIMITED, WILL FORCE POWER SHARING – NO ADDITIONAL CIRCUITS REQ'D.	2	SERIES REGULATOR IS POWER LIMITED, WILL FORCE POWER SHARING. NO ADDITIONAL CIRCUITS REQ'D.	2	DISCHARGE REGULATOR IS POWER LIMITED. WILL FORCE POWER SHARING. NO ADDITIONAL CIRCUITS REQ'D.	2
• NOISE IMMUNITY	INDIVIDUAL REGULATORS PROVIDED INHERENT IMMUNITY	3	EACH SUBSYSTEM MUST PROVIDE ITS OWN LINE FILTER AND DECOUPLING NETWORKS.	2	EACH SUBSYSTEM MUST PROVIDE ITS OWN LINE FILTER AND DECOUPLING NETWORKS	2
REGULATOR AVERAGE RANK		2		2		2
FAULT DETECTORS	ACTIVE REDUNDANCY NO FAULT DET'R REQ'D.	3	SOPHISTICATED FAILURE DETECTOR REQUIRED. (MUST IDENTIFY THE FAILED UNIT FROM OTHERS IN OPERATION.	1	SOPHISTICATED FAILURE DETECTOR REQUIRED. (MUST IDENTIFY THE FAILED UNIT FROM OTHERS IN OPERATION).	1
POWER SOURCES						
• SOLAR ARRAY	ADDITIONAL POWER HANDLING REQUIRED IN VOLTAGE LIMITER	2	ADDITIONAL POWER HANDLING REQUIRED IN VOLTAGE LIMITER	2	ADDITIONAL POWER HANDLING REQUIRED IN SHUNT REGULATOR.	2
• BATTERIES	EASILY CONNECTED TO COMMON BATTERY BUS. ADD CHARGE REGULATOR FOR EACH BATTERY.	3	EASILY CONNECTED TO COMMON BATTERY BUS. ADD CHARGE REGULATOR FOR EACH BATTERY	3	EASILY CONNECTED TO COMMON BATTERY BUS. ADD CHARGE REGULATOR FOR EACH BATTERY.	3
POWER SOURCE AVERAGE RANK		2.5		2.5		2.5
POWER DISTRIBUTION	NO EFFECT	2	NO EFFECT	2	NO EFFECT	2
TOTAL SYSTEM AVERAGE RANK		2.4		1.9		1.9

*NOTE: RANKING OF ELEMENTS IS NOT QUANATIVE; THEREFORE RELATIVE POINTS (1 TO 3) WERE USED.

MAINTENANCE

Power system maintenance pertains to the repair or replacement of failed equipment by shuttle crew after initial spacecraft launch. System maintenance is therefore related to system failures. The failure probability of the system elements was determined and using this data, the system failure probability was found as shown on Table 23. As the probability of failure is less for the shunt system, it was given the highest rank followed by the series system and lastly by the remote system.

INTERCONNECTIONS

The approach to determine the quantity of interconnections within a system was to establish a functional level wire count based on elemental block diagrams in conjunction with the 3 basic power system block diagrams. Included in this count were command, telemetry, and

TABLE 23. MAINTENANCE

APPROACH

- **SYSTEM MAINTENANCE IS RELATED TO SYSTEM FAILURES.**
- **DETERMINE FAILURE PROBABILITY OF SYSTEM ELEMENTS.**
- **COMPUTE SYSTEM FAILURE PROBABILITY AND COMPARE.**

SUMMARY OF SYSTEM ELEMENT FAILURE PROBABILITY

REMOTE	REG'R	.0152	ARRAY	LIMITER	.0086
SERIES	REG'R	.0221	SHUNT	REG'R	.0079
CHARGE	REG'R	.0213	FAILURE	DETECTOR	.0329
DISCHARGE	REG'R	.0191			

SUMMARY OF SYSTEM FAILURE PROBABILITY

REMOTE SYSTEM	RANK	SERIES SYSTEM	RANK	SHUNT SYSTEM	RANK
.1576	3	.1283	5	.1216	5.5

status/diagnostic monitors. Not included was multiple wiring required for redundancy or current handling capability. The detailed count of the system interconnections is provided in Appendix II-5 and summarized below:

Remote System	Rank	Series System	Rank	Shunt System	Rank
144	1.4	106	5	94	6

As the shunt system has the fewest interconnections, it was given the highest rank.

EVALUATION AND SYSTEM SELECTION

Table 24 summarizes the Task II effort of selecting the system best suited for modularity. Each criterion with its weighting factor is listed. The ranking of each system with respect to each criterion that has been developed is included. The product of rank times weighting factor has been calculated here and then summed for each system. The system with the largest number which is the shunt regulated system with a 606 was the one recommended for the conceptional design effort of Task III.

TABLE 24. MODULARIZED SYSTEM CONCEPTS EVALUATION MATRIX

ITEM NO.	SYSTEM CRITERION	WEIGHTING FACTOR (WF)	REMOTE REGULATION		CENTRAL SERIES REGULATION		CENTRAL SHUNT REGULATION	
			RANK	RxWF	RANK	RxWF	RANK	RxWF
1	COST	18	5	90	4.8	86	5.7	103
2	PERFORMANCE	12	5	60	5	60	9.2	110
3	POWER SOURCE FLEXIBILITY	7	5	35	5	35	5.2	36
4	CONVERSION EFFICIENCY	12	5	60	4.5	54	5.4	65
5	RELIABILITY	11	4.5	50	5	55	6	66
6	THERMAL CONTROL & DISSIPATION	6	5	30	4	24	6	36
7	SAFETY	5	5	25	5	25	5	25
8	WEIGHT	4	5	20	5	20	6.8	27
9	VOLUME	4	4.9	20	5	20	6.9	28
10	SYSTEM COMPLEXITY	9	3.8	34	5.2	47	5	45
11	PARALLEL OPERATION	6	5.5	33	5	30	5	30
12	MAINTENANCE	3	3	9	5	15	5.5	17
13	INTERCONNECTIONS	3	1.4	4	5	15	6	18
	TOTAL VALUE			470		486		606

BEST APPROACH

TASK III

CONCEPTUAL DESIGN OF A MODULAR SYSTEM

A summary of the Task I and Task II results which were used as the basis for the development of a conceptual design are presented by Figures 18 and 19.

Figure 18 shows that of the 228 automated (free flying) spacecraft identified by the NASA mission model, most all will use solar arrays as a primary power source with rechargable batteries. Also, a fact that was used to size the power handling capability of the power system selected for development in Task III was that 55% of the spacecraft required less than 400 watts of electrical power.

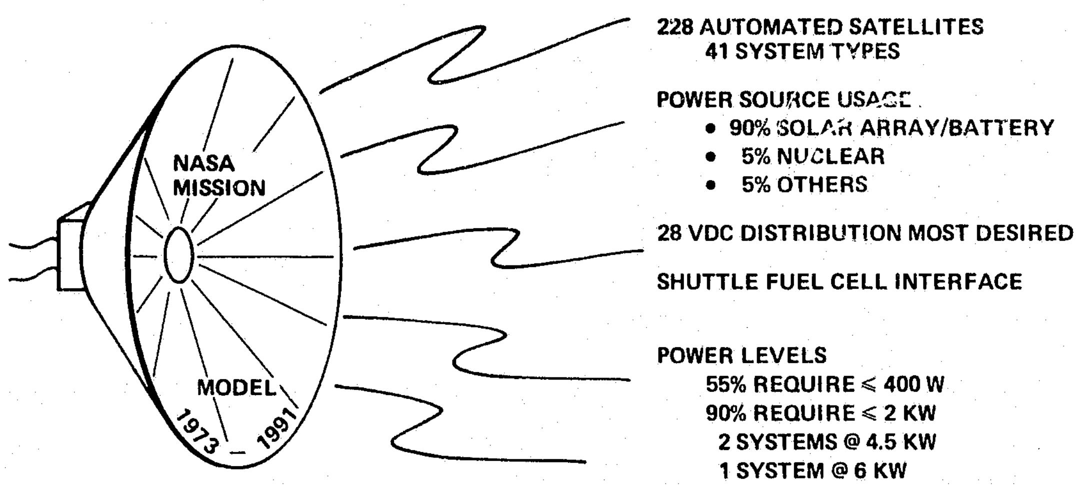


Figure 18. Results of Task 1 Review of Information

Figure 19 reviews the procedure of Task II which resulted in the selection of a shunt regulated system for the conceptual design phase. This was accomplished by developing and weighting thirteen evaluation criteria and then comparing three systems with respect to each of the criterion. A block diagram of the system selected by this evaluation ranking is shown for reference in Figure 20.

BASIC SYSTEM CONFIGURATION

An elaboration of the Task II block diagram which shows the minimal system is provided in Figure 21. A central voltage controller measures the regulated bus voltage and provides a signal to operate the parallel boost converters to maintain regulation during eclipse by battery discharge. The central controller enables the battery chargers when solar array power exceeds the needs of the spacecraft loads. Excess array power above this is dissipated in the partial shunt regulator which is also controlled by the central controller. Currents, voltages, and temperatures important to power system operation are shown on the diagram. A remote decoder/multiplexer is provided to interface the power system commands and

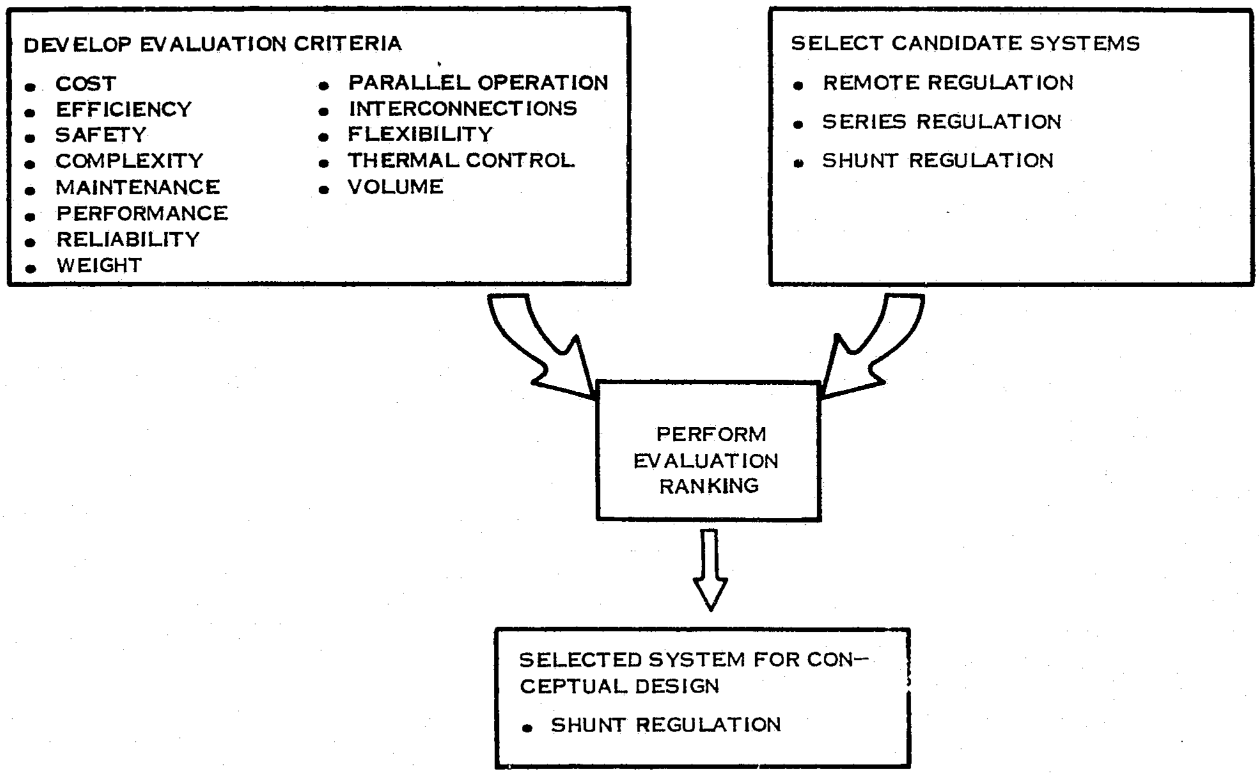


Figure 19. Results of Task II Selection of System Concept

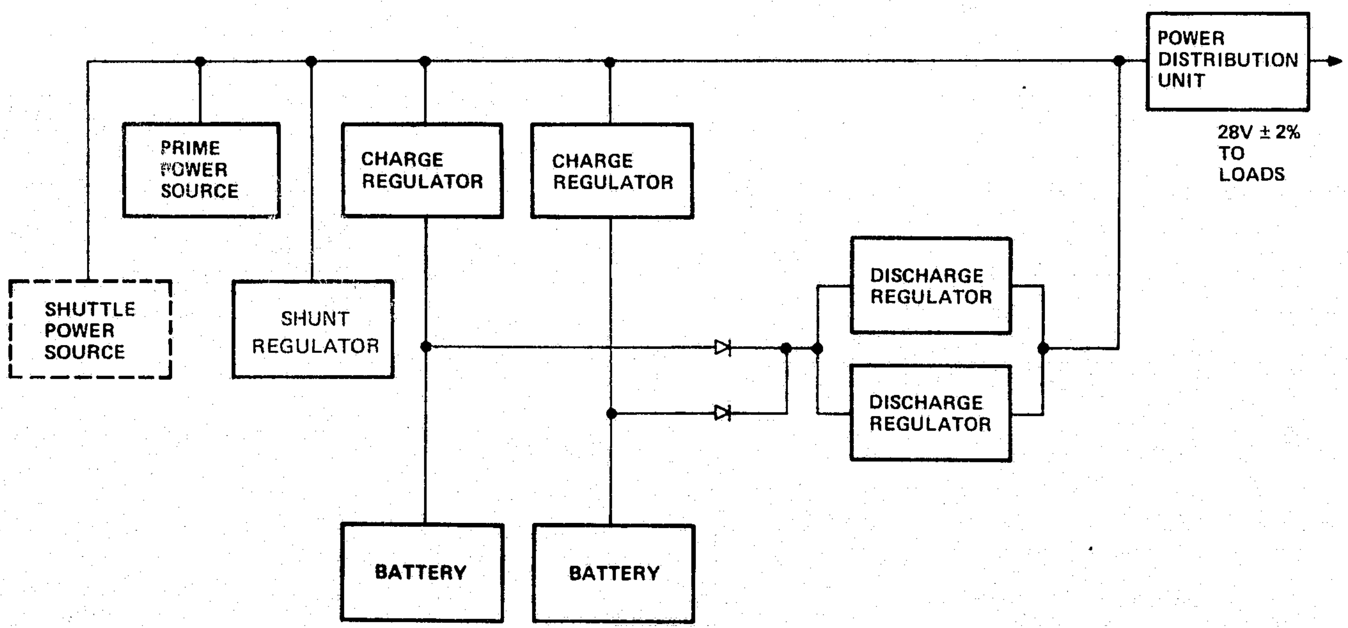


Figure 20. Shunt Regulated Power System Block Diagram

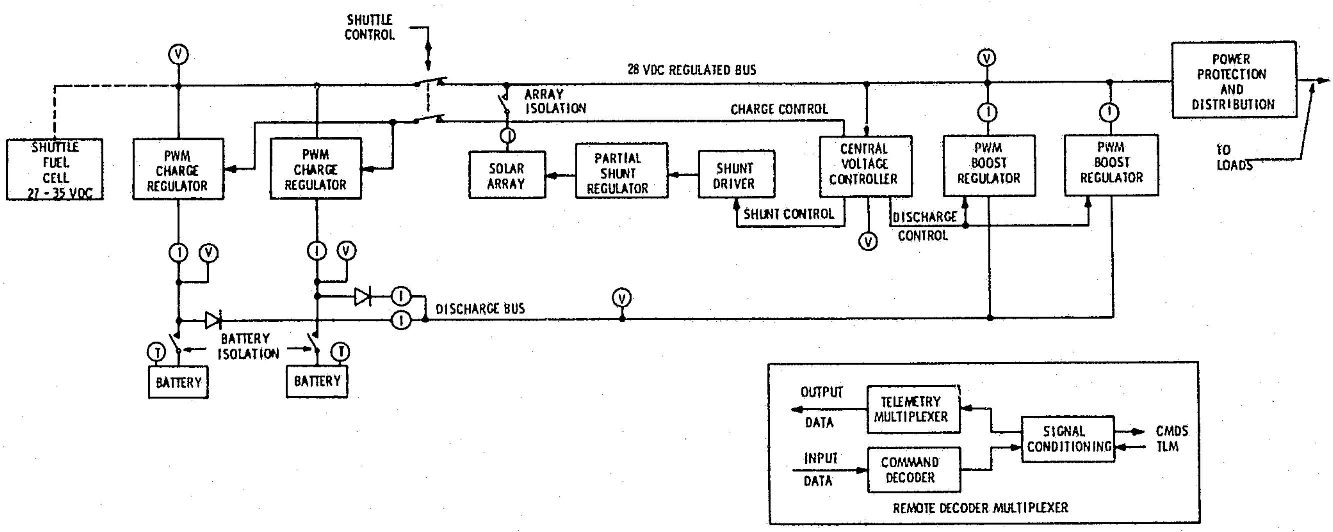


Figure 21. Power System Block Diagram

telemetry and minimize intersystem wiring by use of serial data and command buses. The batteries are joined thru isolation diodes to form a common discharge bus that will force battery load sharing.

A shuttle control switch between the battery chargers and the power system allows the system to interface with the shuttle fuel cell. This can be explained by the use of Figure 22. The unregulated voltage of the shuttle fuel cell varies above and below the 28 volts of the power system. To interface this power source to the regulated bus would require that the fuel cell voltage be bucked down or boosted up to the 28 volt level. As the power system contains both buck and boost regulators for battery charge and discharge a technique was developed to use these devices for fuel cell interfacing thus avoiding the complexity of a buck-boost regulator in the shuttle.

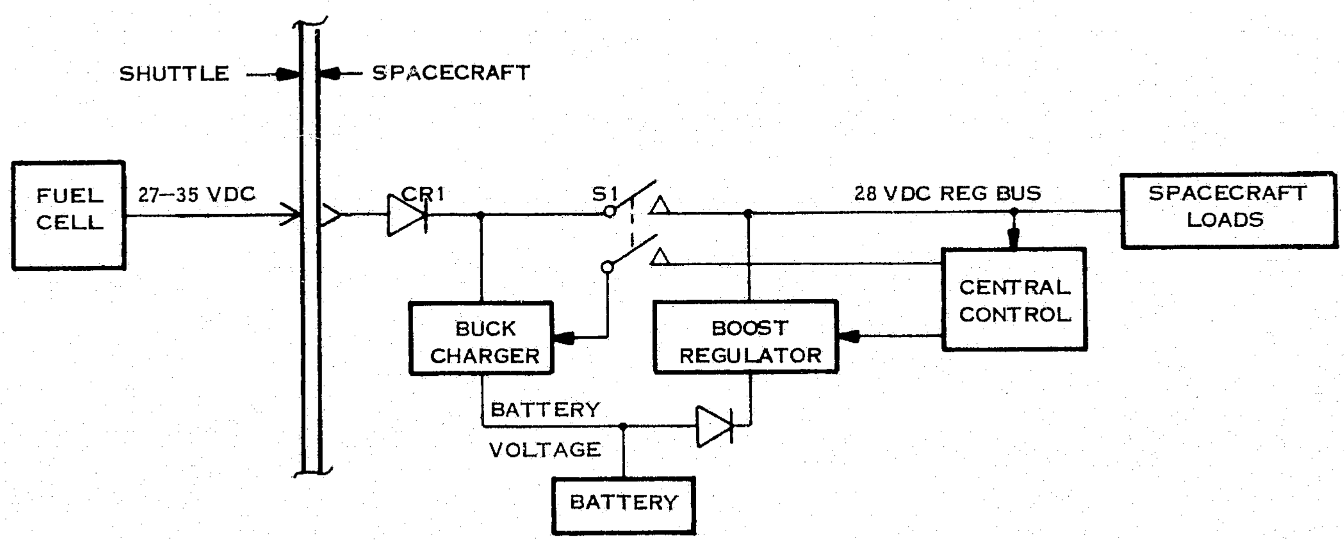


Figure 22. Shuttle Power Interface Using Battery Charger/Discharge Regulator

The buck charger is disconnected from the regulated bus via switch S1. The fuel cell voltage is dropped down to the battery terminal voltage and then boosted up to the regulated bus. In normal operation, the central controller does not permit simultaneous operation of the buck charger and the boost regulator. To overcome this during shuttle operation, a second contact of switch S1 is used to disconnect the central control signal from the charger and the charger operation is determined by only its output voltage and current levels. The diode CR1 of Figure 22 protects the regulated bus from shorts at the spacecraft/shuttle interface after separation of the electrical connector.

DEVELOPMENT OF THE SYSTEM ELEMENTS

The rationale for system element selection and requirements is provided here. Preliminary specifications for the battery charger, discharge regulator, central controller, and shunt dissipator are attached as Appendix III-1.

BATTERY

In developing a modular power system it is important that standardization be a prime requirement. To avoid multiple batteries with various energy storage capabilities, one battery was developed using the 20 ampere-hour standard equipment nickel cadmium cell defined by the NASA Low Cost Systems section.

A 16 cell battery was selected as the baseline design for this modular system design. A minimum of two batteries are provided per system with additional batteries to be added as eclipse loads increase such that n-1 batteries can satisfy load requirements (n being the number of batteries provided).

A low voltage battery (charge and discharge voltage below the regulated bus voltage) was selected as it does provide a smaller modular building block with respect to watt hour capacity and physical parameters. Also with fewer series cells than a battery above the regulated bus voltage, the low voltage battery is a higher reliability design.

To avoid oxygen pressure build-up and hydrogen generation at low temperatures, a maximum charge rate of C/4 was selected. This charge rate limits the depth of discharge to about 22% for low earth orbit spacecraft due to the charge time available.

For geosynchronous orbits, a limit of 55% depth of discharge was used to provide a long life battery (4 years).

BATTERY CHARGER

A low earth orbit requires a high efficiency battery charger to minimize the solar array needed for energy balance. Charger efficiency greater than 85% dictates a switching non-dissipative type such as a buck regulator.

Regulator operation is controlled by three parameters; central control, battery charge current, and temperature compensated battery voltage. The central control parameter is an analog signal which is proportional to the excess solar array power available that can be used for battery charging. Battery charge current will be limited to C/4 or 5 amperes. The temperature compensated signal will terminate the high current charging when battery voltage reaches a preset level by maintaining a constant voltage on the battery and causing a taper charge until full charge is accomplished.

A disconnect switch will be included in the charger to isolate the battery and prevent charge/discharge operation. A diode is provided in the battery discharge path to prevent a battery failure from loading down the other paralleled batteries and also to prevent charging by other than its own charger.

The minimum spacecraft power system would contain two batteries each with its own charger. When the chargers are being used to condition the shuttle fuel cell for spacecraft usage, the 5 ampere charger limit and conversion and distribution losses limit the steady state spacecraft load to 154 watts, and any additional loads will discharge the 2 batteries.

Discharging the batteries during shuttle operations may not be desireable and so an investigation was undertaken to determine the charger current level required to support the spacecraft night load level when conditioning the shuttle fuel cell input. Night load power was selected because in many cases it may be necessary to operate housekeeping subsystems and some experiments to monitor performance and operation during the entire shuttle phase.

The right side of Figure 23 shows the average night load that can be supported by various numbers of batteries. The left side shows the power delivered to the loads via the battery charger path. It can be seen that if the charger current rate is set at 5 amperes, the power provided to the load from the shuttle fuel cell can be no greater than 0.53 or 0.71 times the night load level. However, if during shuttle operations the charger current rate is increased up to 9.5 amperes, 1.04 to 1.39 times the night load power level can be provided by the charger - boost regulator path.

A detailed block diagram of the battery charger is shown on Figure 24. The various controls for the buck charger electronics include the central control signal which prohibits the charger current from exceeding the excess solar array current; battery voltage and battery temperature which are combined to terminate constant current charging and operate in a taper charge mode; charger output current signal which limits the charge current; and the shuttle high current enable signal that can select the 9.5 ampere current limit during fuel cell conditioning for spacecraft use.

The relay K2 shows functionally that when the fuel cell voltage is present, the battery charger is disconnected from the regulated 28 volt bus.

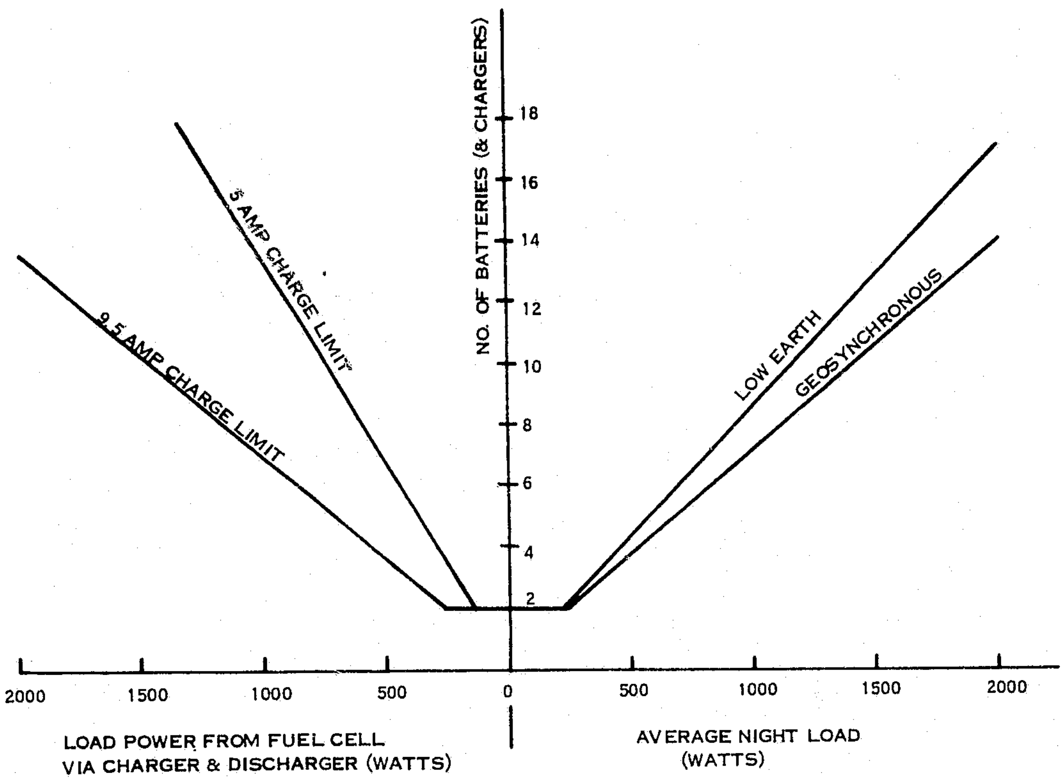


Figure 23. Battery/Charger Load Power

The buck charger electronics block diagram of Figure 25 shows modular circuits such as the duty cycle controller, base drive switching, and Jensen oscillator that are used in the discharge regulator.

BOOST REGULATOR

The boost regulator maintains a constant spacecraft bus voltage during those periods (eclipse, transients, pulsing load currents) where the primary power source is incapable of supplying the total spacecraft loads. This PWM switching regulator effectively adds voltage pulses to the available input voltage and then smooths them to a constant DC level. The energy is extracted from the battery source (or fuel cell via the PWM charger) and is pulse width-modulated and boosted to the spacecraft voltage level via an autotransformer. The modulating control signal is generated in the central controller and is proportional to the error voltage generated at the spacecraft bus.

In order to define the boost regulator requirements and achieve a standard conceptional modular design, several factors must be taken into consideration.

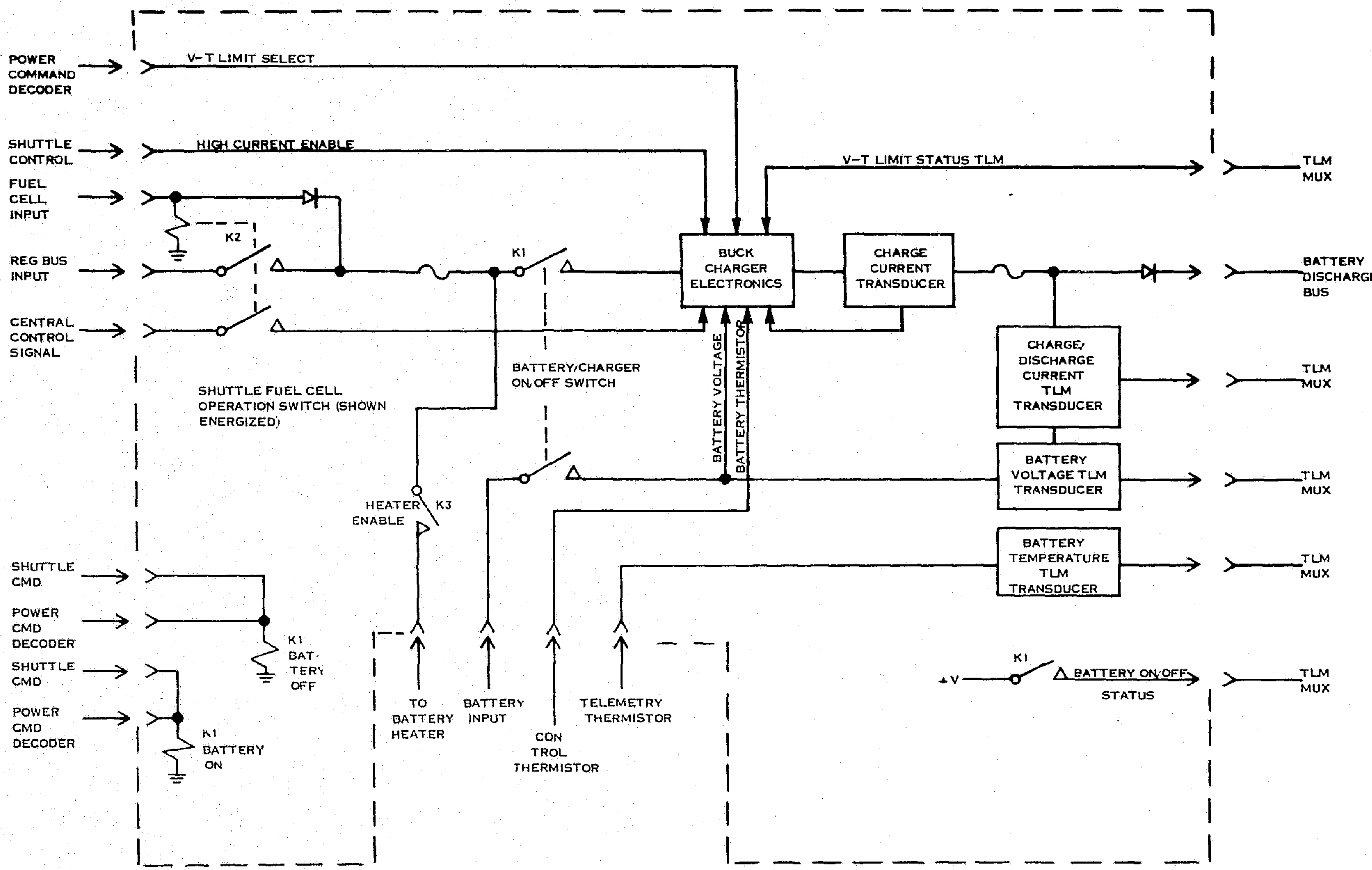


Figure 24. Battery Charger Block Diagram

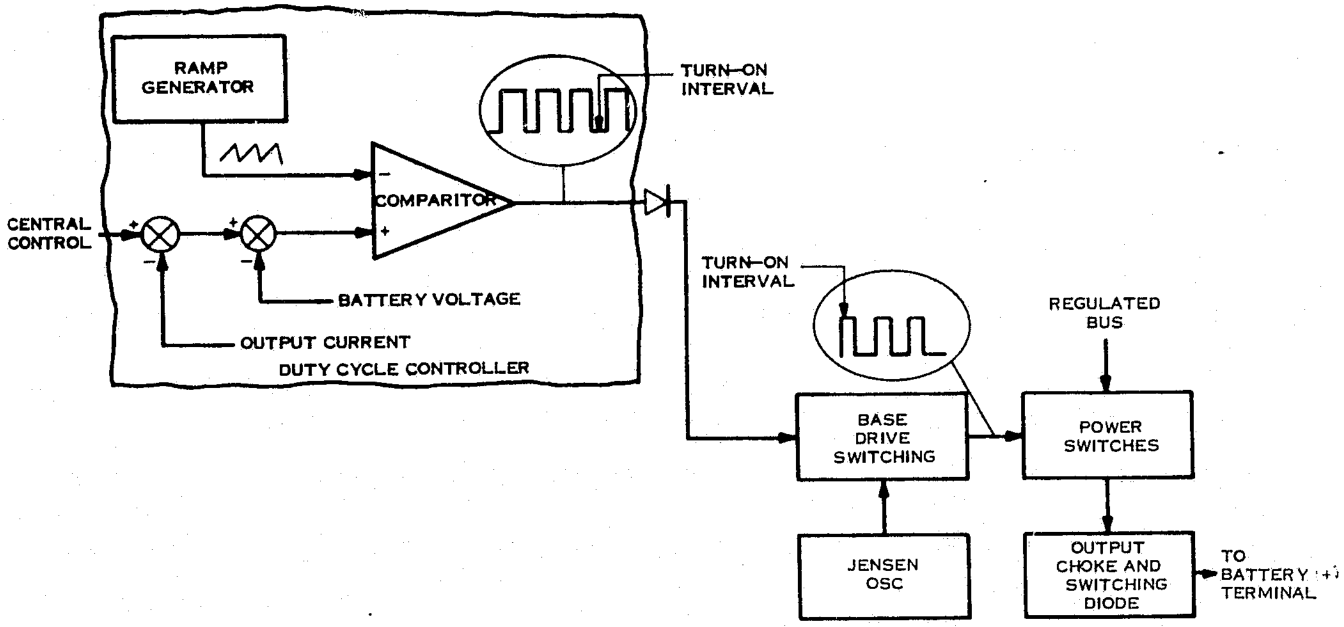


Figure 25. Buck Charger Electronics Block Diagram

From the Task I effort, the spacecraft power levels and the quantity of spacecraft per level is shown in Figure 26. The two systems at 4.5 and 6 kilowatts are considered to require special power systems and are excluded from this study.

Therefore, up to 2 kilowatts of night power must be provided thru the boost regulators, but single aerospace boost regulator rated at 2 kilowatts is beyond present day technology. A one kilowatt design is possible and parallel operation would provide the two kilowatt capability.

A one kilowatt regulator operating with an 85 percent efficiency would develop 172 watts to be dissipated from a rather small volume. In fact, the heat generators fall in 3 major but smaller volumes of the boost regulator. They are the switching transistor, the output diodes, and the output filter. Assuming equal heat distribution, the three small volumes would require heat sinking of 58 watts each. These small volume, high heat generators would unduely complicate the thermal subsystem design for a 1 kilowatt regulator.

Since the 2 kilowatt design is not in todays technology and the 1 kilowatt design is thermally complicated, consideration was given to a design between 200 watts but less than 1 kilowatt. Figure 26 indicates that a 600 watt regulator would satisfy more than 75% of all spacecraft designs from now until the year 1991.

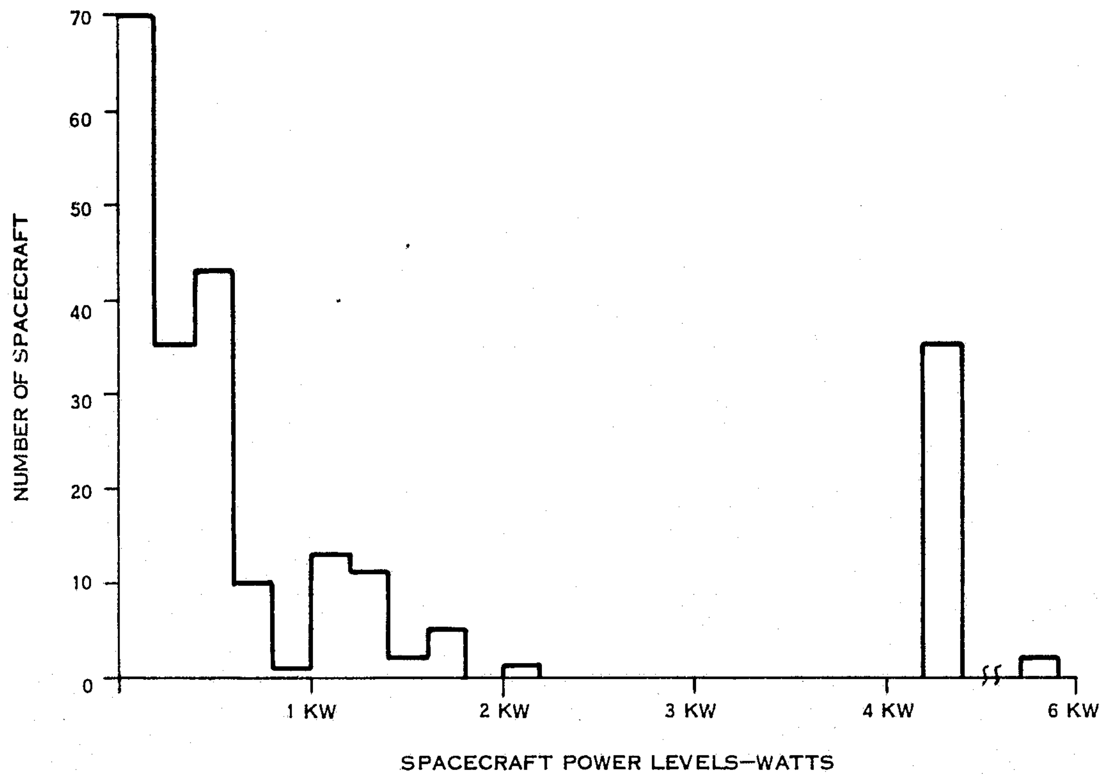


Figure 26. Spacecraft Power Requirements

Accepting the 600 watt regulator as a design requirement, it is now possible to see that 4 regulators operating in parallel would handle the 2 kilowatt night load requirement. Several subtle advantages are also achieved with a 600 watt regulator: (1) the transient power capability can be up to the kilowatt region without introducing a complex thermal design; and (2) the 600 watt power level reduces the design complexity of the voltage and current feedback loops within the regulator. Further, a 600 watt modular design will enhance the spacecraft thermal design by virtue of lower dissipation per module and provide mobility within spacecraft configurations that must handle power levels greater than 600 watts.

A requirement developed in the Failure Modes Analysis section to follow is that the boost regulator be fail-safe to avoid failures that would turn the device on and subsequently discharge the batteries. Figure 27 which is the block diagram of the boost regulator shows how this requirement is met. Two duty cycle controllers are used and both must signal a turn-on pulse to the succeeding stages before action is taken. Both outputs of the duty cycle controller must go low to turn on.

As mentioned in the discussion of the battery charger, many circuits are identical with the differences being in the output stages. (Detailed schematics located in a later section are provided for inspection).

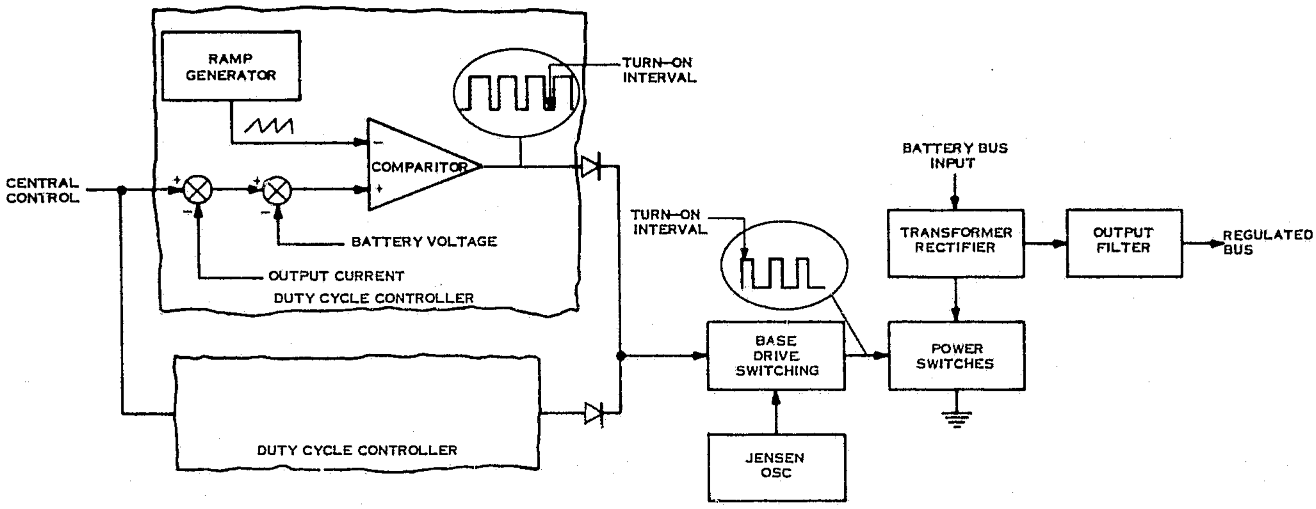


Figure 27. Discharge Regulator Block Diagram

CENTRAL CONTROL

The central control provides the appropriate signals to the shunt dissipator, charger, and discharge regulator to maintain bus regulation. The central control signals are proportional to the bus voltage deviation with the range of acceptable deviation nominally selected to be ± 1.5 percent of the bus voltage.

Figure 28 can be used to expiain system control.

At the upper limit of voltage deviation the shunt regulator is turned full on; full charge demands are satisfied depending on the battery status. With higher load demands or decreased array power, the shunt regulator dissipation is decreased, and completely turned off when the voltage deviation is around +0.5 percent. With further load demands, the array power is preferentially supplied to the load by gradually decreasing the available charge power to a point where charging is totally inhibited at a voltage deviation of around -0.5 percent. At this particular condition, the array power just satisfies the load demand. Further load demands are supplied by the discharge regulator, which is at a full-on condition at a voltage deviation around -1.5 percent.

Solar array power priority is demonstrated by Figure 29 which shows the V-I curve of the array and the operating point forced by the central control. When load demand increases, the operating point is moved down to the right of the V-I curve. Central control senses the lower voltage and removes shunt dissipators to compensate and return to the proper operating point.

Figure 30 shows the basic block diagram of the central control. Because single failure modes would result in loss of the power bus, all functions shown are enchanced by the use of majority voting or quad redundancy which is shown in the detailed schematic of a subsequent section.

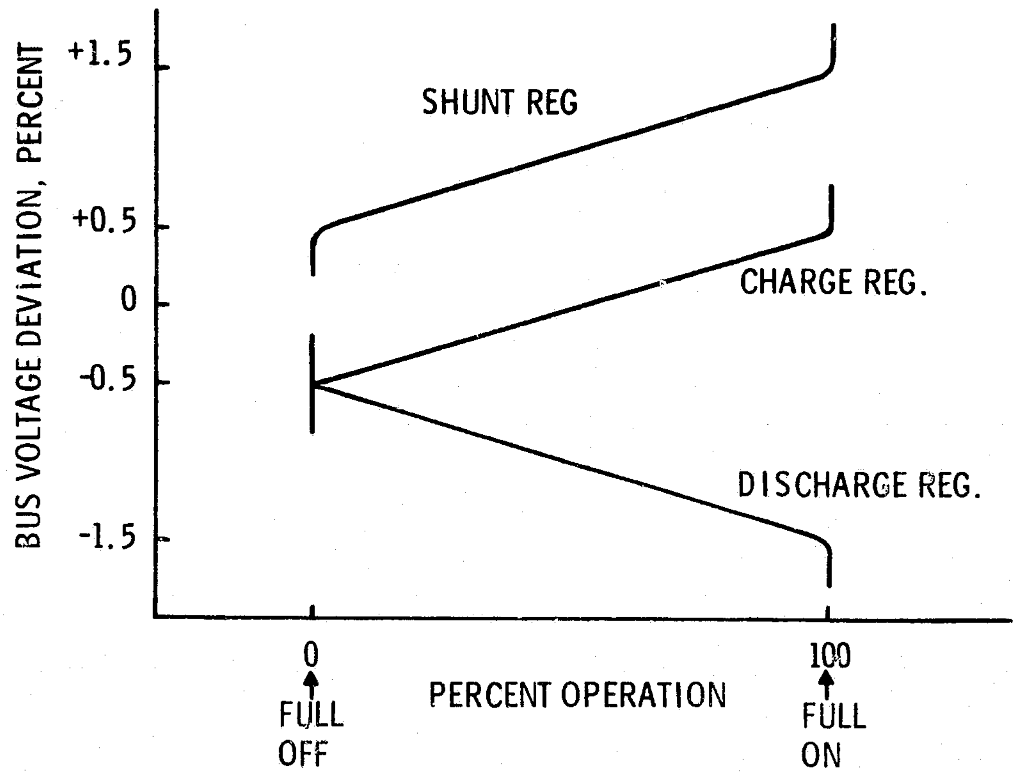


Figure 28. Control of Bus Voltage

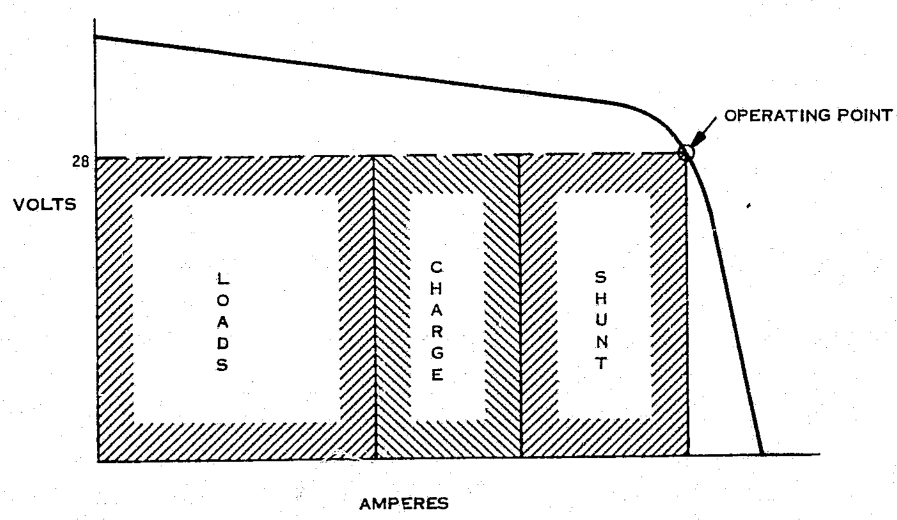


Figure 29. Solar Array Power Priority

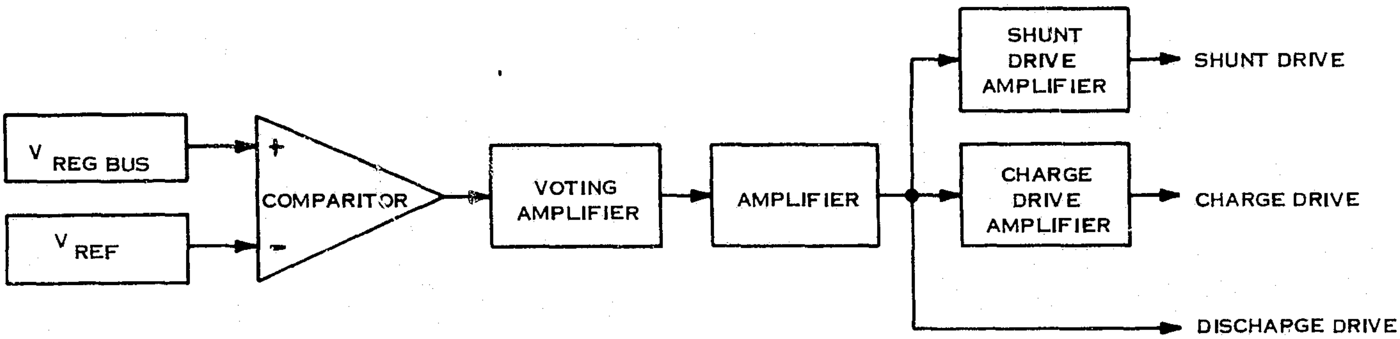


Figure 30. Central Control Block Diagram

SHUNT DISSIPATOR

The function of the shunt dissipator is to absorb the excess solar array power to force constant voltage operation.

Initial designs considered the use of a full shunt as a modular design could be developed independent of the solar array cell circuit configuration. There would not be any electrical interfaces to the shunt other than the 28 volt bus and the central control signal. However, thermal studies proved this approach to be impractical because the radiating area required for a full shunt dissipator becomes prohibitively large. Consequently, a partial shunt approach located on the solar array was selected.

The partial shunt dissipator requirements developed during the study are as follows:

- The design must provide for parallel operation with a common central control signal.
- The modular design is to control a 500 watt solar array segment.
- The shunt elements will be sequenced to minimize peak electronics power on the dissipator panel.

A functional block diagram of the shunt dissipator is provided by Figure 31. The shunt drive signal from the central control crosses the slipring to the power amplifier which, thru a sequencer, operates the 12 shunt elements required to control a 500 watt array segment.

The lower left of Figure 31 shows the principle of operation of the partial shunted solar array. Briefly the voltage of the lower array is varied by changing its current such that the sum of the upper and lower array voltages is equal to the regulated voltage which in this case is +28 vdc.

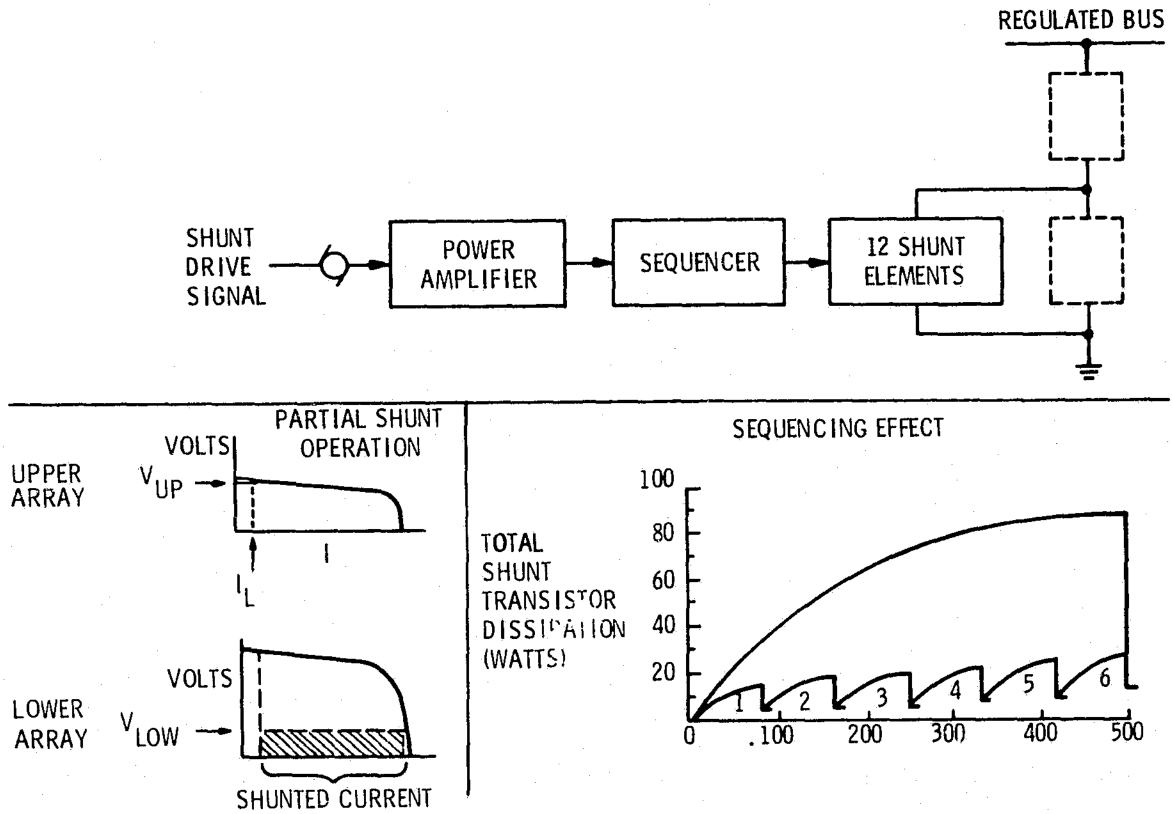


Figure 31. Shunt Dissipator Block Diagram (500 Watt Array Segment)

The effect of sequencing the 12 shunt elements is shown in the lower right of the figure. The upper curve shows the power dissipated in the shunting transistors if all 12 were operated simultaneously in parallel. By sequential operation, the first pair are turned full on before the next pair of shunt transistors start to dissipate. This sequential operation provides the lower transistor power dissipation profile.

Obviously, the shunt dissipator design is very dependent on the solar array circuit configuration and as standardization of the solar array is beyond the scope of this study, the shunt dissipator design presented is only representative of the approach that might be used. Figure 32 shows the mechanical design of a shunt dissipator to control a 500 watt solar array. Transistors Q7 thru Q10 are in the quad redundant power amplifier stage. Terminal boards TB1 and 2 contain the sequencer circuits. The remaining transistors are the 12 array shunt elements.

FAILURE MODE ANALYSIS

The intent of this analysis was to review each function of the power system to determine the effects of failure modes, and to provide protection where failures result in serious overload or catastrophic loss of capability.

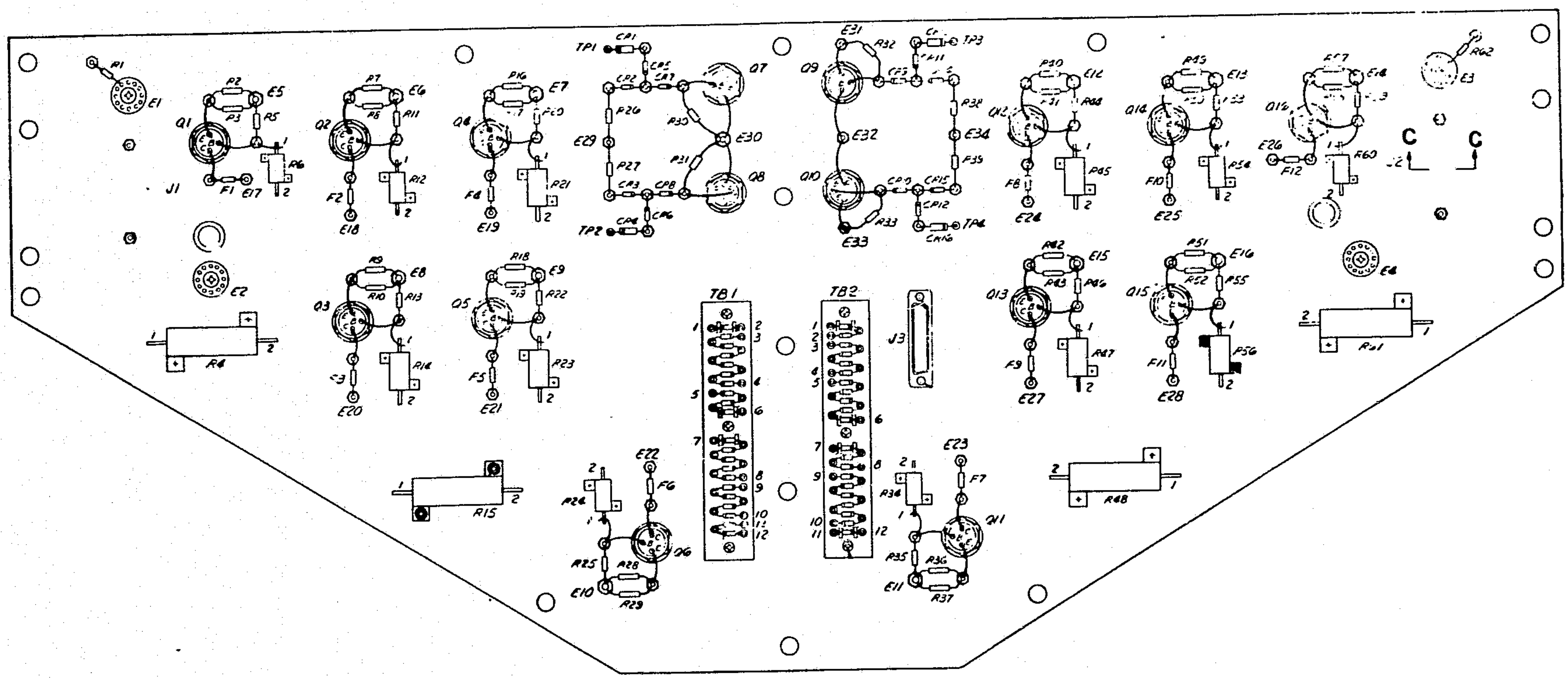


Figure 32. Shunt Dissipator Assembly

BATTERY

Generally, all battery failures result in loss of the battery to the power system. An isolation diode is provided in the battery charger to prevent a shorted battery from discharging remaining batteries.

Loss of a battery can be compensated by either providing n + 1 batteries where n is required to support the load, or the night loads may be reduced to compensate for battery loss.

The former alternate was selected for this study.

CHARGER FAULT PROTECTION

As previously mentioned, the power system must perform after the loss of one battery. This loss may be the result of battery failure, or charger failure. Charger failures can be classified as open circuit or full-off types, short circuit or full-on types, or high resistance types that are neither of the above, but present a power drain to the power system.

Each type has been investigated to insure protection is adequate to prevent propagation of failure to other equipment or insure that continuous power drains can be removed from the power bus.

Failures that result in the inability to pass current from the solar array bus input to the battery are designated as open circuit failures. The result is loss of the battery due to the inability to recharge. This is acceptable.

Failures that result in the inability to stop or control the current from the solar array bus into the battery are designated as short circuit failures. Also included in this category are failures that present a continuous power drain on the regulated bus.

From a fault clearing viewpoint the worst time for a charger short circuit to occur is during the night portion of the mission.

During the day, the solar array provides the load power required, and so the boost discharge regulators are available to handle an overload. However, at night operation, the boost regulators are operating to satisfy normal night loading. Figure 33 shows the elements of the power system associated with this fault situation. The effect on the regulated bus can be examined by determining the reserve capability of the discharge regulators.

A 20 ampere (560 watt)* capability boost regulator was used in this analysis. As with the battery, the boost discharge function must be redundant to allow one unit failure. Effectively, there is one spare boost regulator and therefore 560 watts of spare capability.

---

*When this analysis was performed, the 600 watt power handling capability of the boost regulator had not been selected. Twenty amperes was used as a representative number which was felt to be close to the yet-to-be selected power level. The conclusions of this analysis would not be changed by the less than 10 percent change in power level.

The diagram of Figure 33 was used to determine the magnitude of current thru a shorted battery charger for the conditions shown on Figure 34. These results show that the "shorted" charger does not cause an undervoltage on the regulated bus until the fault current exceeds 20 amperes. This is attributed to the extra 20 amps of boost regulator capacity included to allow boost failure. The current magnitude beyond this point is a function of the series resistance of the failed charger, the battery voltage, and other parameters shown on Figure 34. If a fuse was used to clear this fault, the fuse should blow at the current at which an undervoltage would take place and it should open the fault circuit in a reasonable time. In this case, 20 amps should blow the fuse.

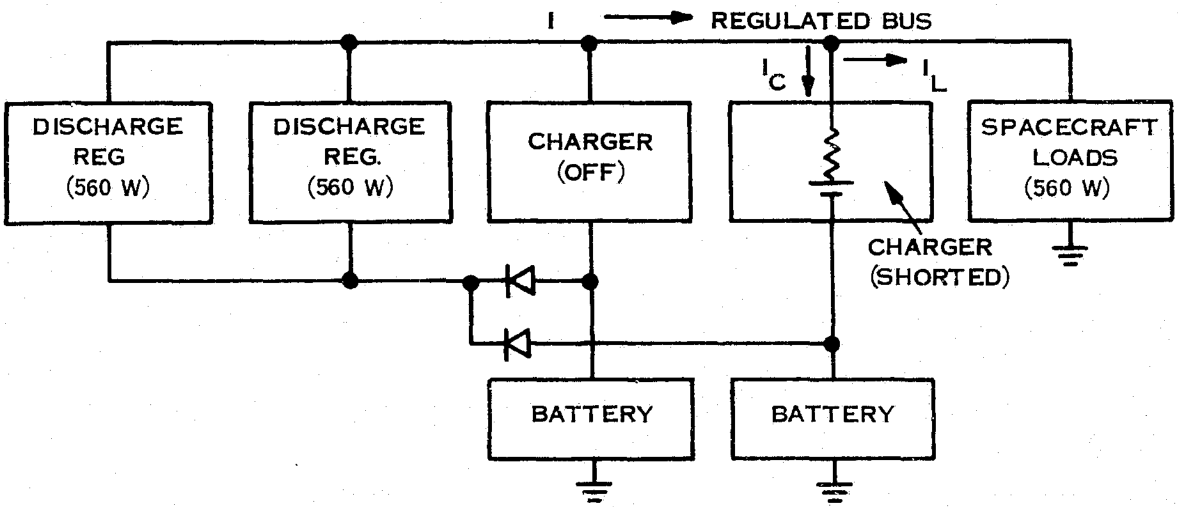


Figure 33. Full-on Charger Failure

The characteristics of the high reliability Pico-Fuse used extensively on the Nimbus/ Landsat spacecraft for many years are shown on Figure 35.

To allow for fuse derating, the minimum fuse size for this application must have a 10 amp rating.

From Figure 35, it can be seen that at the current magnitudes that cause regulated bus undervoltage, the 10 amp fuse blow time is between 70 and 500 milliseconds which is acceptable.

Any short circuit failure of the charger that does not draw enough current to blow the input fuse can be cleared by command as the voltage of the regulated bus would remain within spec. A second pole of the battery ON/OFF relay is connected to the input of the charger and can disconnect all input power as shown on Figure 36.

The output of the charger must also be fused to prevent high battery discharge current into a failed "fly-back" diode or output capacitor in the charger.

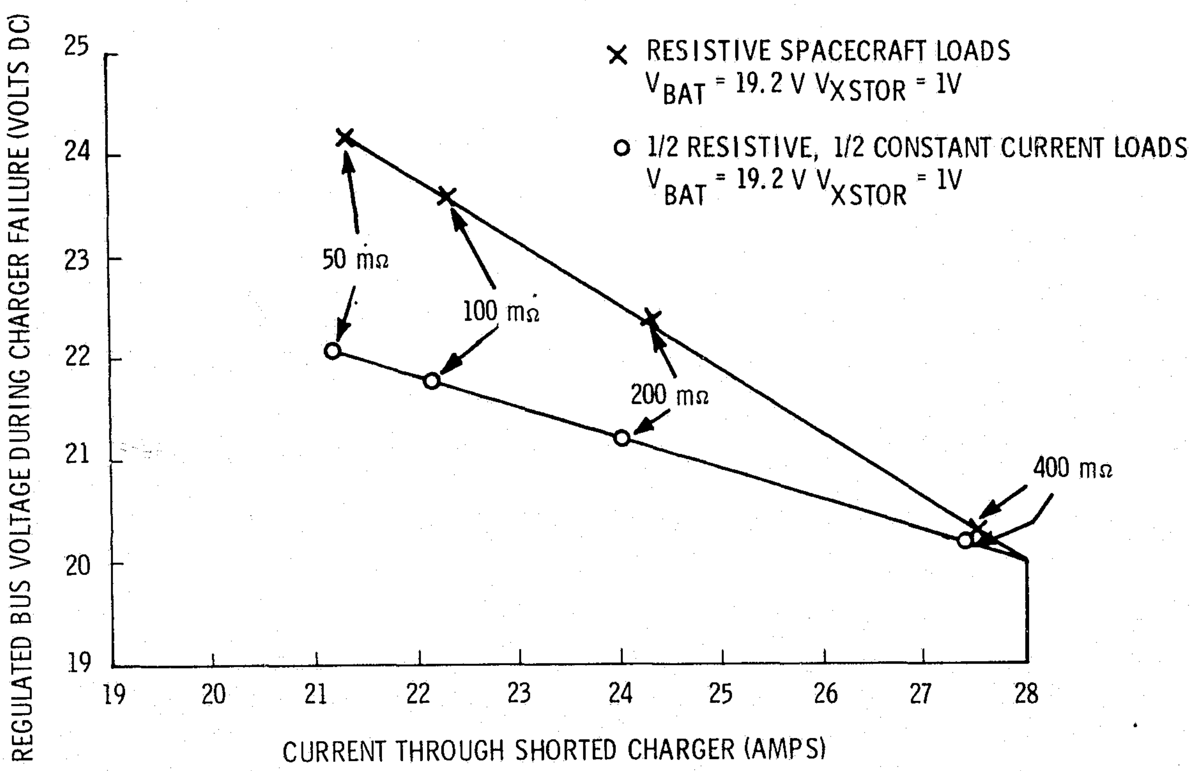


Figure 34. Current Through a Shorted Battery Charger

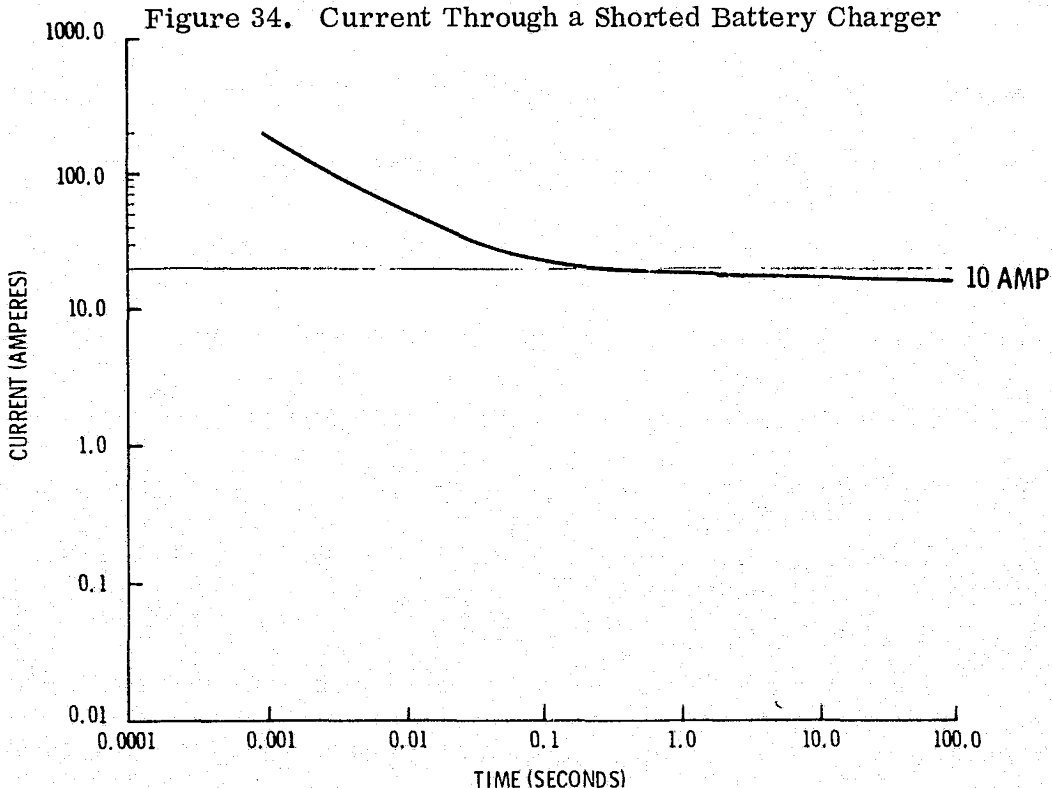


Figure 35. Hi-Rel Picofuse Time/Current Characteristics

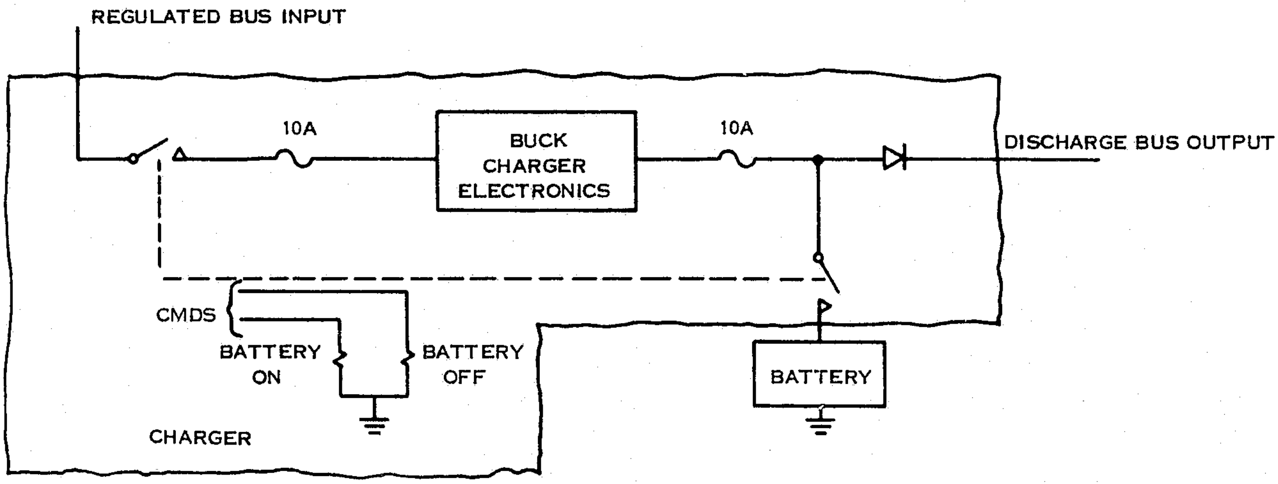


Figure 36. Battery Charger Fault Protection and Isolation

CENTRAL CONTROL

Single point failures within the Central Control function would result in loss of the power bus, and therefore loss of the spacecraft. If the error amplifier failed and its output went "High", all the shunt dissipators would be forced on causing a very low voltage operating point on the solar array V-I curve. Also if the failure resulted in a "Low" output, the battery discharge regulators would come full on resulting in a high overvoltage on the regulated bus.

To prohibit these single point failures, the central control design uses three error amplifier stages that are majority voted such that two of the three must agree on the operating mode of the power system. Quad redundancy is used for the amplification stages after the majority vote stage. A detailed circuit design is provided in the Detailed Schematic section of this report.

DISCHARGE REGULATOR

To provide for a failure that results in loss of operation or "full-off" condition, n + 1 discharge regulators are provided where n is the number of regulators required to support the mission.

A failure that causes operation at or near 100% duty cycle ("full-on") is not acceptable and is prohibited by design. To accomplish this, the design is made "fail-safe" (any failure results in a "full-off" state) by the use of two duty cycle controllers within the discharge regulator. Both circuits must agree to turn-on before operation commences. Design implementation is shown in the Detailed Schematic section.

SHUNT DISSIPATOR

The typical design of the shunt dissipator previously described can control a 500 watt solar array segment. Single failures can result in loss of control over 36 watts of the array due to an open circuit failure or continuous shunting of 36 watts because of a shorted transistor. Both failure modes are considered acceptable and the use of redundancy or fault protection is not justified.

During the analysis of the shunt dissipator, a failure on the solar array was identified which due to the partial shunt design, results in an unacceptable failure mode for which protection must be provided.

Figure 37 will assist in explaining this failure. A typical solar array circuit is shown on the left of the figure. An isolation diode is provided at each solar array circuit to prevent a failed circuit from loading the regulated 28 volt bus. Assume for discussion that the shunting transistor labeled sequence 1 is in saturation and the diode fails short circuited. The voltage developed by the upper array segment is less than the 28 volt bus and as can be seen from the upper right figure, when the voltage across the array increases, large reverse current will flow from the regulated bus, thru the upper array and down thru the saturated transistor. A fuse has been added in the collector of the transistor to limit the duration of this failure mode.

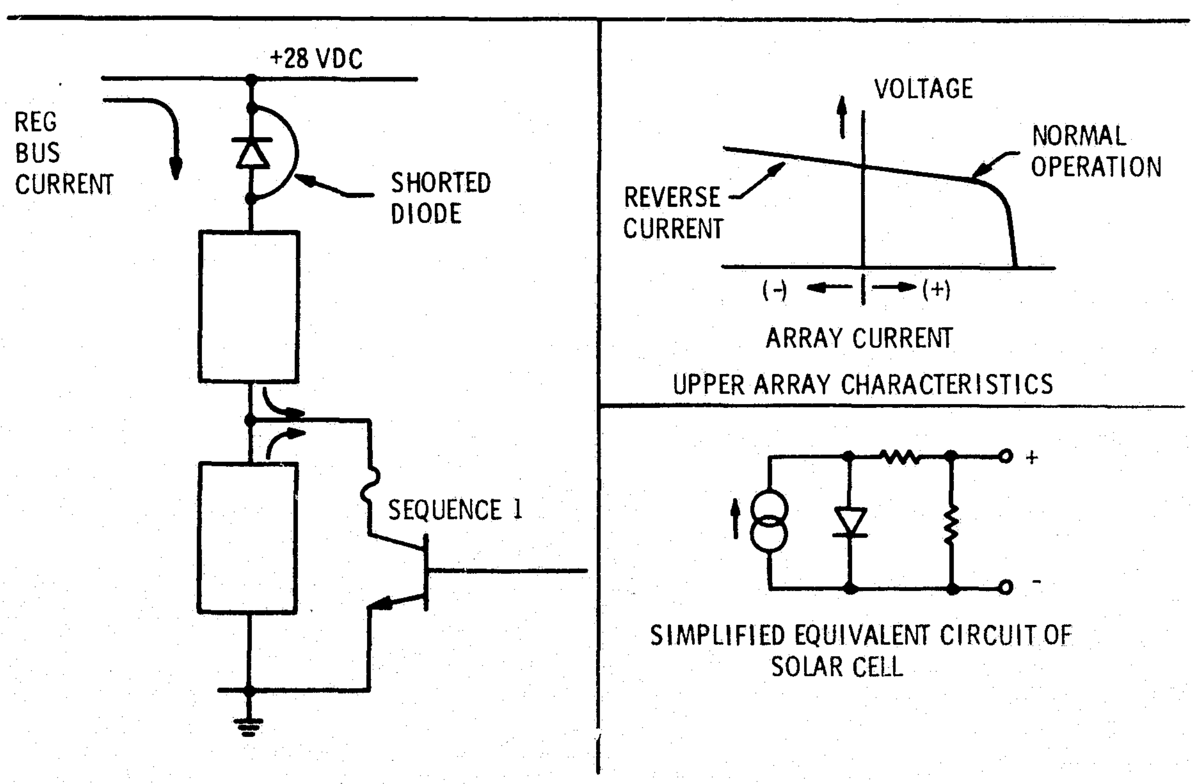


Figure 37. Failure Modes Solar Array Isolation Diode

DETAILED CIRCUIT DESIGN AND HYBRID CANDIDATES

Schematic designs of each of the power conditioning elements were developed during this study for two related reasons. One was to identify those circuits which are candidates for hybrid thick film packaging and second, to define the size, weight, and power requirements for each of the functional elements.

Preliminary specifications which provided the design requirements for the battery charger, discharge regulator, central control, and the shunt dissipator are attached as Appendix III-1.

Figure 38 presents a simplified single channel (without redundant circuitry) schematic of the control circuits of the modular power system. Notice that the input circuits of the discharge regulator and battery charger are identical, thus providing modularity at a level below the functional unit. The asterisk shown at various transistors indicates that quad redundancy is used for those circuits.

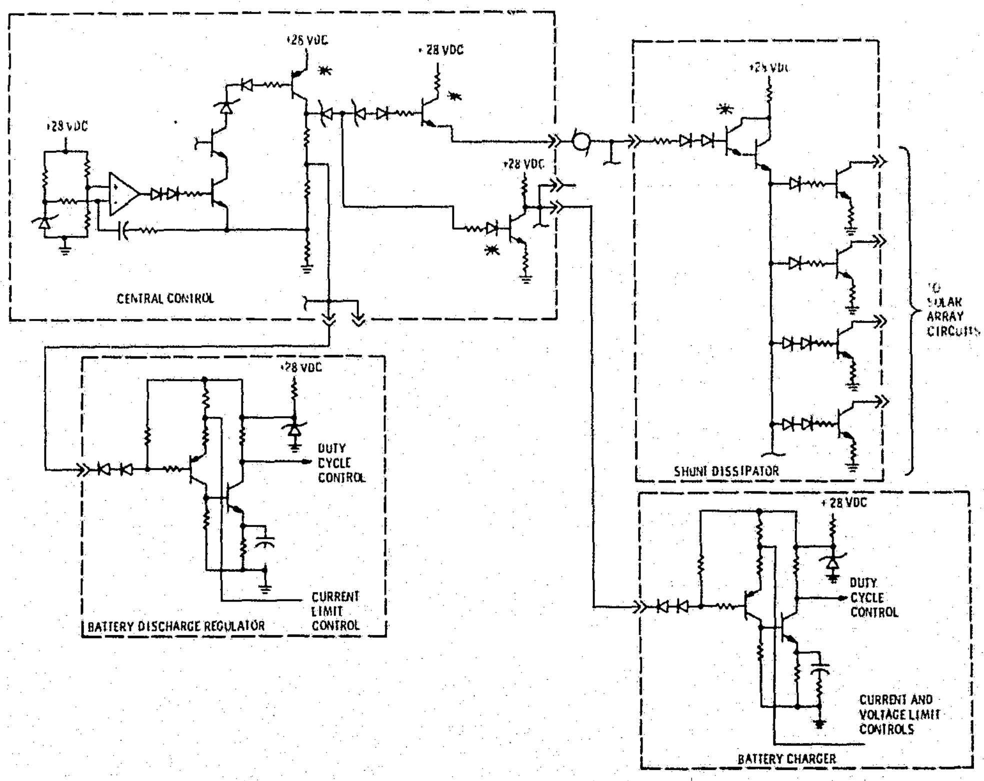


Figure 38. Modular Power System Control Circuits

HYBRID DESIGN CRITERIA

Identification of hybrid circuit candidates in the circuitry to follow, was based on design criteria used at GE Space Systems for power equipment.

- No inductors, capacitors greater than 0.1 u farad, nor .1% resistors less than 100 ohms are included in a hybrid package.
- A maximum power density of 1.25 to 1.5 watts per square inch is used to limit junction temperature to 150°C in a Kovar package.
- Standard package sizes of 1.2 and 2.0 inches are used.
- The number of components is kept below 50 to avoid test and yield problems associated with large hybrids.

DETAILED SCHEMATICS

Hybrid circuits on the equipment schematics are identified by a dotted line around the circuit elements.

The battery charger of Figure 39 identifies the Jensen oscillator, the duty cycle controller, and the current/voltage limit circuits as hybrids. The output power stage, mag amp current transducer, chokes, and relays are packaged on printed circuit boards or are chassis mounted for heat rejection.

The discharge regulator of Figure 40 has three hybrids; the two duty cycle controllers and the Jensen oscillator. All other circuit elements are discrete parts.

The central control of Figure 41 consists entirely of hybrids with the exception of a few select by test resistors, filter capacitors, and divider networks.

Because of the power levels in the shunt dissipator of Figure 42, no hybrids are used. The circuits shown dotted are the solar cell circuits of which a typical design is provided in the box on this figure.

SUMMARY OF EQUIPMENT MECHANICAL CHARACTERISTICS

From the preceding schematic designs, preliminary equipment specifications, and analyses, the size, weight, and power dissipation of the various power subsystem equipment were determined as shown on Table 25. The characteristics of the remote decoder multiplexer, which is considered a telemetry-command system unit placed in the power system, were derived from previous internal studies which developed this equipment.

This equipment defined by Table 25 can be used as required for conventional spacecraft packaging thus providing modularity at the equipment level. However, foreseeing the needs of automated interchangability required for Shuttle generation spacecraft, a concept defined as a Power Module was developed and its capabilities evaluated in the following section.

TABLE 25. COMPONENT MECHANICAL/THERMAL SUMMARY

COMPONENT	SIZE (CM) (INCHES)	WEIGHT (KG) (LB)	MAX POWER DISSIPATION (WATTS) LOW EARTH ORBIT NIGHT 1/3	LOW EARTH ORBIT DAY 2/3	GEO SYNC NIGHT (1.2 HR)	GEO SYNC DAY (22.8 HR)	VOLUME CM^3	VOLUME IN^3	POWER HANDLING CAPACITY	POWER PER CM^3	POWER PER IN^3	POWER PER KG	POWER PER LB
BATTERY	20.3 x 22.9 x 20.3 8 x 9 x 8	20 44	35	7.5	40	6.8	9439	576	394 WATT-HR	0.042	0.684	19.7	8.95
CHARGER	15.2 x 24.1 x 21.6 6 x 9.5 x 8.5	4.1 9	7.5	15	9	7	7948	485	120/228 WATTS	0.015/0.029	0.247/0.470	29.3/55.6	13.3/25.3
CENTRAL CONTROL	10.2 x 11.4 x 10.2 4 x 4.5 x 4	1.8 4	1	1 - 5	1	1 - 5	1180	72	---	---	---	---	---
DISCHARGE REG	21.6 x 24.1 x 12.7 8.5 x 9.5 x 5	5.4 12	100 MAX 17/BAT	4	100 MAX 25/BAT	4	6620	404	600 WATTS	0.091	1.48	111	50
REMOTE DECODER MUX	15.2 x 10.2 x 5.1 6 x 4 x 2	0.9 2	1.2	1.2	1.2	1.2	---	---	---	---	---	---	---
SHUNT REGULATOR	---	2.7 6	---	---	---	---	8898	543	500 WATTS	0.056	0.921	185	83.3

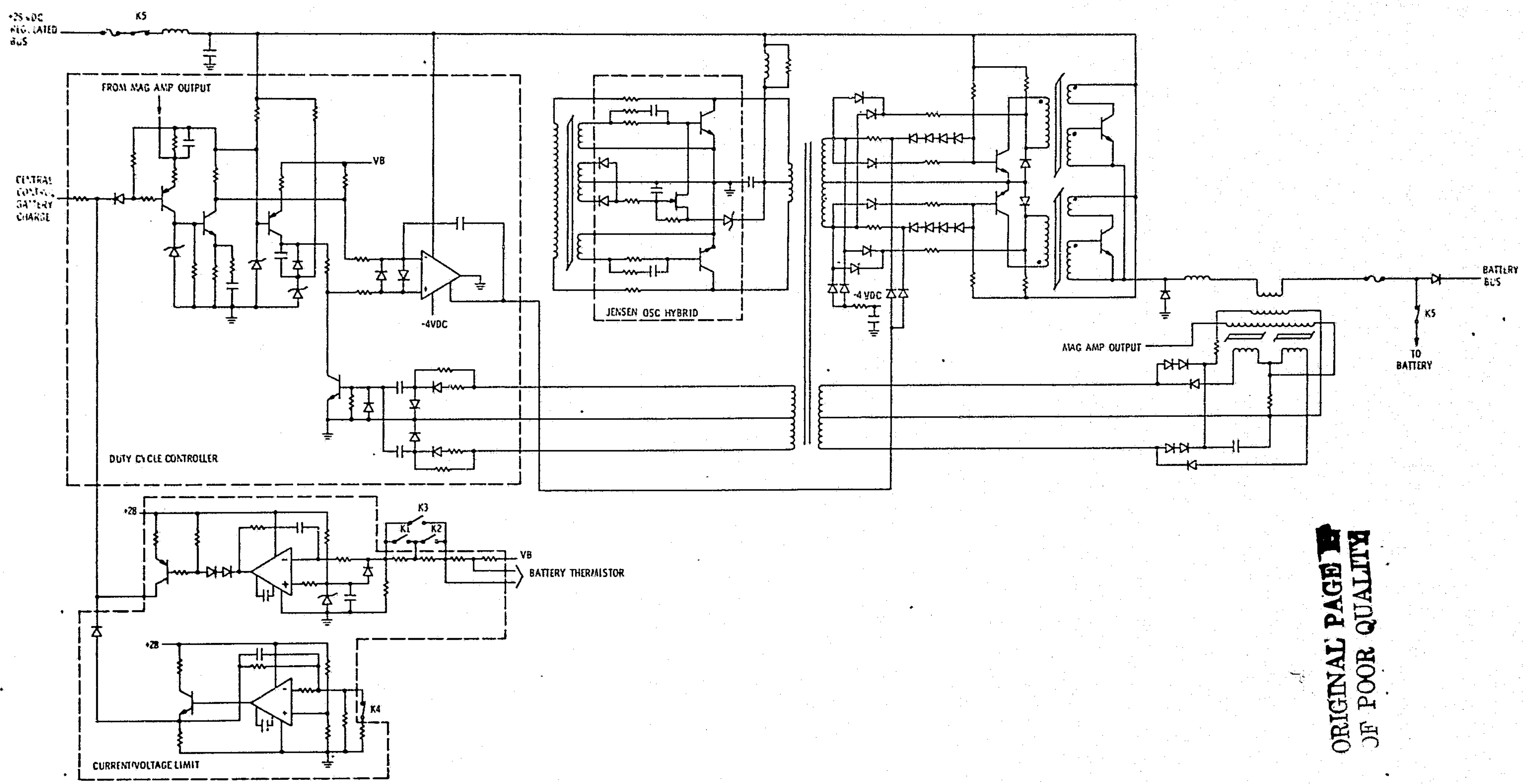


Figure 39. Battery Charger Schematic

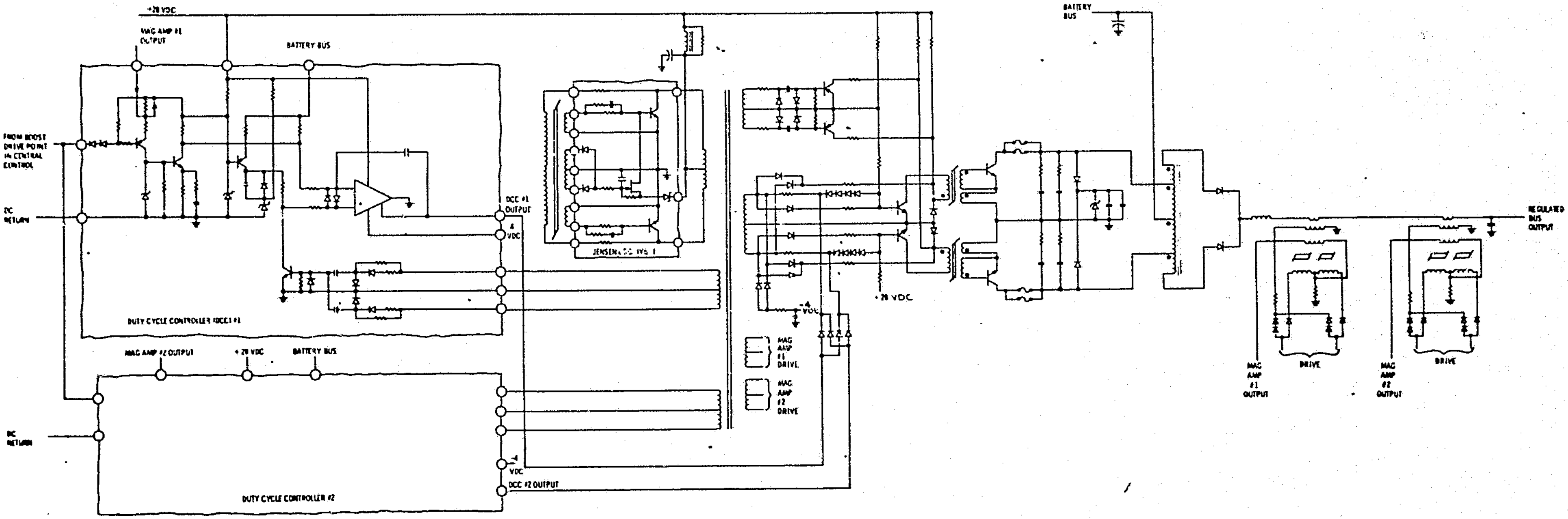


Figure 40. Discharge Regulator Schematic

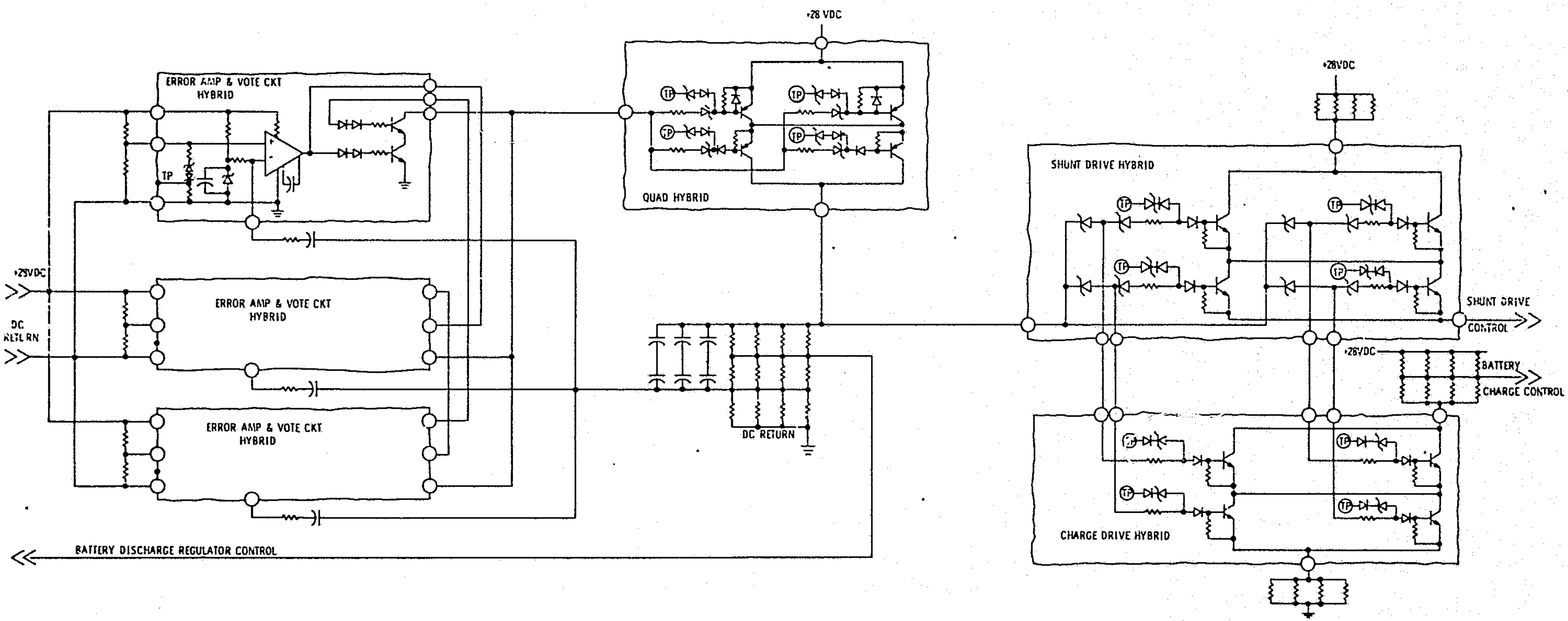


Figure 41. Central Control Schematic

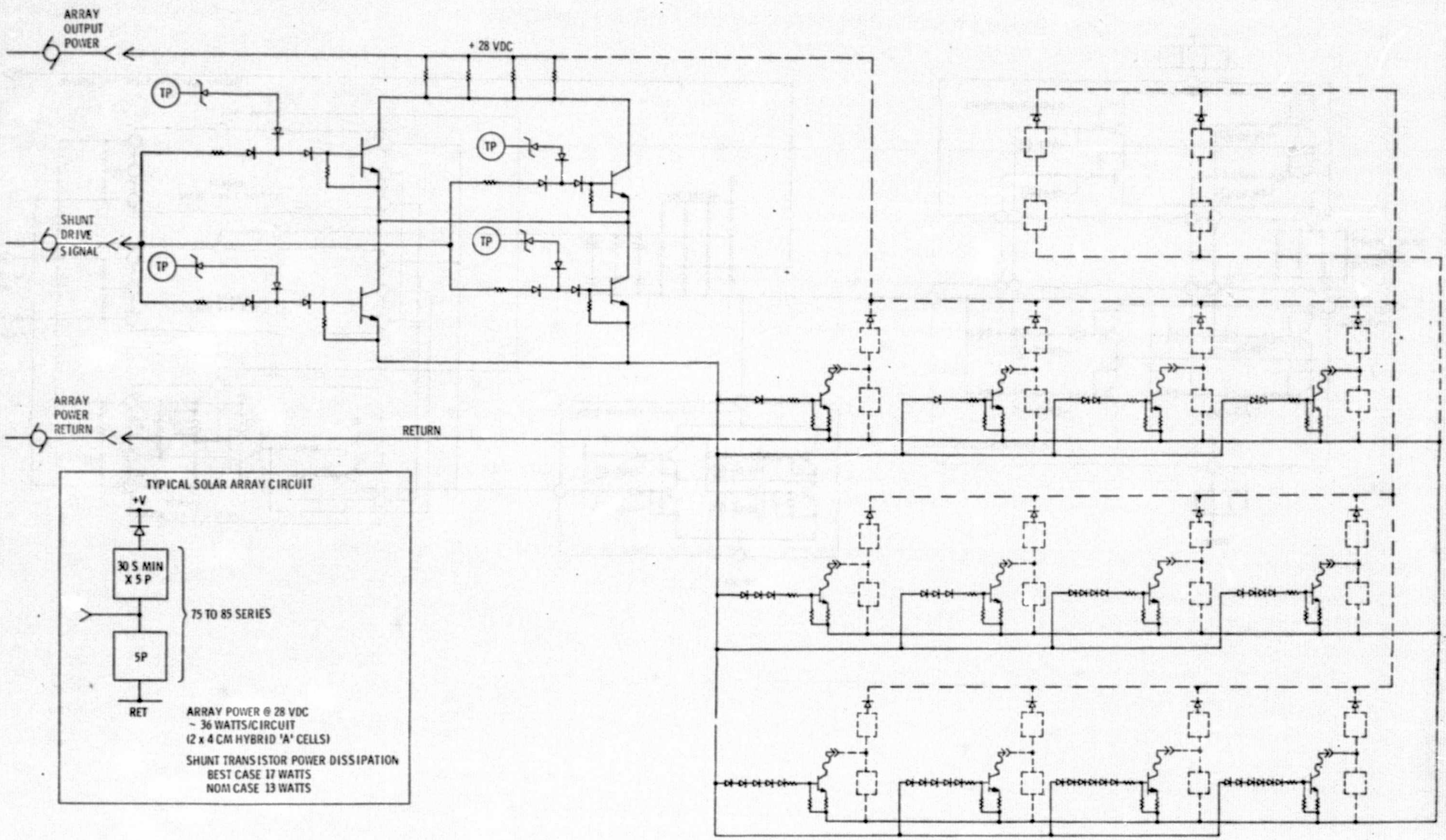


Figure 42. Shunt Dissipator Schematic

POWER MODULE ASSEMBLY

Having developed the modularized system to the equipment level, the next step was to assemble this equipment into a standard unit that may be attached to the spacecraft structure. This standard unit called the Power Module is shown on Figure 43.

The module is designed to reject all waste heat outboard with all side and inboard surfaces covered with multi-layer insulation blankets. High power dissipation components are mounted directly to the inner face of the aluminum honeycomb sandwich outer panel. The outer panel is integrally stiffened by keels tailored to the individual component arrangements. A subsystem harness interconnecting the components and interface and test connectors is designed for fabrication and installation as a unit. Once the harness is installed and clamped to the keel the module may be bench tested prior to installation of the frame structure and insulation covers. This "breadboard" subsystem assembly on the outer panel provides maximum ease of installation and replacement of components during the assembly cycle.

Once panel and harness assembly and test is completed, the panel is bolted to the open box frame structure and the interface and test connectors attached to the frame brackets. Installation of the insulation blankets completes the module assembly.

With the exception of the solar array assembly and shunt dissipator panel, all components are mounted within the Power Module. The shunt dissipator panel is mounted on the solar array.

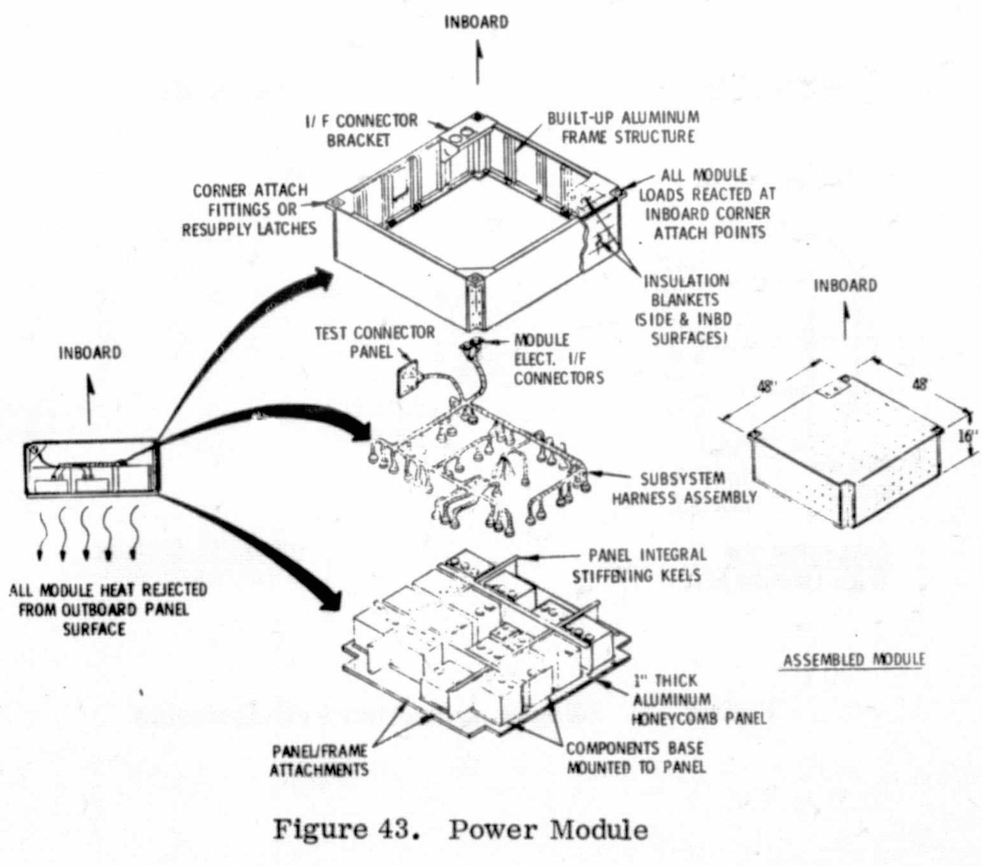


Figure 43. Power Module

Two sizes of Power Modules were used to determine the optimum packaging density of the equipment modules. These two sizes were selected as shown by Figure 44. Although shuttle launched spacecraft were the primary concern during this study, a standard modularized design should also be compatible with spacecraft launched by individual boosters. As the Delta is a popular vehicle for this purpose, it's size limitations were used to develop system module sizes. The rectangular arrangement of modules results in a basic size of 101.6 x 121.9 x 40.6 cm (40 x 48 x 16 inches). The triangular arrangement can accommodate a 121.9 x 121.9 x 40.6 cm (48 x 48 x 16 inches) module size.

The next step was to determine the maximum amount of equipment that can be packaged into each module to select the best size.

For identification purposes, the Power Modules will be refered to as the 40 x 48 and the 48 x 48 units, respectively.

Figure 45 shows the layout of the 40 x 48 module. Module No. 1 is the basic spacecraft unit and module No. 2 can be added to increase the power handling capability of the power system. Module No. 1 can accommodate 5 batteries, 5 charges, 3 discharge regulators, 1 central control, and 1 remote decoder multiplexer and provide the power as shown in the box. If module No. 2, which can contain up to 8 batteries with their chargers, is added, one of the batteries and chargers, is added, one of the batteries its charger of module No. 1 is deleted to make room for another discharge regulator to handle the additional power. The power capability shown in the box is the combined Module No. 1 and No. 2 maximum.

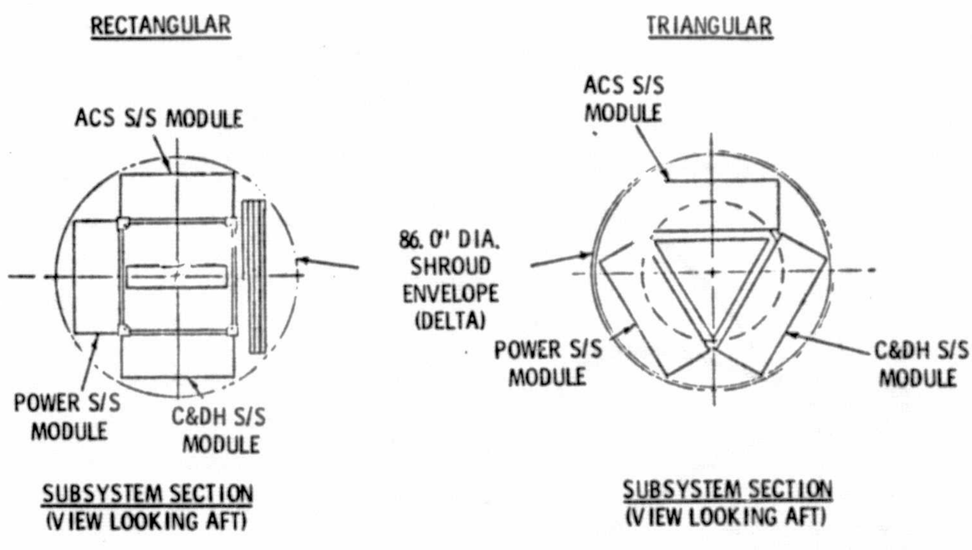


Figure 44. Subsystem Section Arrangements

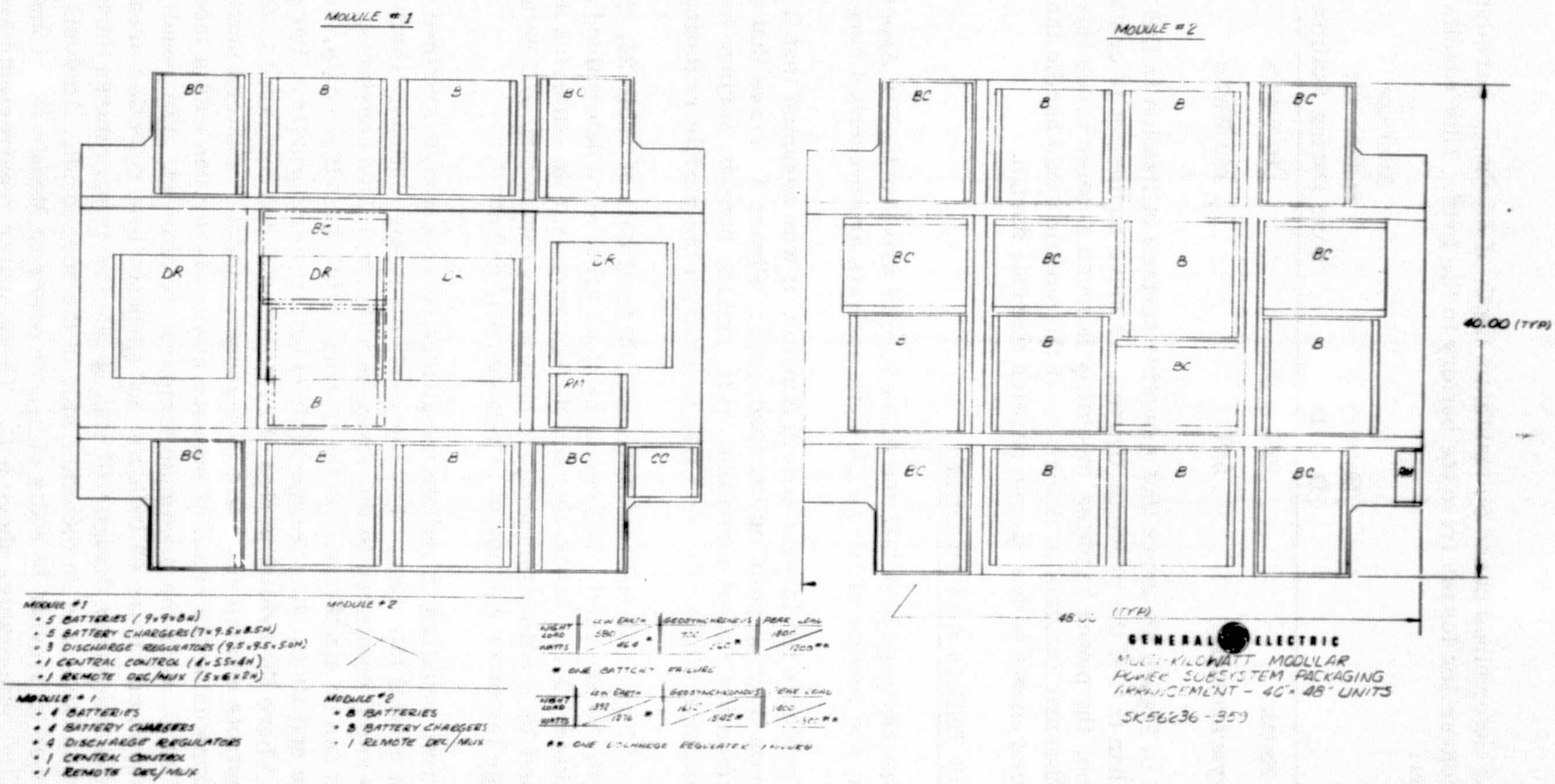


Figure 45. Packaging Arrangement
40 x 48 Inch Units

The power capability was determined based on maximum depth of discharge permissible and also considered power conversion losses from the battery to the load. This results in the load capability as shown below.

	Battery D. O. D.	Average Spacecraft Load During Eclipse
Low Earth	22%	116 Watts
Geosynchronous	55%	140 Watts

The layout of equipment in Figure 46 shows that mounting surface utilization in the 48 x 48 module is much better than that of the previous figure. As more equipment is contained in this 48 x 48 configuration, the power handling capability is much greater hence this configuration was selected for a thermal analysis to determine if the module could handle the heat generated in all these components with a simple passive thermal design.

THERMAL ANALYSIS OF THE 48 X 48 MODULE

Both Power Modules were analyzed in detail and these results are provided in Appendix III-2, but for the above reason, the results of the 48 x 48 unit analysis are presented here.

As the power module is probably the largest heat dissipator, it was assumed that it would be placed at the most optimum thermal point on the spacecraft. Figure 47 shows that this location is the north surface for a geosynchronous orbit, and the non-sun surface for a low earth orbit. These locations minimize the heat flux incident on the module radiating area.

Using the module layout of Figure 46 and the component thermal data of Table 25, the total power dissipation for both the low earth and geosynchronous orbit were determined as shown on Table 26. The limitations placed on the thermal design were that the batteries must be maintained between 0° and 25°C with a temperature spread no greater than 5°C between batteries. Also the range for other equipment is between 0°C and 40°C.

Based on this data, the area required to radiate the generated heat was determined for modules No. 1 and No. 2 for both orbital missions. For low earth orbit applications, where the orbital period of 100 minutes is small compared to the component thermal time constants, the module radiation areas required can be sized based on the average orbital environments, average orbital power dissipation and 10°C, the average desired battery temperature. For geosynchronous orbit applications, where the orbital period of 24 hours is large compared to the component thermal time constants and the sun angle is seasonal, module radiators must be sized for the maximum conditions using the summer solstice power dissipation with a module radiator temperature of 20°C, near the maximum temperature allowed. The resulting radiator area requirements shown on Figure 48 indicate that adequate heat rejection area is available in all cases. The geosynchronous design will experience low temperatures during the winter solstice period when the sun is on the other side of the spacecraft. Analysis of this condition identified the need to provide 16 watts of heater power to Module No. 1 and 7 watts to Module No. 2 to keep the temperature above 0°C. These heater requirements are acceptably small.

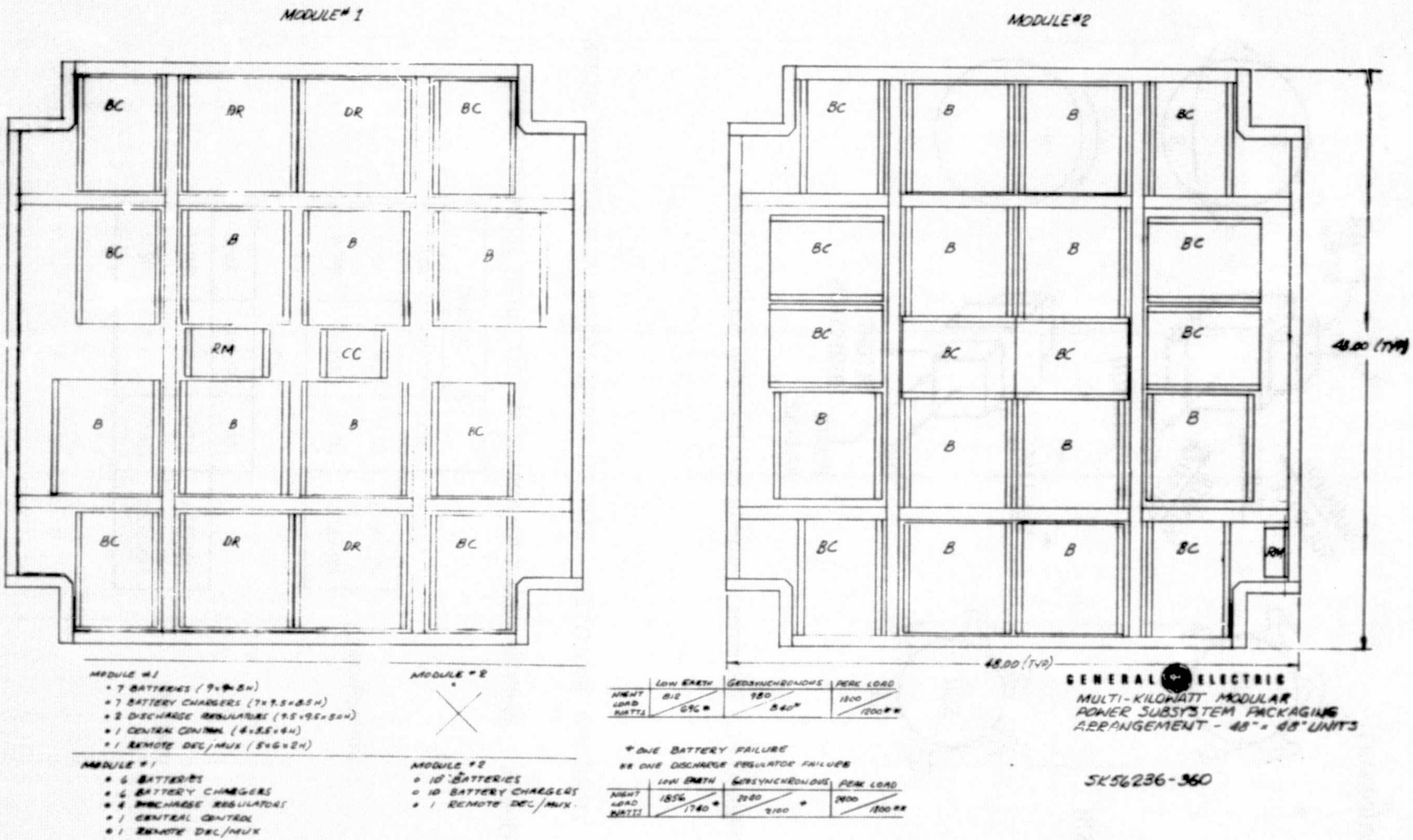


Figure 46. Packaging Arrangement
48 x 48 Inch Units

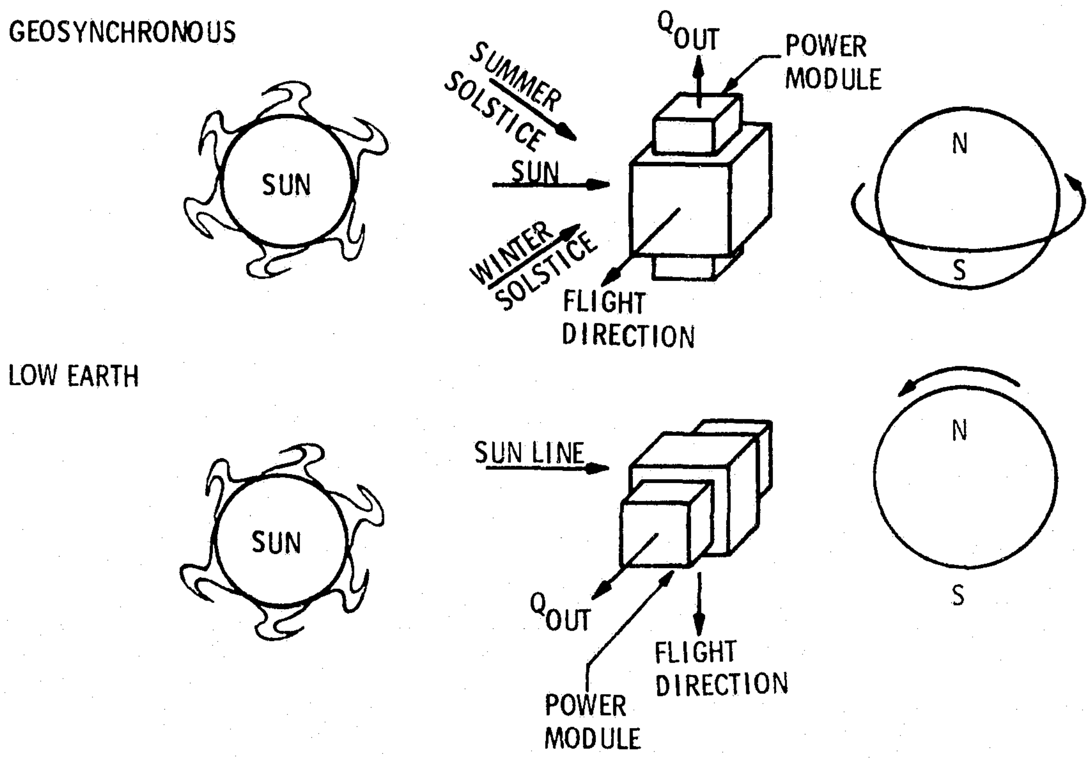


Figure 47. Thermal Analysis Sun Position Assumptions

TABLE 26. THERMAL ANALYSIS - POWER DISSIPATION PER MISSION MODE

GEOSYNCHRONOUS - MAINTAIN AVERAGE TEMPERATURE AT + 20°C DURING SUMMER SOLSTICE

			EQUINOX	
		DAY	NIGHT	AVG
GEOSYNCHRONOUS	MODULE 1	105 W	696 W	135 W
	MODULE 2	139 W	491 W	157 W

		DAY	NIGHT	AVG
LOW EARTH	MODULE 1	157 W	529 W	280 W
	MODULE 2	226 W	426 W	292 W

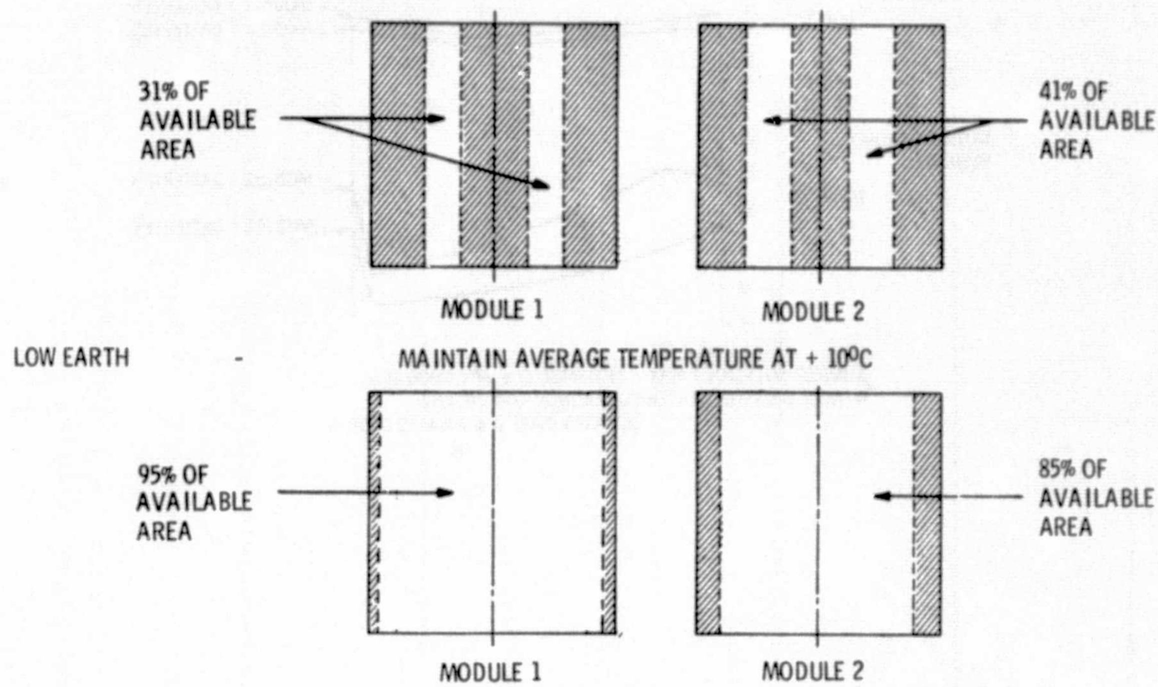


Figure 48. Thermal Analysis - Radiating Areas

A thermal transient analysis was then performed to determine the variation from end of day to end of night temperatures, and also the variation between batteries in Module No. 1 and No. 2. Figure 49 graphically shows for the low earth orbit, the day to night temperature variation is only 3°C and the difference between the two power modules is about 2°C which is very good for a first cut thermal design.

The thermal transition during the equinox period of the geosynchronous orbit is within the 0° to 20°C limit for the batteries, however, the difference between module temperatures must be reduced by a slight change in the radiating area of one of the modules.

As can be seen for both orbits, the module temperature increases during the night period due to battery discharge and discharge regulator operation. Module No. 1 of the geosynchronous case continues to heat up as the spacecraft comes into the day period. This is caused by the thermal capacitance of the four discharge regulators which had been operating at a high power level during the 72-minute eclipse. Heat continues to dump from these units for about 2 hours after they have been shut off.

These results show that the maximum amount of components can be put into the power module and thermal control can be maintained by a simple passive design using blankets and thermal coatings.

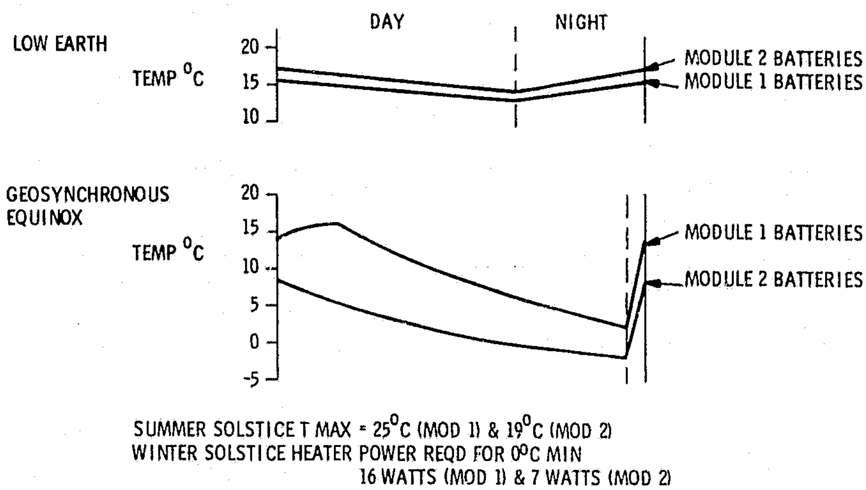


Figure 49. Battery Transient Temperatures

CONCLUSIONS OF THE STUDY

The review of requirements for free flying spacecraft in the next 15 years has shown that with the exception of a small percentage of broadcast missions, power requirements are below 2 kilo watts, with 65% of the spacecraft requiring less than 600 watts. Therefore, a modular power processing system need not be multi kilowatt, but growth to 2 kilowatts is needed.

Developing a modularized universal spacecraft power processing system will result in economic savings as non-recurring design and development costs are eliminated from each spacecraft system. Standardization of any design will be beneficial in this respect, but this study showed that a shunt regulated direct energy transfer power system design will provide the greatest advantages compared to other designs using unregulated power buses. The basic modular units of the power system are shown in Figure 50. The equipment consists of:

- A partial shunt regulated solar array that is modularized in 500 watt segments.
- A highly redundant central control that governs the operation of the power equipment to maintain bus regulation.
- A 16 cell nickel cadmium battery using 20 ampere-hour capacity standard cells.
- A dedicated battery charger that 'bucks' the regulated bus voltage down to the battery terminal voltage while limiting the maximum charge current to 5 amperes (c/4).
- A pulse width modulated boost discharge regulator with a 600 watt output capability that can operate in parallel with like units as required to satisfy eclipse load demands.

When selecting a power system design, and comparing it with others, all the power processing functions must be considered whether they are within the boundaries of the power system or are provided by the user loads. That is to say, when analyzing an unregulated bus system, the regulators located at the loads must be considered a power system item even though historically their penalty has been allocated to the load.

The design effort of this study has produced modular concepts at the circuit level, equipment level, and system level. Design rationale developed during the study and failure modes analysis has provided preliminary design specifications for the power processing equipment. These can be used for the next logical phase of development which is breadboard hardware. Thermal analysis has verified that the power module layouts can be thermally controlled by passive radiator designs.

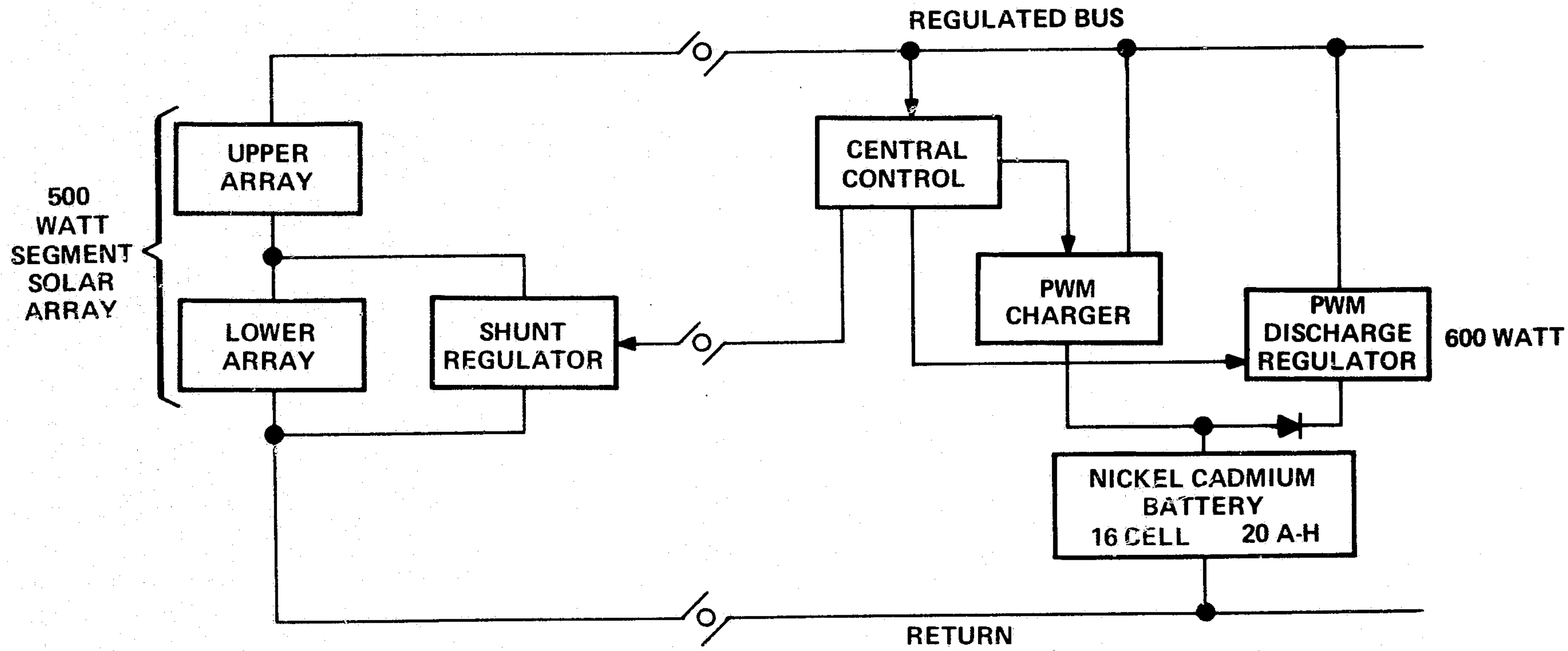


Figure 50. Simplified Shunt Regulated - Direct Energy Transfer Power System

APPENDIX I-1

LITERATURE REVIEW OF SPACE POWER SYSTEMS AND POWER CONDITIONING EQUIPMENT

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
1.	A Failure-Tolerant Sequencing Shunt Regulator R. E. Andrews, General Electric Co. Space Division 9th Intersociety Energy Conversion Engineering Conference: 1974 Paper #749136	Requirements, design and tests are described for an analog sequencing shunt regulator for a multi-hundred watt RTG. The approach used resulted in a significant piece-parts reduction with increased reliability, as compared to previous implementation. The improvement resulted from a change of philosophy in regard to the system response to component failure, yielding a more-direct, failure-tolerant concept.									X		

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
2.	Solar Array Cost Reductions D. T. Bernatowicz, NASA/Lewis 1972/1973	Historical Solar Array costs are presented for a broad cross section of flight projects over the past 10 years covering a range of average output from less than 100 watts to 4 KW.	X					X					X
3.	Analysis and Performance of Paralleling Circuits for Modular Inverter-Converter Systems Arthur G. Birchenough & Francis Gourash NASA/Lewis March 1972 NASA-TN-D-6713	Paralleling Circuits for any number of inverters (or converters) are analyzed and test results presented. Good correspondance between theoretical and actual performance is shown. Transient load sharing is not considered, but effects of output filter parameters is noted.			X								
4.	Solid State Power Controllers (Proceedings of the Space Shuttle Integrated Electronics Conference, Volume 2 Jack C. Boykin and William C. Stagg NASA-MSC and Donald E. Williams, NASA-MSFC 1971 N71-35039(NASA-TM-X-58063)	Advantages of solid state power controllers are discussed, especially as applied to distributed bus/remote controlled systems. The need for firm definition of requirements for these controllers is identified, so that qualified hardware can be developed.				X	X					X	

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
5.	Electrical Power Subsystem for the Synchronous Meteorological Satellite (SMS) D. C. Briggs & H. N. McKinney Philco-Ford Corporation 1972 729080(7th IECEC Conf.)	This paper describes a direct energy transfer system, with a regulated bus voltage of 29 volts. The battery charge method employed is one of command-selectable solar array elements in series with the main array, providing inherent current-limiting. A linear partial shunt voltage regulator is used for array stabilization, and a PWM boost regulator for battery output. System capability is 145 to 240 watts.	X	X				X		X	X		
6.	A Sequenced PWM Controlled Power Conditioning Unit for a Regulated Bus Satellite Power System A. Capel, D.M.O'Sullivan ESRO; PPESC-1972 Record IEEE AES Group May 1972	A 28V, 200 watt Direct Energy Transfer System is described, which is stated to be applicable to modular multi-kilowatt requirements. A single analog error signal is developed, which by means of naturally-exclusive threshold levels, controls the functions of array regulation, battery charge and battery discharge. Each function is implemented using synchronized PWM techniques. The array output is controlled by multiphase digital full shunts, (one shunt per string), resulting in higher ripple frequency than otherwise attainable. Each of the other battery related functions is also PWM-operated, minimizing variations in thermal dissipation over spacecraft life. Test data presented appears to show excessive bus noise, although filtering might reduce this to workable levels		X				X		X	X		

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
7.	An Advanced Concept in Electrical Power Distribution Control and Management Capt. J.R. Courter and Capt. T.R. Murrow U.S. Air Force IEEE Power Processing and Electronics Specialists Conference Record-1972 May 22, 1972	The application of a central digital computer is presented for the logical determination and control of loads distributed throughout a vehicle. Advantages highlighted are related to reduction of control (signal) wire routing, and elimination of individual feeders for each load. In their place are signal wires routed to remote computer input/output terminals, and their connection to the central computer accomplished by means of a PCM data bus. Individual loads are handled by local solid state power controllers, under computer/data bus commands. Signal logic is to be accomplished by software rather than hardware interconnections. However, no hardware is offered to accomplish these ends.				X	X			X		X	
8.	Optimized Solar-Array-Battery Space Power System R. A. Engelhardt, Martin Company, Baltimore, Maryland Oct. 27, 1965	Describes the virtues of maintaining the operation of solar arrays near the maximum power voltage point through the use of switching step down regulators. Also describes the classical solar array-battery "lockup" situation which is eliminated with this design.	X		X					X	X		
9.	Space Shuttle Electrical Power Distribution Considerations. J. L. Felch, Electrical Division, Astrionics Lab. NASA/MSFC N71-35042 1971	The existence of multiplexed control of distributed bus systems is acknowledged and identified as a candidate approach for space shuttle					X			X		X	X

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
10.	A Technique for Estimating Space Power System Costs J. G. Fish Aerospace Corporation for USAF-SAMSO Under contract: F04695-67-C-0158 October 1967 SAMSO #TR-67-112	Estimating methods are present for evaluating costs (development and procurement) of several classes of power systems. The techniques are said to be valid; they require expansion to cover alternate approaches, later technologies, and more current cost parameters.								X			X
11.	Past and Present Manned Spacecraft Electronics and Implications for the Space Shuttle R. A. Gardiner and G. Xenakis NASA-MSC (Presented to the 21st International Astronautical Congress, October 4-10, 1970 N71-10465 (NASA-TM-X-58054) November 1970	Power subsystems are reviewed from the Mercury through Apollo programs, and projections made for space shuttle load magnitudes. Distributed bus with remote control is recommended for shuttle era large spacecraft.				X	X	X	X	X			
12.	ERTS Reference Manual General Electric, Space Division	Description of ERTS Power Subsystem and hardware						X		X			

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
13.	Nimbus E and F Experiment Interface Requirements General Electric Co. for NASA, GSFC, Greenbelt, Md. NASA Doc. X-450-68-415 June 1972	An excellent example of how to coordinate the electrical interface between the power subsystem and user loads.				X		X	X	X			
14.	Review of Shuttle Systems Payload Data Activity Results General Electric Co., Space Division, Advanced NASA Programs Valley Forge, Pa. January 24, 1974	A publication of the viewgraphs that were presented to introduce Space Division users to the latest information on shuttle payloads; to place GE effort in perspective relative to total payload activity; and to stimulate GE use of Payload Data Bank for business projections, technical analyses, proposal background data, and IR&D ideas.					X		X				
15.	An Automatic Electrical Distribution System Dr. M. A. Geyer Westinghouse Electric Corporation Aerospace Electrical Div Lima, Ohio N71-35040 1971	Advantages of a distributed bus, multiplexed remote control distribution system are briefly presented. Use of a dedicated control computer is mentioned, but not expanded upon.				X	X			X		X	

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
16.	Power Conditioning Equipment for a Thermoelectric Outer Planet Spacecraft Power Subsystem; Quarterly Technical Report J. H. Hayden General Electric Co. Space Div. (For JPL, under contract #952536 & NASA7-100) 1J86-TOPS-480 (Gen'l Elect. Rpt) December 15, 1969	a. Effects on power subsystem weight and efficiency, from source to user loads, are analyzed as a function of converter frequency. With the source incremental output per unit of weight, and semiconductor switching losses available at the time, the optimum operating frequency was found to lie between 2 to 6 KHz. Additionally, AC(distributed converter) vs DC (distributed converter) vs. DC (single converter) power distribution systems were compared. On the basis of weight, reliability, EMC and overall efficiency, the distributed converter DC system was found to be most desirable. b. Development of a quad-redundant, analog-sequenced shunt regulator is also reported.						X	X	X	X		
17.	ATS Spacecraft System Configuration Study Hughes Aircraft Co., Space Systems Div. (for NASA GSFC, Greenbelt, Maryland under contract #NAS5-21123) N-71-10655 CR-111140 September 1970	Comparison of ATS A thru E power subsystem designs and other data to aid in optimization of regulation and conversion of spacecraft electrical power.						X		X			X
18.	Advanced Power Conditioning System N. L. Johnson Electro-Optical Systems, Inc. (for JPL, under contract #953097) with appendix by Dr. S.J. Lindena. November 11, 1971 N72-24321(NASA-CR-126641)	Test results are discussed (in comparison with requirements) for a breadboard converter. The converter employs triangular/trapezoidal current waveforms, and thereby allows inherent paralleling and load-sharing without auxiliary interconnections.			X								

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
19.	ATM CBRM Engineering and Development Report Dr. G. M. Jones, L. K. Jarrett & L. N. Mercer, Space Support Division, Sperry Rand Corporation Huntsville, Alabama for NASA, G.C.Marshall Space Flight Center Huntsville, Alabama under contract no. NAS8-21812 MSFC Rpt.No.40M26995 July 21, 1972	A description of the engineering and development of the Apollo Telescope Mount (ATM) charger, battery, regulator module (CBRM). This system is modular containing 18 separate modules. Selection of interest include 1.0 and 2.0 which describe system requirements, justifications, and reliability. Sections 3.0 and 5.0 describe the Charger and Regulator respectively. Section 10.0 describes design problems and failure modes experienced.		X				X		X	X		
20.	Integrated Electronics Solar Array Control Unit Stewart G. Kimble, TRW Systems & Joseph F. Wise AFAPL,Wright - Patterson AFB PCSC Record, 1971 (IEEE AES Group) April 1971	A control approach is described for a nominal 2KW unit, applicable however to units from 500 watts through 20 Kwatts, projected for use during the 1975-1980 period. The requirements include a goal of 7-10 year life, and the ability to withstand nuclear indiation. Of the nine system configurations analyzed, the Direct Energy Transfer method was selected based on power-to-weight ratio and use of components (count & type) which can be made radiation resistant. A hybrid partial-shunt regulator is used, with both linear & digital elements, providing a near-constant thermal dissipation through mission life. Not discussed are the transient response characterictics of the digital portion of the partial shunt regulator.					X			X	X		X

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
21.	Low Cost Modular Power Systems for Multi-Mission Earth Observatories A. Kirpich & J.Schumacher General Electric Space Systems, Phila., Pa.	A Direct Energy Transfer (DET) power systems was compared with the present Nimbus and ERTS hardware showing improved performance with resulting reduction in solar array and control functions which provides significant <u>cost savings</u>. In addition, the DET system eliminates certain power management functions which reduces the cost of ground operations' power management.					X	X		X			X
22.	Flight Performance of the ERTS-1 Spacecraft Power System A. Kirpich, H. Thierfelder M. Lamnin, & D. Wise Gen. Electric Co., Space Division PESC Record-1973 (IEEE AES Group) June 1973	Described is a centralized power system with approx. 500 watts capability using photovoltaic solar cells with nickle-cadmium batteries. Of interest is the power management technique employed which involves ground selection of "Science" and "Auxiliary" loads, orientation of solar arrays, and full-shunt regulation.						X	X	X			
23.	Nickel-Cadmium Battery Performance Prediction Models Apollo Telescope Mount Application W. R. Kirsch, Sperry Rand Corp.,Huntsville,Ala. IECEC739011 August 13, 1973	The paper describes the design, test results, and the development of the performance prediction model of the ATM Ni-Cd battery.		X				X					

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
24.	Electric Power Processing Distribution and Control for Advanced Aerospace Vehicles A. Krausz, TRW Systems Group & J.L. Felch, NASA-MSFC PPESC-1972 Record (IEEE AES Group) May 1972	An extensive review of power distribution systems, comparing weights and costs of several concepts. Conclusion is drawn that distribution at 100 VDC (or greater) as well certain AC voltages are required to meet future large-scale spacecraft requirements. (AC voltages would be required only for large motor/rotating source cases). The lack of space qualified high-voltage components is identified as the major stumbling block at the time of the report.				X	X		X	X		X	X
25.	Space Vehicle Electrical Power Processing Distribution and Control Study, Vols. 1 & 2 A. Krausz, TRW Systems Group & J.L. Felch, NASA-MSFC N72-33053/54 (NASA CR-123907/08 June 1972	An extensive review and projection of power processing & distribution systems, comparing weights and costs for several concepts, including different levels of AC & DC voltages. Distributed bus with remote control circuit breakers (solid state) is recommended, at a voltage level of 100 VDC, although present technology has not provided space-qualified components for this concept at required power levels. Projected shuttle-era loads are summarized, with static power sources anticipated. No source or processing methods are evaluated, since their effects on distribution are considered to be comparable, and only a small portion of the overall electrical system cost of ownership over program life.				X	X		X	X		X	X

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
28.	Battery Cell Control and Protection Circuits H. L. Layte, D. W. Zerbel, TRW Systems Group IEEE Power Processing and Electronics Specialists Conference Record-1972, page 106 May 22, 1972	Describes functions and circuits to protect battery cells from over-charge and over-discharge and improve battery performance for long life applications. Functional block diagrams, circuit schematics, and functional characteristics are presented. Techniques presented are currently being flown on Intelsat III, Prioneer 10 and the Apollo 15 & 16 Particle and Fields sub-satellites. Calculated cell string (24 cells) reliability in excess of 0.95 for seven years operation.		X				X					
29.	Study and Analysis of Satellite Power Systems Configurations for Maximum Utilization of Power (Phase II Technical Report) J. G. Leisenring & D. N. Stager TRW Systems Group for NASA-GSFC under contract #NAS5-9178 N69-19134(NASA-CR-100189) December 31, 1968	The EPSOM computer technique for power system optimization. It is claimed that the technique provides analytical evaluations of concepts, rather than reliance on experience and intuition. Also discussed is SRAP, which was projected to be a self-regulating and protection technique, employing pre-programmed responses to failures as a means of providing corrective actions. This method was deemed impractical because of the rapid reaction time required (to be effective) and the lack of forewarning detector technology.								X			X

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
26.	An Evaluation and Comparison of Power Systems for Long-Duration, Manned Space Vehicles John G. Krisilas & Harrison J. Killian Aerospace Corporation for AF Systems Command under contract # AF04(695)-1001 Air Force#SSD-TR-65-123 Aerospace#TR-1001(2730-01)-2 May 1967	Various solar and nuclear power sources are compared with fuel cells for use on 2 to 10 KW (average)spacecraft for up to 5 yrs. life. The authors conclude that radioisotope/dynamic systems are best suited for these applications, although no mention is made of their availability. Solar/Photovoltaic and Radioisotope/Thermoelectric systems are ranked lower for reasons of weight and integration difficulties.	X				X						X
27.	Asdtic Duty-Cycle Control for Power Converters V. R. Lalli & A. D. Schoenfeld IEEE Power Processing and Electronics Specialists Conference Record-1972, pg. 96 May 22, 1972	Describes reasons for beginning Analog Signal to Discrete Time Interval Converter (ASDTIC) development, the design goals sought in the development and design details. ASDTIC attempts to standardize for general purpose the "Sense" portion of the clasic switching regulator and at the same time include a novel second feedback loop. The second loop uses a signal which senses voltage as a function of inductor stored energy. This technique claims to approach superior regulator stability, regulation, transient response, and freedom from the effects of variations in parts characteristics. Paper based on work done under contract NAS12-2017.			X						X		

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
30.	Design and Performance of Intelsat IV Power Subsystem E. Levy, Jr. and Fred S. Osugi Hughes Aircraft Co & Comsat Corp. 729078 (7th IECEC Conf. 1972)	The power subsystem is described as having 2 Solar panel arrays/buses, each with 200 watts nominal capacity. Batteries are charged from separate charge arrays. The subsystem design has functioned nominally on three spacecraft, and incorporates commanded relay contacts to switch from normal operation (trickle charge, blocking diode discharge) to a recondition mode. Redundancy is provided through backup relays with contacts connected in "three-way switch" style. Two unregulated buses are used, with relay protection and paralleling.	X			X		X		X			
31.	Evaluation of Space Station Solar Array Technology & Recommended Advanced Development Programs Lockheed Missiles & Space Co.,Space Systems Div. Power Systems for Manned Spacecraft Center,Houston Texas. under contract #NAS9-11039 N71-16462(Accession No.) CR-114828 (NASA No.) 28 December 1970	1970 technology review including on-array electronics and power transfer devices. On-array electronics consists of conditioning solar cell power at the source, isolation diodes, and bypass diodes. Significant advantages are reduced vehicle heat load, and potential reliability increase. The big disadvantage is that only voltage limiting has had flight history where only two applications used zener diodes (passive). Other conditioning electronics has been conceptual only because of reluctance to operate electronic devices below -55^{o}C in the absense of performance data. Slip ring power transfer devices were reviewed listing seven major suppliers. The largest current device flown is Nimbus at 10 amperes per circuit continuous.	X				X	X			X		

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
32.	Photovoltaic Power Systems on Flight Spacecraft -Nimbus 2 K.F.Merten,K.L.Hanson, W.J.Schlotter, General Electric Co.,Space Div. (for NASA-OART, under contract #NAS7-547) NASACR-62045 (GE68SD4222) 23 February 1968)	The power subsystem of Nimbus 2 is described, having a payload power requirement of 186 watts, and a maximum (BOL) in excess of 400 watts available at the array. Of historical interest is the power management technique, which involves the command selection of functional loads and auxiliary loads (resistors) sufficient to load the bus and maintain a bus voltage of -24.5 $\pm$0.5 VDC. The selection is made each time a load change is required, when batteries are full-charged and cannot act as bus loads, at times of spacecraft night, etc.						X	X	X			
33.	A Solar Array and Battery Electrical Power Subsystem for the Shuttle-Launched Modular Space Station W. E. Murray, McDonnell Douglas Astronautics Co. 729073 (7th IECEC Conf. 1972)	Tradeoff alternatives and selections are listed to provide 23 to 31 KW to the space station during its growth period. 115 VDC, with sequential partial shunt regulation was chosen for the source. Batteries with low voltage charge, reconnected for high-voltage discharge are planned. 115 VDC transmission, with mixed 115 VDC/115-200 VAC distribution, employing central DC regulation are selected. Automatic remote control with manual backup is foreseen.	X	X			X			X			
34.	The 1973 NASA Payload Model NASA Mission & payload Planning Office, Program Development, Marshall SFC. October 1973	Schedule and description information which portrays the 1973 NAS Payload Model covering all NASA programs and the anticipated requirements of the user community, not including DOD, for the 1973 to 1991 period.					X		X				

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
35.	Electrical Power Subsystem Definition for Shuttle Launched Modular Space Station A.A. Nussberger, Space Division, North American Rockwell Corp. for NASA-MSC under contract NAS9-9953 729071 (7th IECEC Conf. 1972)	Description of implementation planned for space station electric power system. Included are power levels and certain constraints and requirements which lead to the particular 150 VDC configuration selected. Bus arrangement and malfunction work-arounds are described, with either the central computer control or separated control modes.				X	X			X			
36	High Performance High Reliability Switching Regulator Development. S. R. Peck, J.H.Hayden, etal, General Electric Co. Space Division GE No.N-22927 23 April 1973	Design techniques are described for a switching regulator for V_{in} = 200-400 Vdc, V_{out} = 56 Vdc $\pm$1%, efficiency = 90% min. The development of weighting factors for five different design approaches is presented against nine different parameters such as "relative size of input filter" and "relative design complexity".			X						X		
37	Space Vehicle Electrical Power Systems Study (Second Interim Technical Report, Project A-1251) S.L. Robinette, G.W. Bechtold & G.W.Spann (Georgia Inst. of Tech., Electronics Div., Atlanta Georgia) for NASA-MSFC under contract #NAS8-25192	A comparison of electrical system elements and their interrelationships. Power profiles for large-scale spacecraft are presented (to the extent available), and mixed distribution voltages are recommended, so as to permit more tolerant loads to be operated directly from various buses, without incurring efficiency penalties.											

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
38.	Space Vehicle Electrical Power Systems Study (Final Technical Rpt., Project A-1251) S.L.Robinette, G.W.Bechtold, and G.W.Spann (Georgia Inst.of Tech., Electronics Div.,Atlanta, Ga.) For NASA-MSFC under contract #NAS8-25192 N71-37781(NASA-CR-119961) Sept. 22, 1971	A comparison of advanced electrical power systems applicable to large-scale spacecraft in the Shuttle Era. Compatibility is recommended between Shuttle and related spacecraft power systems. Solid-State vs. electromechanical switching and power control is discussed, and the use of a dedicated control computer with a data bus is recommended to enhance maintainability, self-test, and to reduce power system installed weight in large spacecraft. An excellent bibliography is contained. Problems related to high-voltage DC systems (100 VDC) are identified.				X	X		X	X			
39.	Integrated Power Attitude Control System (IPACS) Mid-Term Briefing Summary Rockwell International Corporation, Space Division for NASA Langley under contract NAS1-11732 Rockwell NO.SD73-SA-0017 Jan. 17, 1973 com	Rockwell is performing a study to analyze the virtues of using spinning flywheels for both electrical energy storage and attitude control of spacecraft. The IPACS consists of a power generating source, energy/momentum wheels, motors and generators, power conditioning assemblies and computer assemblies necessary to provide for non-interacting power and attitude control functions. The presentation material defines the program plan and study logic, and the mission, competing subsystems, and IPACS requirements. In addition, it defines IPACS operational fundamentals, trade studies, and some system details. Although the study shows cost advantages for geosynchronous and high power low earth orbit missions, planetary and 30 day shuttle missions do not show IPACS benefit. In addition the issue of reliability and failure modes and effects was not treated in this report and should be carefully considered.					X		X	X			X

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
40.	Space Power Supply Study Richard V.Silverman Technology Division Navy Space Systems Activity NSSA-R40-68-5(AD672772) May 1968	This study compares relative merit of fuel cell, battery and RTG power systems, in several combinations and for orbital life spans up to one year. Average power levels of 300 and 600 watts are investigated, with peak power requirements to 5 KW for various durations.	X				X						X
41.	Space Electrical Power Systems for the Mid-1970 Richard V. Silverman Technology Division Navy Space Systems Activity NSSA-R40-69-4 September 1970	Power systems are compared on several parameters, including cost, size, weight and state of development for requirements of approx. 300W (average) and 2700 watts peak, with several duty-cycles. It is concluded that Nuclear Reactor/Thermo-electric systems offer the greatest merit, however, solar photovoltaic/battery systems are acknowledged as best for low duty-cycle applications.	X				X	X					X
41.	Space Shuttle Payload Descriptions, Vol.I & II (I: Automated Payloads; II:Sortie Payloads) Space Shuttle Payload Planning Working Groups & Nat'l Academy of Sciences. October 1973	Engineering descriptions of automated and sortie payloads. Power type and consumption data is listed by power type, average and peak power levels and energy needs. The automated missions include power requirements from Shuttle, Tug, and also describe spacecraft power system design capability.					X		X				

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
43.	Study and Analysis of Satellite Power Systems Configurations for Maximum Utilization of Power TRW Systems for NASA-GSFC under contract #NAS5-9178 NASA-CR-898 October 1967	This study compares power systems for post-1967 satellites with sub-kilowatt loads, various user voltages are listed and tradeoffs made between various distribution voltages and frequencies. It is concluded that for the class of satellites studied, a central inverter with AC distribution is desirable. Standardization of voltages is urged, which could lead to standardization of circuits and components, and later to their modularization.				X	X	X		X			
44.	System Design Considerations for A 25 KW Space Station Power System Gary Turner, Alan K. Johnson, and Martin G. Gandel Lockheed Missiles and Space Co., Inc. 729074(7th IECEC Conf. 1972)	Power system design is discussed, with photovoltaic source and regenerative fuel cells appearing to be the most likely energy storage method. Development activity in this area of storage is urged, enhanced by the development of 115 VDC switchgear to control the power distribution.	X			X	X			X		X	
45.	Logic-Controlled Solid-State Switchgear for 270 volt DC D. Waddington and E. Buchanan, Jr.,Martin-Marietta Corp. and G. Sundberg, NASA-Lewis PESC Record - 1973 (IEEE AES Group) June 1973	Development of an SCR-based 15 amp, 270 VDC remotely controlled circuit breaker is described. While not a prototype, it demonstrated the feasibility of such a device, an essential item for distribution systems in excess of 50 VDC. A novel commutation circuit is described, which does not employ transient current paths through the source nor load impedances. Also incorporated was an automatic reclosed function, with a selectable number of attempts, and a fixed reset time. It is considered that the particular design is most applicable to aircraft, and would require considerably more design activity to be suitable to spacecraft applications.										X	

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
46	A design for Thick Film Microcircuit dc-to-dc Converter Electronics H.M. Wick, Jr. and S. Capodici, General Electric Co.,Space Div. IEEE Power Conditioning Specialists Conference April 20, 1970	The design concept for a thick film microcircuit dc-to-dc converter electronics is presented. The techniques used reduce weight by 70 percent, volume by 80 percent, and interconnections by 75 percent. The close piece-part spacing allowed short interconnections, lower dissipation, and reduce noise coupling. The developed microcircuit handled total power levels from one watt to 25 watts.			X			X			X		X
47.	Decentralized Power Processing for Large-Scale Systems James W. Williams, Hughes Aircraft Co. PESC Record-1973 (IEEE AES Group) June 1973	Relative merit of centralized, decentralized and combination processing systems are discussed. Hughes contention is that the combination approach, with a central pre-regulator with 100 VDC output, and local converter/regulators for individual users is most desirable based on: Regulation, fault isolation, reliability, efficiency, cost and weight. Hybrid regulators are described as having high reliability than similar discrete units, thereby offsetting an apparently higher parts count.					X	X		X	X		

REF. NO.	SOURCE IDENTIFICATION	SUMMARY	ARRAYS/SOURCES	BATTERIES/CHARGERS	CONVERTERS	DISTRIBUTION	FUTURE PREDICTIONS	HISTORICAL DATA	LOADS	POWER SYSTEMS	REGULATORS	SWITCH GEAR	WEIGHT/COST
48.	The Application of Standardized Control and Interface Circuits to Three DC-to-DC Power Converters. Yuan Yu, John J. Biess, Arthur D. Schoenfeld TRW Systems and Vincent R. Lalli, NASA-Lewis PESC Record - 1973 (IEEE AES Group) June 1973	Application of the two-feedback loop ASDTIC concept is described for several low-power converters. Common Analog processors and Digital processors were used, with comparible good results in each case. Proven was the basic concept that a standardized control concept can be implemented for various types of power converters, without specialized design.			X						X		

APPENDIX I-2

POWER REQUIREMENTS OF AUTOMATED SPACECRAFT PER NASA MISSION MODEL - SHUTTLE SYSTEM PAYLOADS DATA STUDY

PAYLOAD CODE NUMBERS

ASTRONOMY

AS-01-A - Large Space Telescope
AS-02-A - Extra Coronal Lyman Alpha Explorer
AS-03-A - Cosmic Background Explorer
AS-05-A - Advanced Radio Explorer

HIGH ENERGY ASTROPHYSICS

HE-01-A - Large X-Ray Telescope Facility
HE-03-A - Extended X-Ray Survey
HE-07-A - Small High Energy Observatory
HE-08-A - Large High Energy Observatory A
HE-09-A - Large High Energy Observatory B
HE-11-A - Large High Energy Observatory D

SOLAR PHYSICS

SO-03-A - Solar Maximum Satellite

ATMOSPHERIC AND SPACE PHYSICS

AP-01-A - Upper Atmosphere Explorer
AP-02-A - Medium Altitude Explorer
AP-03-A - High Altitude Explorer
AP-04-A - Gravity and Relativity Satellite - LEO
AP-05-A - Environmental Perturbation Satellite - Mission A

EARTH OBSERVATIONS

EO-07-A - Synchronous Meteorological Satellite
EO-08-A - Earth Observations Satellite
EO-09-A - Synchronous Earth Observation Satellite
EO-10-A - Special Purpose Earth Observation Satellite
EO-12-A - TIROS 'O'
EO-56-A - Environmental Monitoring Satellite
EO-57-A - Foreign Synchronous Meteorological Satellite
EO-58-A - Geosynchronous Operational Environmental Satellite
EO-59-A - Geosynchronous Earth Resources Satellite
EO-61-A - Earth Resources Satellite (Low Orbit)

EARTH AND OCEAN PHYSICS

OP-01-A - GEOPAUSE
OP-02-A - Gravity Gradiometer
OP-03-A - Mini-LAGEOS
OP-04-A - GRAVSAT
OP-05-A - Vector Magnetometer Satellite
OP-06-A - Magnetic Field Monitor Satellite
OP-07-A - SEASAT - B

LIFE SCIENCES

LS-02-A - Biomedical Experiment Scientific Satellite

SPACE TECHNOLOGY

ST-01-A - Long Duration Exposure Facility

PLANETARY

PL-01-A - Mars Surface Sample Return
PL-03-A - Pioneer Venus Multiprobe
PL-07-A - Venus Radar Mapper
PL-11-A - Pioneer Saturn/Uranus Flyby
PL-12-A - Mariner Jupiter Orbiter
PL-13-A - Pioneer Jupiter Probe
PL-18-A - Encke Rendezvous
PL-21-A - Encke Slow Flyby
PL-22-A - Pioneer Saturn Probe

COMMUNICATIONS/NAVIGATION

CN-51-A - INTELSAT
CN-52-A - U. S. DOMSAT 'A'
CN-53-A - U. S. DOMSAT 'B'
CN-54-A - Diaster Warning Satellite
CN-55-A - Traffic Management Satellite
CN-56-A - Foreign Communications Satellite
CN-58-A - U. S. DOMSAT 'C'

LUNAR

LU-01-A - Lunar Orbiter

Payload Number	Maximum Shuttle Power (Watts)	Spacecraft Power System					No. of New Buy Satellites	
		Type	Power Form & Quality	Maximum Power (Watts)	Maximum Energy Storage (Watt-Hrs)	Notes		
AS-01-A	2200	Array Bat	28vdc ± 2%	2120	5400		1	2K -2.2K
AS-02-A	350	" "	" " "	150	15 Cell, 12 AH	$3.6\ ft^2$SA	6	0 -.2K
AS-03-A	350	" "	" " "	150	" "	$3.6\ ft^2$SA	7	0 -.2K
AS-05-A	610	" "	" " "	150	" "	$3.6\ ft^2$SA	8	0 -.2K
HE-01-A	1540	" "	28vdc ±0.02%	1800	4800		1	1.6 -1.8
HE-03-A	1311	" "	" " "	1800	4800		1	1.6 -1.8
HE-07-A	650	" "	28vdc ± 2%	150	15 Cell, 12 AH	$3.6ft^2$ SA	4	0 -.2
HE-08-A	900	" "	28 vdc	1800	4800		1	1.6 -1.8
HE-09-A	844	" "	" "	1800	4800		1	1.6 -1.8
H-11-A	1540	" "	28vdc ±0.02%	1800	4800		1	1.6 -1.8
SO-03-A	1030	" "	28 vdc	250			2	.2 -.4
AP-01-A	1100	" "	28 vdc	200	40 AH		4	0 -.2
AP-02-A	1100	" "	" "	200			4	0 -.2
AP-03-A	1300	" "	" "	300			6	.2 -.4
AP-04-A	1220	" "	" "	500			2	.4 -.5
AP-05-A	6000	" "	" "	6000			2	5.8 -6
EO-07-A	450	" "	" "	900	2@ 40 AH		1	.8 -1
EO-08-A	1000	" "	" "	1200	20 AH		13	1 -1.2
EO-09-A	410	" "	" "	557			9	.4 -.6
EO-10-A	160	" "	" "	364	One 6AH		16	.2 -.4
EO-12-A	1000	" "	" "	1600	20 AH		2	1.4 -1.6
EO-56-A	1000	" "	28vdc ± 7V	680	20 AH		9	.6 -.8
EO-57-A	109	" "	28 vdc	170	20 Cell, 3AH		6	0 -.2
EO-58-A	109	" "	" "	170	20 Cell, 3AH		9	0 -.2
EO-61-A	450	" "	24.5 vdc	550			13	.4 -.6
OP-01-A	800	" "	28 vdc	500			2	.4 -.6
OP-02-A	485	" "	" "	200			1	0 -.2
OP-03-A	0	None	NA	0			12	——
OP-04-A	1000	Array Bat	28 vdc	350			2	.2 -.4
OP-05-A	430	" "	" "	100			9	0 -.2
OP-06-A	430	" "	" "	100			3	0 -.2
OP-07-A	1145	" "	" "	650			1	.6 -.8

Payload Number	Maximum Shuttle Power (Watts)	Spacecraft Power System: Type	Power Form & Quality	Maximum Power (Watts)	Maximum Energy Storage (Watt-Hrs)	Notes	No. of New Buy Satellites	
LS-02-A	400	Array Bat	27.5±2.5vdc	TBD			3	—
ST-01-A	0	None	NA	0			1	—
PL-01-A	13000 115 Vac	Array		250		Orbiter	2	.2 -.4
PL-01-A		Rtg.		70		Lander	2	0 -.2
PL-01-A		Bat.		70		Rover	2	0 -.2
PL-03-A	13000 @ 115Vac	Array		180			5	0 -.2
PL-07-A		Rtg's (3)		400			3	.2 -.4
PL-11-A	" " "	Rtg's (2)		140		Bus	2	0 -.2
PL-11-A		Bat's		50		Probe	2	0 -.2
PL-12-A	" " "	Rtg's (3)		400			3	.2 -.4
PL-13-A		Rtg's (2)		140		Bus	2	0 -.2
PL-13-A	" " "	Bat's		50		Probe	2	0 -.2
PL-18-A	" " "	Arrays		200			3	0 -.2
PL-21-A	" " "	Arrays		200			1	0 -.2
PL-22-A		Rtg (2)		140		Bus	2	0 -.2
PL-22-A	" " "	Bat's		50		Probe	2	0 -.2
CN-51-A	28vdc @ /5	Array Bat	82 ± 11vdc	4400	4 Batteries	Full Up Eclipse	21	4.2 -4.4
CN-52-A	28vdc @ 68	Array Bat	27-43 vdc	380	2@8 AH		7	.2 -.4
CN-53-A	75 @ 28 vdc	" "	82 ± 11vdc	4400	4 Batteries		14	4.2 -4.4
CN-54-A	83 @ 28 vdc	Hv Solar Array	400V	5000			4	4.8 -5.0
CN-55-A	90 @ 28 vdc	Array Bat	28 vdc	1320			11	1.2 -1.4
CN-56-A	84 @ 28 vdc	Det Array Bat.	28 vdc	550			11	.4 -.6
CN-58-A	60 @ 28 vdc	Solar Array Bat	28.1 vdc	466	2 Batteries		6	.4 -.6
LU-01-A	13000	Array Bat					2	—

Power Range (K-W)	Number of Different Systems	Total Number of New Buy Satellites	Percent of Total Number of Systems Satisfied	Accumulated Number of Systems	$\frac{\text{Col "B" Sums}}{\text{Col "B" Total}} \times 100\%$ @ $\leq$220 watts(191 Sys.)
0-200	14	70	34.1%	14	36.5
201-400	6	35	48.7	20	55.0
401-600	6	43	63.3	26	77.0
601-800	2	10	68.1	28	84.0
801-1000	1	1	70.7	29	86.0
1001-1200	1	13	73	30	89.5
1201-1400	1	11	75.4	31	96.0
1401-1600	1	2	77.9	32	96.5
1601-1800	5	5	90.2	37	99.0
1801-2000					99.0
2001-2200	1	1	92.6	38	100
4201-4400	2	35	97.4	40	
5801-6000	1	2	100	41	
	41	228			
	Col "A"	Col "B"	Col "C"	Col "D"	

Data $\frac{\text{Col "A" Sums}}{\text{Col "A" Total}} \times 100\%$

APPENDIX II-1

SOLAR ARRAY POWER REQUIREMENTS

APPENDIX II-1

SOLAR ARRAY POWER REQUIREMENTS

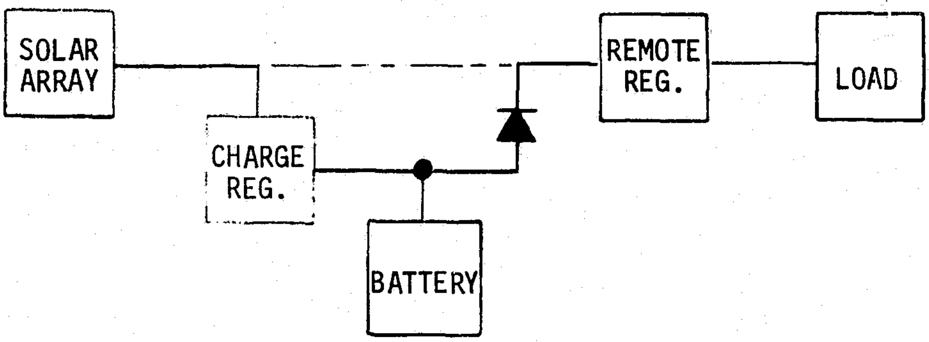
Low Earth Battery Discharge Effects on Solar Array Size:

Ground Rules

104 minute orbit
35 minute night
69 minute day
400 w day and night power level

BATTERY OPERATION

Remote Regulated:



Charge Regulator Eff	=	90%
Discharge Regulator Eff	=	96%
Remote Regulator Eff	=	90%
Distribution from Bat to Load	=	98%
Distribution from Array to Bat.	=	98%
Array Isolation Diode	=	97%

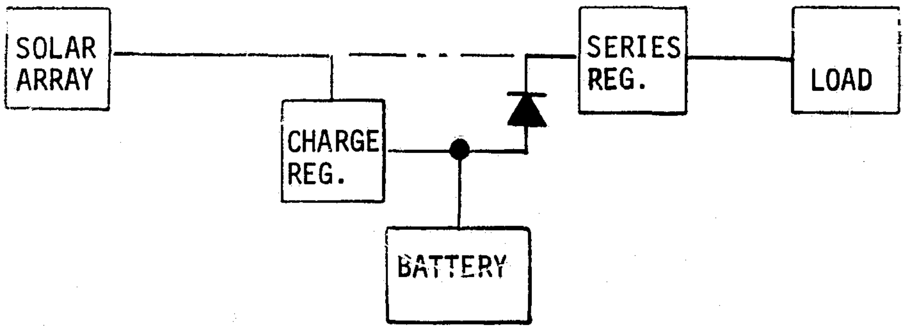
$$\text{Power from Battery} = \frac{400}{(.90)(.96)(.98)} = 472\ \text{W}$$

$$\text{Battery Watt Hrs.} = \frac{(472\ \text{W})(35\ \text{m})}{(60\text{m})} = 275\ \text{W-H}$$

$$\text{Energy Returned to Battery} = \frac{(275\ \text{W-H})(1.4\ \text{C/D})}{(.97)(.90)(.98)} = 451\ \text{W-H}$$

$$\text{Power from Array} = \frac{(451\ \text{W-H})(60\ \text{M/H})}{69\ \text{M}} = 392\ \text{W}$$

- Series Regulated:



Charge Regulator Eff.	= 90%
Discharge Regulator Eff.	= 96%
Series Regulator Eff.	= 85%
Dist. from Bat to Load	= 98%
Dist. from Array to Bat	= 98%
Array Isolation Diode	= 97%
Battery Eff.	= 1/1.4

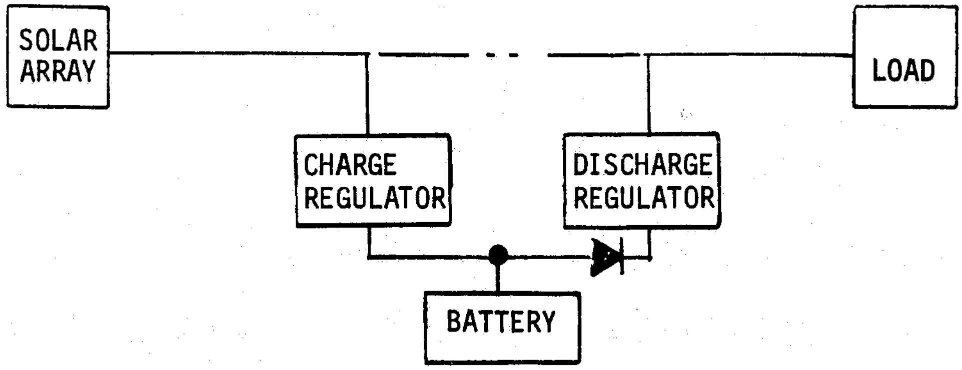
$$\text{Power from Battery} = \frac{400W}{(.98)\ (.85)\ (.96)} = 500\ W$$

$$\text{Battery Watt Hrs.} = \frac{500\ (35m)}{60M/H} = 292\ WH$$

$$\text{Energy Returned to Battery} = \frac{(292\ WH)\ (1.4\ C/D)}{(.90)\ (.98)\ (.97)} = 477\ WH$$

$$\text{Power from Array} = \frac{(477\ WH)\ (60\ M/H}{69\ M} = 415\ W$$

- Shunt Regulated



Charge Regulator Eff. = 90%
Discharge Regulator Eff. = 85%
Battery Diode Eff. = 96%
Dist from Bat to Load = 98%
Dist from Array to Bat = 98%
Array Isolation Diode = 97%

Power from Battery $= \dfrac{400\ \mathrm{W}}{(.98)\ (.85)\ (.96)} = 500\ \mathrm{W}$

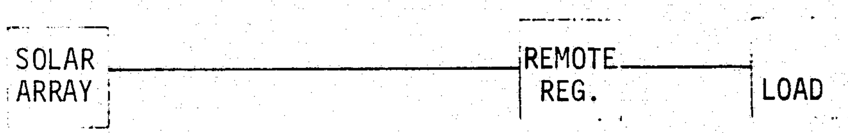
Battery Watt Hrs. $= \dfrac{(500\ \mathrm{W})\ (35\ \mathrm{M})}{60\ \mathrm{M/H}} = 292\ \mathrm{WH}$

Energy Returned to Battery $= \dfrac{(292\ \mathrm{WH})\ (1.4\ )}{(.90)\ (.98)\ (.97)} = 477\ \mathrm{WH}$

Power from Array $= \dfrac{(477\ \mathrm{WH})\ (60\ \mathrm{M/H}}{69\mathrm{M}} = 415\ \mathrm{W}$

Solar Array Operation

Remote



Remote Regulator Eff. = 90%
Dist from Array to Load = 96%
Array Diode Eff. = 97%

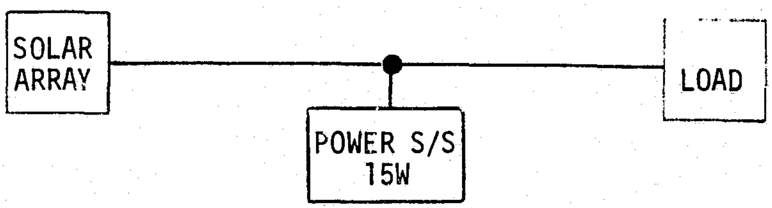
Power from Array $= \dfrac{400\ \mathrm{W}}{(.9\ )\ (.96)\ (.97)} = 477\ \mathrm{W}$

Series

Series Regulator Eff.	= 85%
Dist from Array to Load	= 96%
Array Diode Eff.	= 97%

$$\text{Power from Array} = \frac{400\ \text{W}}{(.85)\ (.96)\ (.97)} = 505\ \text{W}$$

Shunt



Dist from Array to Load	= 96%
Array Diode Eff.	= 97%

$$\text{Power from Array} = \frac{400\ \text{W} + 15\ \text{W}}{(.96)\ (.97)} = 446\ \text{W}$$

Total Array Power	Low Earth
Remote	392 + 477 = 869 Watts
Series	415 + 505 = 920 Watts
Shunt	415 + 446 = 861 Watts

Total Array Power - Synchronous	
Remote	520 Watts
Series	550 Watts
Shunt	486 Watts

Degradation Factors

12% Degradation/Year Low-Earth
6% Degradation/Year Synchronous

Array Power BOM

	Remote	Series	Shunt
Low-Earth	1671	1769	1656
Synchronous	645	682	603

For EOM Powers of (at 4 years)

Low-Earth	869	920	861
Synchronous	520	550	486

APPENDIX II-2

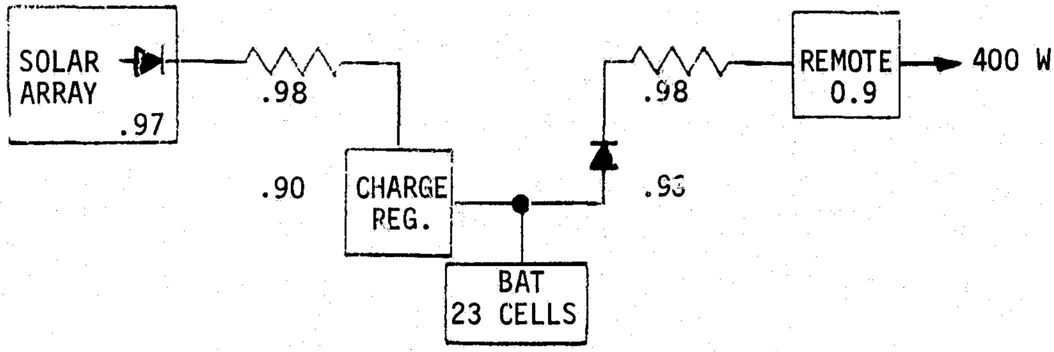
BATTERIES FOR LOW EARTH ORBIT

APPENDIX II-2

BATTERIES FOR

LOW EARTH ORBIT

Remote Regulated



Battery Discharge Watts

$$\frac{400}{(.9)\ (.98)\ (.96)} = 472 \text{ Watts}$$

Battery Discharge Amps

$$\frac{472 \text{ W}}{27.6 \text{ V}} = 17.1 \text{ Amps}$$

Discharge Amp Hrs.

$$\frac{17.1 \text{ A} \times 35 \text{ M}}{60 \text{ M/H}} = 9.98 \text{ AH}$$

$$\text{DoD} = \frac{9.98}{50} = 20\%$$

$$\text{Discharge Rate} = \frac{50 \text{ AH}}{17.1 \text{ A}} = 2.94$$

$$= C/2.94$$

Series Regulated

Battery Discharge Watts

$$\frac{400}{(.85)\ (.98)\ (.96)} = 500 \text{ Watts}$$

Battery Discharge Amps

$$\frac{500}{27.6 \text{ V}} = 18.1 \text{ Amp}$$

Discharge Amp Hrs

$$\frac{18.1 \times 35}{60} = 10.6 \text{ AH}$$

$$\text{DoD} = \frac{10.6}{50} = 21\%$$

Shunt Regulated

Battery Discharge Watts

$$\frac{400}{(.98)\ (.85)\ (.96)} = 500 \text{ Watts}$$

Battery Discharge Amps

$$\frac{500 \text{ Watts}}{(1.2\text{V/C})\ (16 \text{ Cells})} = 26 \text{ Amps}$$

$$\text{Discharge AH} = \frac{26 \text{ A} \times 35 \text{ M}}{60 \text{ M/HR}} = 15.2 \text{ AH}$$

$$\text{DoD} = \frac{15.2 \text{ AH}}{50 \text{ AH}} = 30.4\%$$

Battery Weight

Wt/Cell = 4.5 Lb

$$(23 \times 4.5)\ (1.4) = 145 \text{ Lb}$$
$$(16 \times 4.5) \times 1.4 = 101 \text{ Lb}$$

Remote Regulated

Battery Discharge Watts

$$\frac{400}{(.9)\,(.98)\,(.96)} = 472 \text{ W}$$

Battery Discharge Amps

$$\frac{472 \text{ W}}{27.6 \text{ V}} = 17.1 \text{ Amps}$$

Discharge Amp Hrs.

$$\frac{(17.1 \text{ A})\,(72 \text{ M})}{60 \text{ M/H}} = 20.5 \text{ AH}$$

DoD

$$\frac{20.5}{50} = 41\%$$

Discharge Rate

$$\frac{50 \text{ AH}}{17.1 \text{ A}} = 2.9$$

C/3

Charge AH

$$1.2\,(20.5) = 24.6 \text{ AH}$$

Charge Current

$$\frac{24.6 \text{ AH}}{22.8 \text{ H}} = 1.07 \text{ Amps}$$

Rate

$$\frac{50}{1.07} = 46.7$$

C/46.7

Series Regulated

Battery Discharge Watts

$$\frac{400}{(.85)\ (.98)\ (.96)} = 500 \text{ W}$$

Discharge Amps $= \dfrac{500}{27.6\text{V}} = 18.1$ A

Discharge AH $= \dfrac{(18.1)\ (72)}{60} = 21.7$ AH

DoD $= \dfrac{21.7}{50} = 43\%$

Discharge Rate

C/3

Charge AH

$$(1.2)\ (21.7 \text{ AH}) = 26 \text{ AH}$$

Charge Current

$$\frac{26 \text{ AH}}{22.8 \text{ H}} = 1.14 \text{ Amps}$$

Rate

$$\frac{50}{1.14} = 44$$

C/44

Shunt Regulated

Battery Discharge Watts

500

Discharge Amps

$$\frac{500}{(1.2 \text{ V/C})\ (16)} = 26 \text{ Amps}$$

Discharge AH

$$\frac{(26\text{A})\ (72)}{60} = 31.2 \text{ AH}$$

DoD

$$\frac{31.2}{50} = 62\%$$

Charge AH

$$(1.2)\ (312) = 37.4\ \text{AH}$$

Charge Current

$$\frac{37.4\ \text{AH}}{22.8\ \text{H}} = 1.64\ \text{Amps}$$

Rate

$$\frac{50}{1.64} = 30.5$$

C/30.5

		Discharge of 1 Battery						Charge Battery				
	No. of Cells	Watts	Amps	Rate	A-H	DoD	Cell Capacity	A-H	Amps	Rate	Weight of 2 Batteries (4.5 Lb/Cell)	Cost of 2 Batteries $600/Cell
Remote	23	472	17.1	C/3	9.98	20%	50 AH	11.98	10.4	C/4.8	290 Lb	55K
Series	23	500	18.1	C/3	10.6	21%	50 AH	12.7	11	C/4.5	290 Lb	55K
Shunt	16	500	26	C/1.9	15.2	30%	50 AH	18.2	15.9	C/3.2	202 Lb	38K

BATTERY CHARACTERISTICS

LOW EARTH ORBIT

			Discharge 1 Battery					Charge 1 Battery			
	No. of Cells	Watts	Amps	Rate	A-H	DoD	Cell Capacity	A-H	Amps	Rate	
Remote	23	472	17.1	C/3	20.5	41%	50	24.6	1.07	C/47	Same Weight & Cost As Low Earth
Series	23	500	18.1	C/3	21.7	43%	50	26	1.14	C/44	
Shunt	16	500	26	C/1.9	31.2	62%	50	37.4	1.69	C/30	

BATTERY CHARACTERISTICS

GEOSYNCHRONOUS

APPENDIX II-3

RELIABILITY CALCULATIONS

APPENDIX II-3

RELIABILITY CALCULATIONS

	Failure Rates (X 10^{-6})
IC's	0.05
SIGNAL XSTOR	0.003
POWER XSTOR	0.01
DIODES	0.002
RESISTORS	0.002
CAPACITORS	0.002
MAGNETICS	0.009 PER WINDING
RELAYS	0.20
FUSES	0.10

SHUNT REGULATOR RELIABILITY

For 4 Years (3.504×10^4 Hrs)

Reliability of Voltage Sense/Majority Vote Circuitry

$$R = R_S^3 + 3R_S^2 Q_S + 3R_S P_L P_H$$

$Q_S = P_L + P_H$
$t = 3.504 \times 10^4$ (4 yrs)

λ Per Stage

	Op Amp	Signal Transistor	Resistor Metal Film	Diode	Zener	
λ L	.5	1	1		1	0.25 + .003 + .002 +.003 = $.031 \times 10^{-6}$
λ H	.5	1	5	2	1	.025 + .003 + .101 + .004 + .003 = .095

$\lambda L = .031 \times 10^{-6}$
$\lambda H = .045 \times 10^{-6}$
$\lambda T = .076 \times 10^{-6}$

$$PL = \frac{.031}{.076}\left[1 - e^{-(.076 \times 10^{-6})(t)}\right.$$

$PL = .00108$
$PH = .00157$
$Q_S = .00265$

$R_S = .99735$

$R_S^3 = .9920$

$3R_S^2 = 2.984$

$3R_S^2 Q = .0079$

$3R_S \; P_L \; P_H = .000005$

$\underline{R_{MV} = 0.99990}$

Reliability of Power Amp

Electronics

	Pwr Xstor .01	Resistor Film .002		
λ_S	1 50%	0		.005
λ_0	1 50%	1		.007

$\lambda = .012 \times 10^{-6}$

$P_0 = \frac{.007}{.012} \left[1 - e^{-.012 \times 10^{-6} (t)} \right]$

$Po = 2.45 \times 10^{-4}$

$Ps = 1.75 \times 10^{-4}$

$R_Q = 1 - (2Po^2 - Po^4 + 4Ps^2 - 4Ps^3 + Ps^4)$

$12.005 \times 10^{-8} - 36.03 \times 10^{-16} + 12.25 \times 10^{-8}$

$-21.44 \times 10^{-12} + 9.38 \times 10^{-16}$

$R_Q = 1 - .000000242$

$\underline{R_Q = .9999997}$

Resistor of Power Amp

$\lambda = 6 \times .03 = .18 \times 10^{-6}$

$R_R = e^{-\lambda t} = .9937$

$R_T = R_R \quad R_Q$

Power Amp = .9937

Shunt Element #1

	Pwr Xstor .01	Resistor Film .002	Diode .002	
T_0	.5 50%	1	1	$.005+.002+.002=.009\times10^{-6}$
T_S	.5 50%			$.005\times10^{-6}$

$\lambda = .014\times10^{-6}$

$$P_0 = \frac{.009}{.014}(1-e - \lambda t)$$

R=.99951

$P_0 = 3.15 \times 10^{-4}$

$P_S = 1.75 \times 10^{-4}$

$\text{Relec} = R^2 + 2RP_S$

$= .9990 + .00035$

Relec = .99935

$\lambda_{RES} = 5 \times .03 = 0.15 \times 10^{-6}$

$\underline{R_{RES} = .99475}$

$\underline{R_{ELEMENT} = .9941}$

Sequencing Diodes

$1+2+3+2+2.5 = 10.5$ diodes

$\lambda_D = 10.5 \times .002 = .021$

$\underline{R\ _{DIODES} = .99926}$

6 of 7 shunt elements situation

$R = R_E^7 + 7\ R_E^6 Q_E$

$.9594 + 7\ (.9651)\ (.0059)$

$.9594 + .0399$

$\underline{R_{TE} = .99926}$

$R_{SHUNT} = (R_{MV})\ (R_{PA})\ (R_{DIODES})\ (R_{TE})$

$= (.99990)\ (.9937)\ (.99926)\ (.99926)$

$\boxed{R_{SHUNT} = .9921 \quad \text{for 4 years}}$

	IC (.05)	Signal Xstor (.003)	Pwr Xstor (.01)	Diode (.002)	Resistor (.002)	Capacitor (.002)	Magnetics (.009 Per winding)	Relay (.2)	Fuse (.10)	Total $\lambda \times 10^{-6}$
	(3)	(17)	(4)	(25)	(32)	(14)	(6)	--	--	
Load Preregulator	.15	.051	.04	.05	.064	.028	.054	--	--	.437
	--	(13)	(6)	(41)	(60)	(42)	(28)			
Series Reg.		.039	.06	.082	.120	.084	.252	--	--	.637
	(4)	(21)	(2)	(25)	(91)	(19)		(2)		
Failure Det	.20	.063	.02	.050	.182	.038	--	.40	--	.953
	(1)	(29)	(3)	(18)	(95)	(10)		(1)		
Charge Regulator	.05	.087	.03	.036	.190	.02	--	.02	--	0.613
	(2)	(21)	(6)	(23)	(28)	(18)	(21)			
Discharge Regulator	.10	.063	.06	.046	.056	.036	.189	--	--	0.550
Shunt Regulator		Special	Case							
		(5)	(8)	(11)	(60)	(4)	--	--	--	
Array Limiter	--	.015	.08	.022	.120	.008	--	--	--	0.245
Power Dist Unit	--	--	--	--	--	--	--	--	--	--

SUMMARY OF COMPONENT

RELIABILITY FOR 4 YEAR MISSION

Load Preregulator Reliability	= .9848
Series Regulator	= .9779
Standby Series Regulator	= .9978
Charge Regulator	= .9787
Discharge Regulator	= .9809
Standby Discharge Regulator	= .9981
Array Limiter	= .9914
Shunt Regulator	= .9921
Failure Detector	= .9671

UNREGULATED SYSTEM RELIABILITY

R_L = .9914

Rcr = .9787

R_{PREREG} = .9848

$(R_L)\ (Rcr^2 + {}^2Rcr\ Qcr)\ (R_{TTC}{}^2 + {}^2R_{TTC}\ Q_{TTC})\ (Rcr^2 + {}^2Rcr\ Qcr)\ (Rp^3 + 3Rp^2Qp)$

(.9914) (.9579 + .0417) (.9698 + .0299) (.9698 + .0299) (.9551 + .0442)

(.9914) (.9996) (.9997) (.9997) (.9993)

$\boxed{R_{UNREGULATED} = .9897}$

SERIES REGULATED SYSTEM

R_L = .9914

Rcr = .9787

R_{SR} = .9779

R_{FD} = .9671

$R_{STBY\ SR}$ = .9978

$(R_L)\ (Rcr^2 + 2Rcr\ Qcr)\ (R_{SR} + Q_{SR}\ R_{FD}\ R_{STBY\ SR})$

(.9914) (.9996) [.9779 + (.0221) (.9671) (.9978)]

(.9914) (.9996) (.9779 + .0213)

(.9914) (.9996) (.9992)

$\boxed{R_{SERIES\ REG} = 0.9902}$

SHUNT REGULATED SYSTEM RELIABILITY

$R_{SHUNT} = .9921$

$Rcr = .9787$

$R_{DR} = .9809$

$R_{STBY\ DR} = .9981$

$R_{FD} = .9671$

$$(R_{SHUNT})\ (Rcr^2 + 2Rcr\ Qcr)\ (R_{DR} + Q_{DR}\ R_{FD}\ R_{STBY\ DR})$$

$$(.9921)\ (.9996)\ \left[.9809 + (.0191)\ (.9671)\ (.9981)\right]$$

$$(.9809 + .0184)$$

$$(.9921)\ (.9996)\ (.9993)$$

$$\boxed{R_{SHUNT\ REGULATED} = .9910}$$

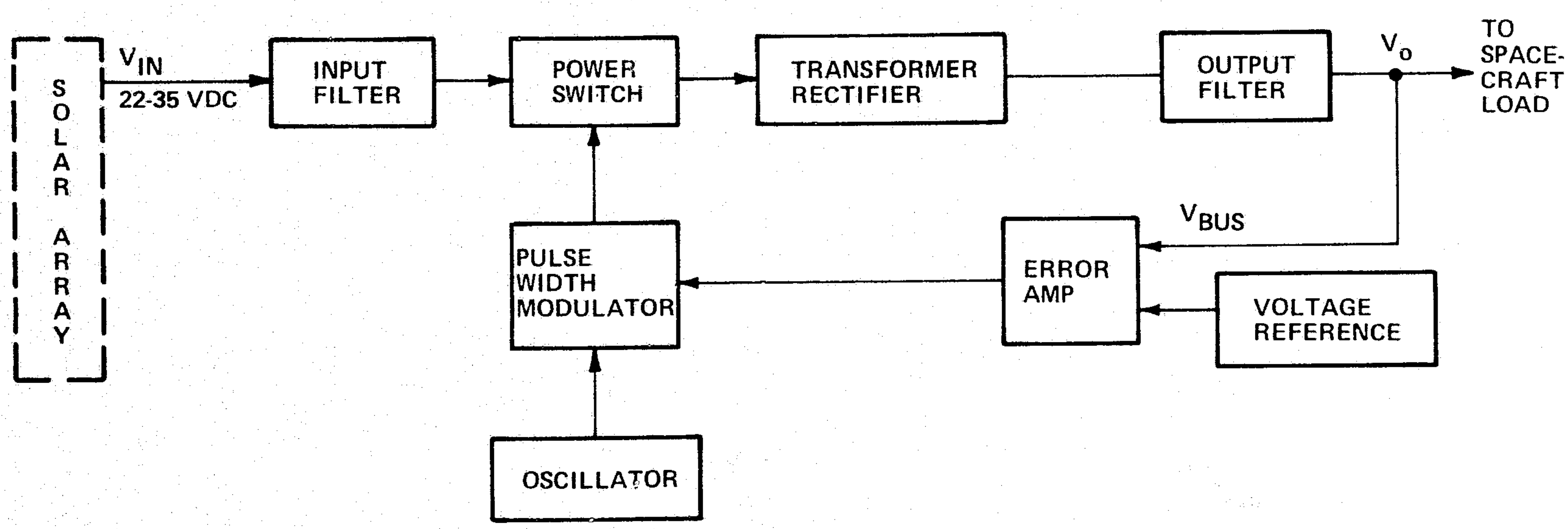
APPENDIX II-4

EQUIPMENT FUNCTIONAL BLOCK DIAGRAMS USED FOR SYSTEM COMPLEXITY ANALYSIS

SPACE DIVISION

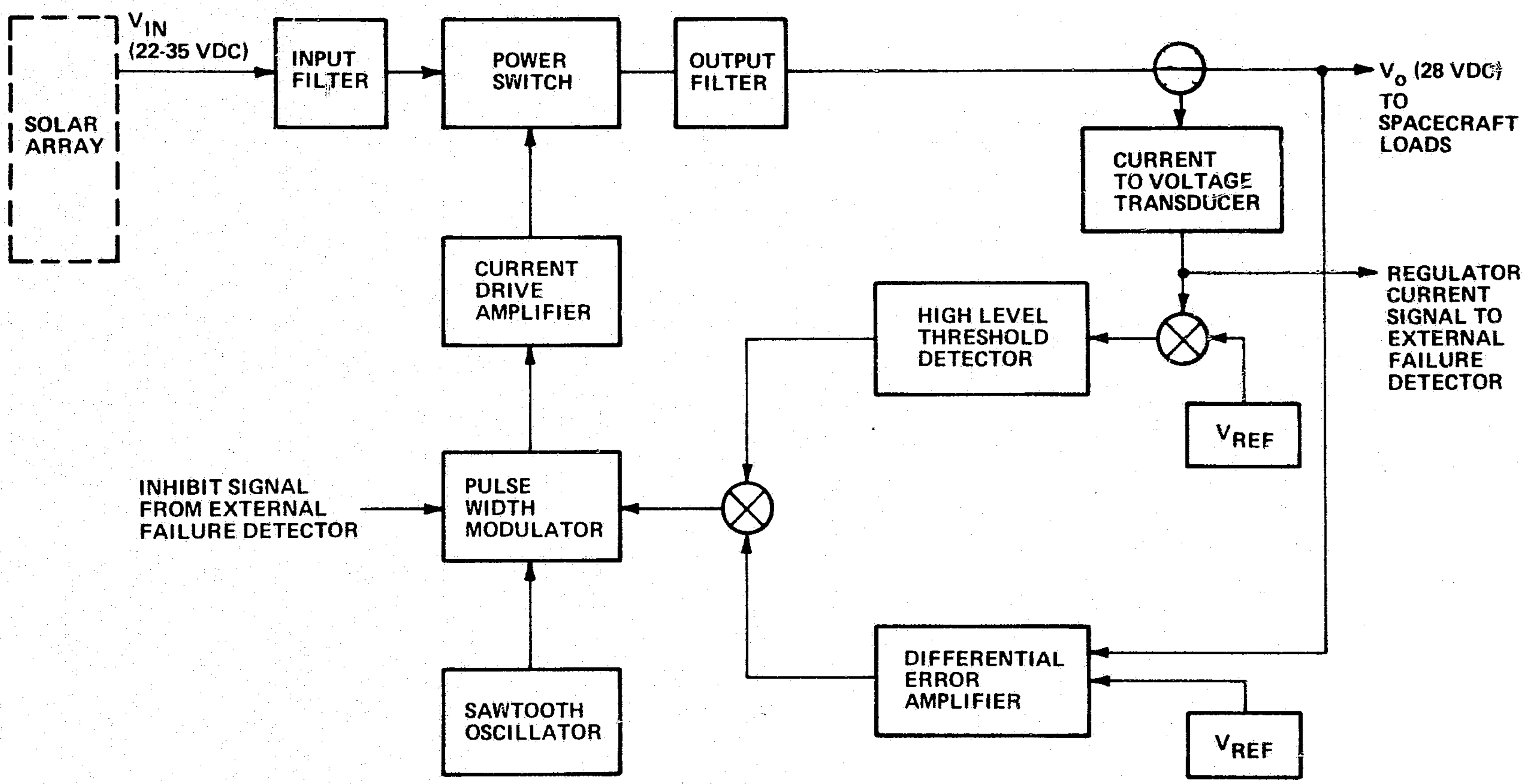
REMOTE REGULATOR FUNCTIONAL BLOCK DIAGRAM

FIGURE CO-1



SPACE DIVISION

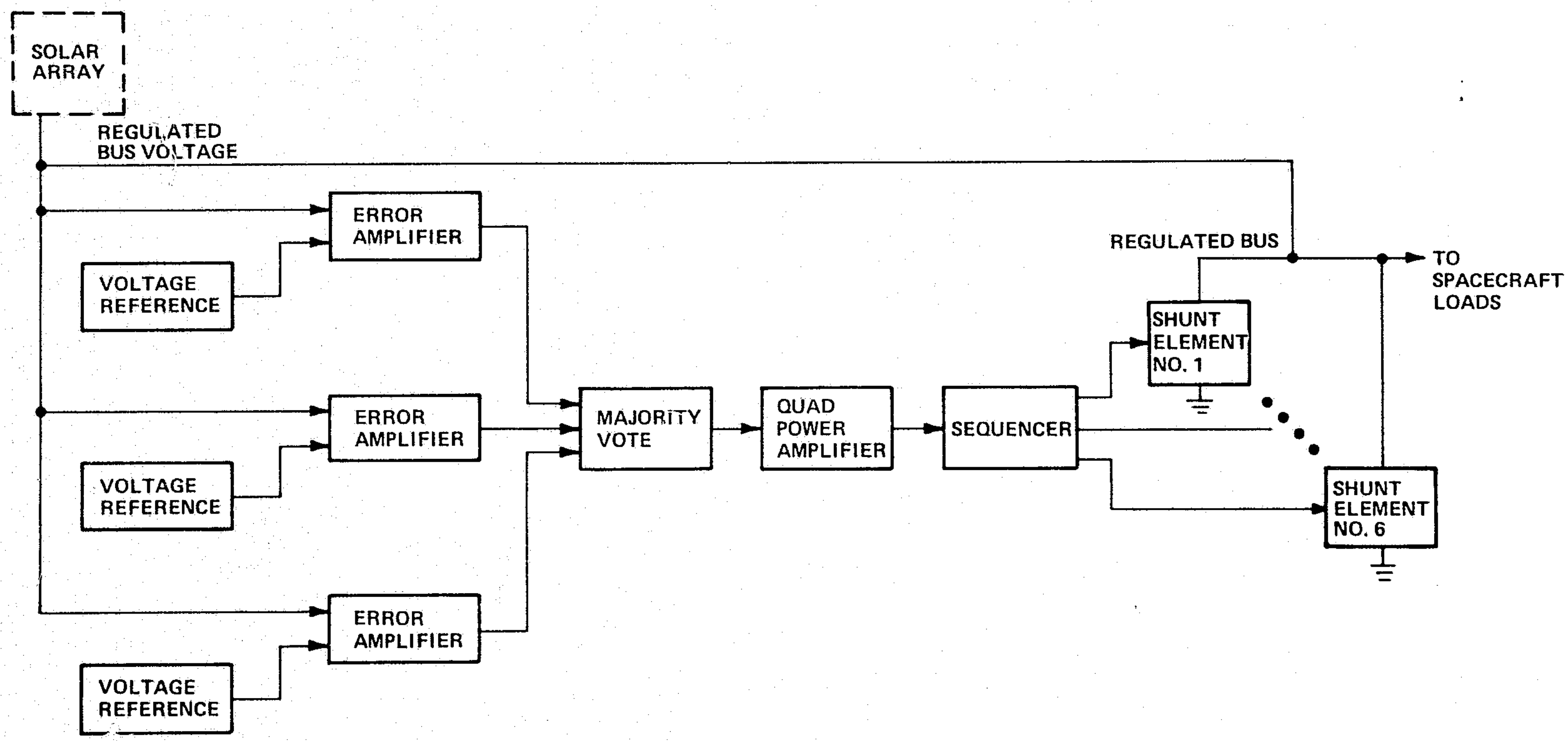
SERIES PWM REGULATOR
(WITH CURRENT LIMITING & FAULT PROTECTION)
FUNCTIONAL BLOCK DIAGRAM
FIGURE CO-2



SPACE DIVISION

SHUNT REGULATOR FUNCTIONAL BLOCK DIAGRAM

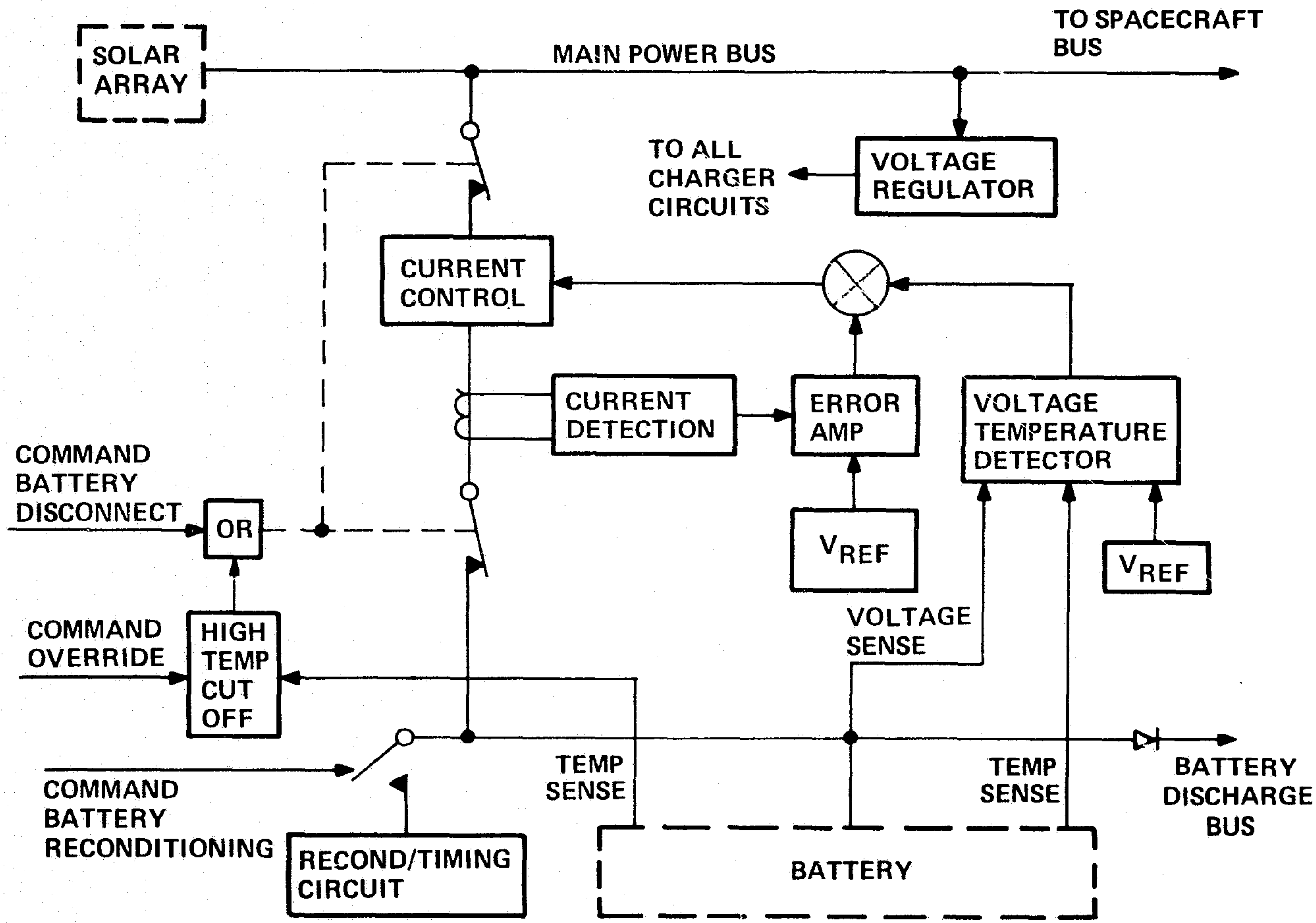
FIGURE CO-3



SPACE DIVISION

BATTERY CHARGE REGULATOR (VOLTAGE/TEMP. COMPENSATED, WITH COMMAND OVERRIDE & RECONDITIONING) FUNCTIONAL BLOCK DIAGRAM

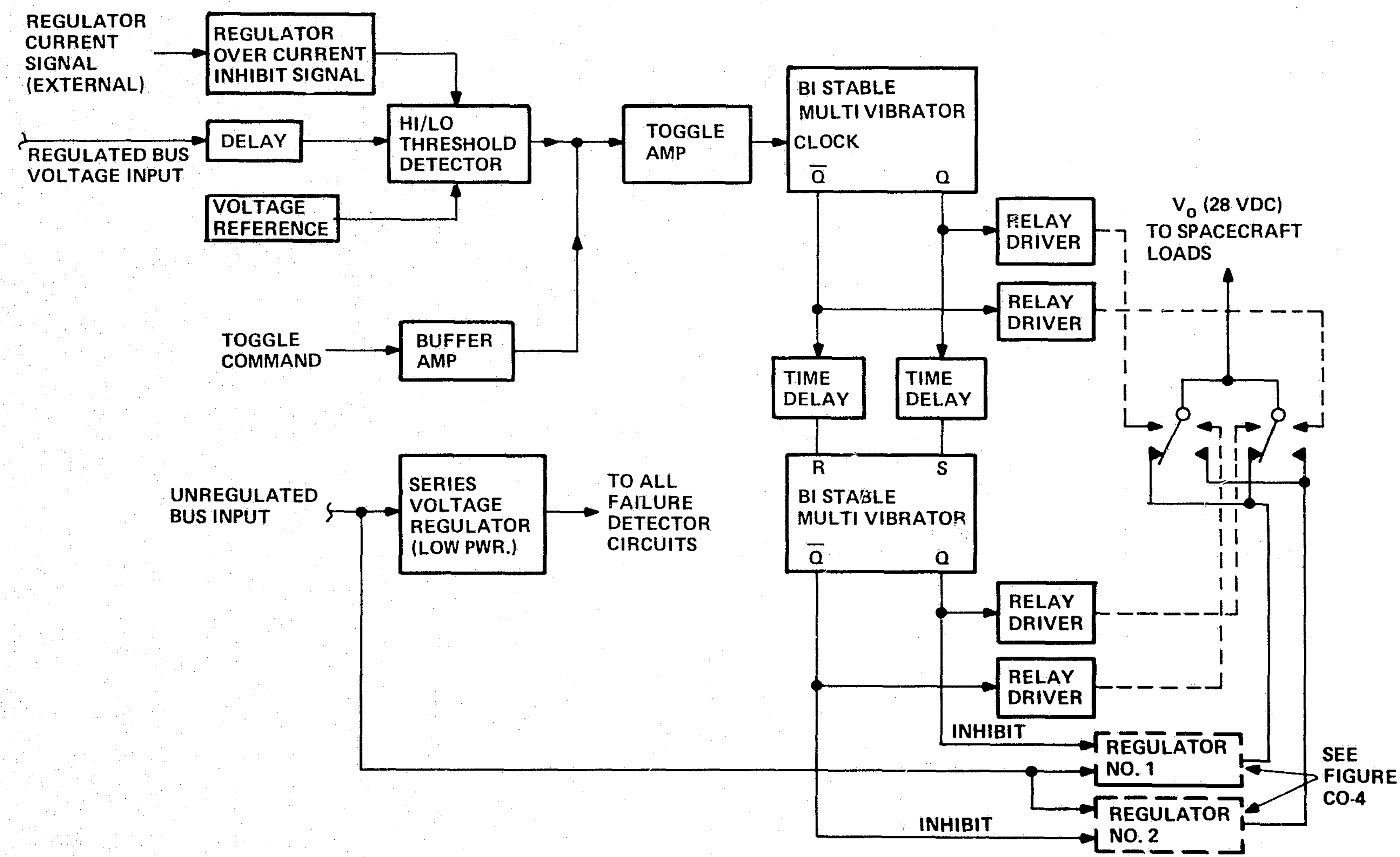
FIGURE CO-4



SPACE DIVISION

FAILURE DETECTOR (WITH HI/LO VOLT. SENSE. & SWITCHING) FUNCTIONAL BLOCK DIAGRAM

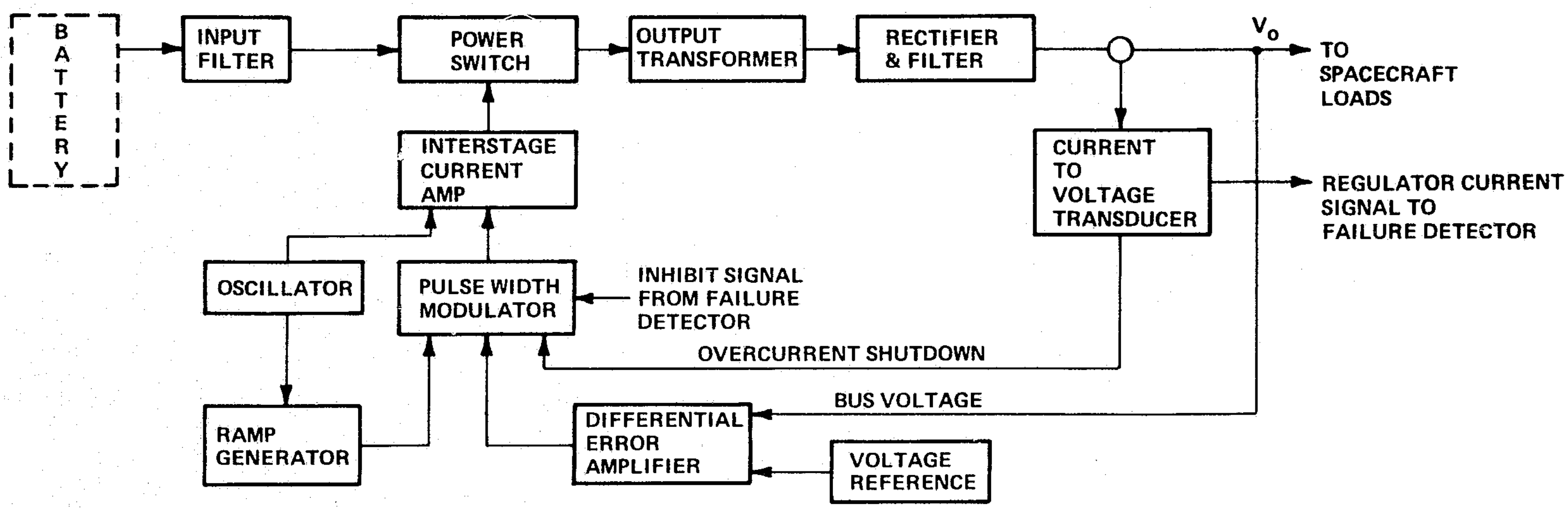
FIGURE CO-5



SPACE DIVISION

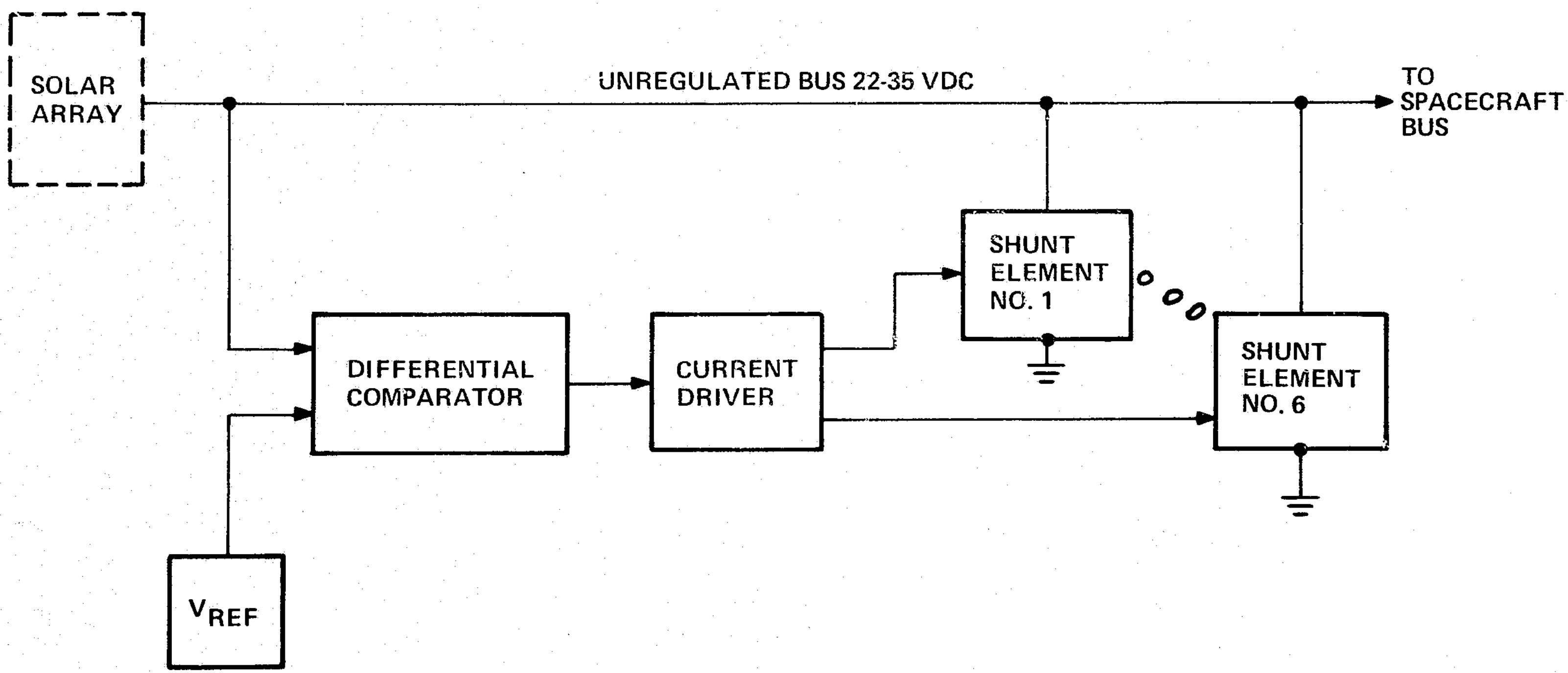
BATTERY DISCHARGE PWM REGULATOR (WITH CURRENT LIMIT & FAULT PROTECTION) FUNCTIONAL BLOCK DIAGRAM

FIGURE CO-6



SPACE DIVISION

ARRAY LIMITER
FUNCTIONAL BLOCK DIAGRAM
FIGURE CO-7



APPENDIX II-5

INTERCONNECTIONS REMOTE SYSTEM

APPENDIX II-5

INTERCONNECTIONS

REMOTE SYSTEM

Voltage Limiter	DC Input	1	
	DC Return	1	
	Resistor Panel	6	
	Back Up Commands to Disconnect Failed Shunt Leg	6	
	Individual Current Monitors	6	
	Bus Voltage	1	
	Temperature	1	
			22

Charge Regulator Plus Discharge Diode	Batt Input (+)	1	
	" " (-)	1	
	Main Bus Input	1	
	Regulated Input	1	
	Discharge Bus	1	
	Command & Override Disconn	6	
	TLM Signals V, IC, I_O, T,	4	
	TLM CMD	3	
	Failure Bus Clamp	1	
			19

Remote Regulator	DC Input (+)	1	
	" " (-)	1	
	DC Output (+)	1	
	" " (-)	1	
	Command (On)		
	" (Off)	2	
	TLM V_{IN}, V_{OUT}, I_O, Temp	4	
	Command Monitor	2	
			12

Summary

Voltage Limiter = 22
Charge Reg'r x (2) = 38
Remote Reg'r x (7) = 84
144 System Total

INTERCONNECTIONS

SERIES SYSTEM

Voltage Limiter	(Same As Remote)	22
Charge Reg'r	(Same As Remote)	19
Series Regulator	DC Input (+)	1
	" " (-)	1
	DC Output (+)	1
	" " (-)	1
	Command (On)	
	" (Off)	2
	TLM V, I_O, Cmd	3
	V_{AUX} I_{AUX}	2
	Failure Det'r Interface	2
		13

Failure Detector

Assume Instrumentation for Parallel Operation & 2 Reg's for 400 W System	DC Input (+)	1
	" " (-)	1
	Reg'r I Sig Input	2
	Reg'r Volt Sig Input	2
	Inhibit Feedback Sig	2
	Command Toggle Pwr (On)	1
	(Off)	1
	TLM V^3, I^2, T^1	6
	TLM Status	3
		20

Summary

Limiter	= 22
Charge Reg'r x (2)	= 38
Series Reg'r x (2)	= 26
Failure Det'r	= 20
	106 System Total

INTERCONNECTIONS

SHUNT SYSTEM

Charge Reg'r	(Same As Remote System)	19
Failure Det'r	(Same As Series System)	20
Shunt Regulator	DC Input (+)	1
	" " (-)	1
	Resistor Panel	6
		8

Boost Regulator		
	Regulated DC Input (+)	1
	(-)	1
	Output (+)	1
	(-)	1
	TLM I_O, T_1	2
	Batt Input (+)	2
	" " (-)	2
	Over Current Dig to Fail Det'r	1
	Failure Det'r Disable	1
	Command (On)	1
	(Off)	1
		14

Summary

Charge Reg'r x (2)	=	38
Failure Det'r	=	20
Shunt Reg'r	=	8
Boost Reg'r x (2)	=	28
		94 System Total

APPENDIX III-1

POWER EQUIPMENT PRELIMINARY REQUIREMENT

3.0 Requirements

3.1 Operating Modes

The battery charger shall operate in two distinct modes.

3.1.1 Orbital Mode (O.M.)

In this mode, the charger will control charge current from the solar array bus to a nickel cadmium battery consisting of 16 twenty ampere hour series connected cells.

3.1.2 Shuttle Mode (S.M.)

The charger will condition the shuttle unregulated fuel cell power and make it available at the battery bus to charge the battery and/or satisfy spacecraft power demand thru the discharge regulator.

3.2 Orbital Mode Performance

3.2.1 Central Control

An external analog signal from the central control circuitry will allow excess solar array current to be used for battery charging. The charge current permitted by the central controls is that required to control regulated bus voltage.

3.2.2 Charge Voltage Limits

The charger shall limit the battery voltage at the charger interface as modified by the battery thermistor resistance as shown in Figure 3-1.

Four temperature compensated voltage limits shall be selectable by ground command.

3.2.3 Charger Current Limits

The charger shall limit its output charge current to the following:

A) Low Earth Orbit - 5 amperes (c/4)

B) GEO Synchronous Orbit - 1 ampere (c/20)

The orbital charge limit shall be preset to one of the above prior to launch.

3.3 Shuttle Mode Performance

3.3.1 Fuel Cell Operation

The presence of shuttle fuel cell voltage into the charger shall cause disconnection of the charger from the regulated solar array bus and the central control charge signal. The absences shall cause reconnection to both of the above interfaces.

3.3.2 Charger Current Limits

The charger shall limit its output current to the following levels:

A) The preset orbital rate

B) During the presence of the shuttle high current enable signal, the output current limit will be changed from the orbital rate to a 9.5 ampere rate.

3.3.3 Charge Voltage Limits

The voltage limits of 3.2.2 are applicable during the normal orbital charge current rate operation. During the presence of the shuttle high current enable signal, the lowest voltage limit will be selected. Upon removal of the high current enable signal, the voltage limit will return to its previously commanded level.

3.4 Interfaces

3.4.1 Input Voltage

3.4.1.1 Regulated Bus

Voltage:	28vdc ± 2%
Dynamic Impedance:	$\leq$ 100 m . to 100 KHz
Ripple Voltage:	$\leq$ 10 mv p-p

3.4.1.2 Shuttle Fuel Cell

Voltage:	27 to 35 vdc
Ripple:	1v p-p

3.4.2 Battery

3.4.2.1 Charge Voltage

The charger shall operate to these requirements when battery voltage is between 17.6 and 24 vdc.

3.4.2.2 Battery Heater

The charger shall provide battery heater power to TBD watts from the input buses of section 3.4.1. A commandable enable/disable switch shall be included in the charger for this power line.

3.4.2.3 Battery Isolation

The charger shall provide a battery isolation switch to disconnect battery power from the charger. This switch shall be commandable to either the OFF or ON state by the spacecraft commands and by shuttle control.

A second cont act of this switch shall open the input power lines of section 3.4.1.

A digital signal shall be developed for switch status telemetry.

3.4.2.4 Battery Thermistor

Two battery thermistors will be provided to the charger. One will be used for the temperature compensation of the battery voltage limit described in section 3.2.2. The second will be conditioned to an analog signal of 0 to 5vdc for telemetry.

3.4.3 Discharge Bus

The charger shall provide a battery discharge bus to interface with the Discharge Regulator. Isolation is to be provided in the charger to prevent battery failure from loading the discharge bus and to prevent battery charging from another charger in the system.

3.4.4 Telemetry

The charger shall condition the following signals for telemetry:

Analog

1. Battery charge/discharge current
2. Battery voltage
3. Battery temperature

Digital

1. Battery/Charger On/Off Switch status
2. Voltage Limit status

3.4.5 Command

The charger shall respond to the following from the spacecraft command subsystem:

1. Voltage Limit bit 1
2. Voltage Limit bit 2
3. Voltage Limit Reset
4. Battery/Charger ON
5. Battery/Charger OFF
6. Battery Heater Enable
7. Battery Heater Disable

3.4.6 Shuttle Control

The charger shall respond to the following signals when connected to the shuttle:

1. Battery/Charger ON
2. Battery/Charger OFF
3. High Current Enable/Disable (level signal)

3.5 Fault Protection

The charger shall be fused at the input junction of the two power sources to prevent continious undervoltage on the regulated bus due to a failed charger.

The charger output to the battery shall be fused to prevent battery discharge into a failed charger.

3.6 Detailed Requirements

3.6.1 Charge Efficiency

The charger shall have a minimum charge efficiency of 85 percent when operating in the orbital charge rate modes.

3.6.2 Standby Power

The charger standby power when not charging shall not exceed TBD watts.

3.6.3 Maximum Power Dissipation

Maximum power dissipation when operating in the shuttle mode shall not exceed 40 watts.

3.6.4 Discharge Efficiency

The discharge path between battery input and battery bus output thru the isolation diode shall have a maximum voltage drop of 0.5 volts when conducting 7 amperes.

3.6.5 Input Ripple

The charger shall not produce a ripple current on the regulated solar array bus in excess of TBD milliamperes. This applies to orbital mode operation only.

3.6.6 Undervoltage

The charger must sustain an undervoltage condition for TBD seconds. Charge current will be terminated during this condition by the magnitude of the central control signal.

The minimum charger input voltage during the undervoltage period will be 15 vdc.

3.0 Requirements

The discharge regulator shall boost the battery voltage in a controlled manner to support the regulated power bus during eclipse and peak periods.

3.1 Performance

3.1.1 Central Control

An external analog signal from the central control circuitry will activate and control the discharge regulator to maintain the voltage of the regulated bus.

3.1.2 Discharge Current Limit

The discharge regulator shall limit its output current to the regulated bus at a maximum of 36 amperes.

3.2 Interfaces

3.2.1 Battery Bus Input Voltage

The discharge regulator shall provide the current of 3.1.2 to the regulated bus with a battery bus input voltage between 16 and 24vdc.

3.2.2 Regulated Bus Output

Voltage:	28vdc ± 2%
Ripple:	100 mV P-P max
Dynamic Impedance:	200 m to 100 KHz

3.2.3 Telemetry

The discharge regulator shall provide a conditioned analog signal of its output current magnitude for telemetry.

3.3 Fault Protection

The discharge regulator shall be designed to prohibit a single failure from causing a full - ON discharge state. The unit shall be fail-safe such that any single failure will result in a shut-down mode.

3.4 Parallel Operation

The discharge regulator must be capable of operation with as many as 3 identical units in parallel with it. A common central control signal will operate all parallel discharge regulators. Load sharing of the parallel regulators shall be within 15%.

3.5 Detailed Requirements

3.5.1 Efficiency

The discharge regulator shall have a minimum efficiency of 85 percent when operating below the current limit of 3.1.2.

3.5.2 Standby Power

The discharge regulator power consumption when not supporting spacecraft loads shall not exceed 4 watts.

3.0 Requirements

The central control shall provide the signals to automatically control the power bus voltage throughout all phases of mission operation.

3.1 Performance

The central control shall measure the regulated power bus and provide signals to the discharge regulator, charger, and shunt dissipator to maintain the bus within ±2 percent regulation.

3.1.1 Solar Array Priority

The central control shall provide priority for solar array power as follows:

1. Loads
2. Battery charging
3. Shunt dissipation

3.1.2 Operating Modes

The central control shall cause operation of the three following modes:

1. Battery Discharge
2. Battery Charge
3. Shunt Dissipation

The percent of modal operation vs the percent bus voltage deviation is shown in figure 1.

There shall not be any overlap of the three operating modes. That is, the operating mode must be full ON or full OFF before the ajacent mode turns ON.

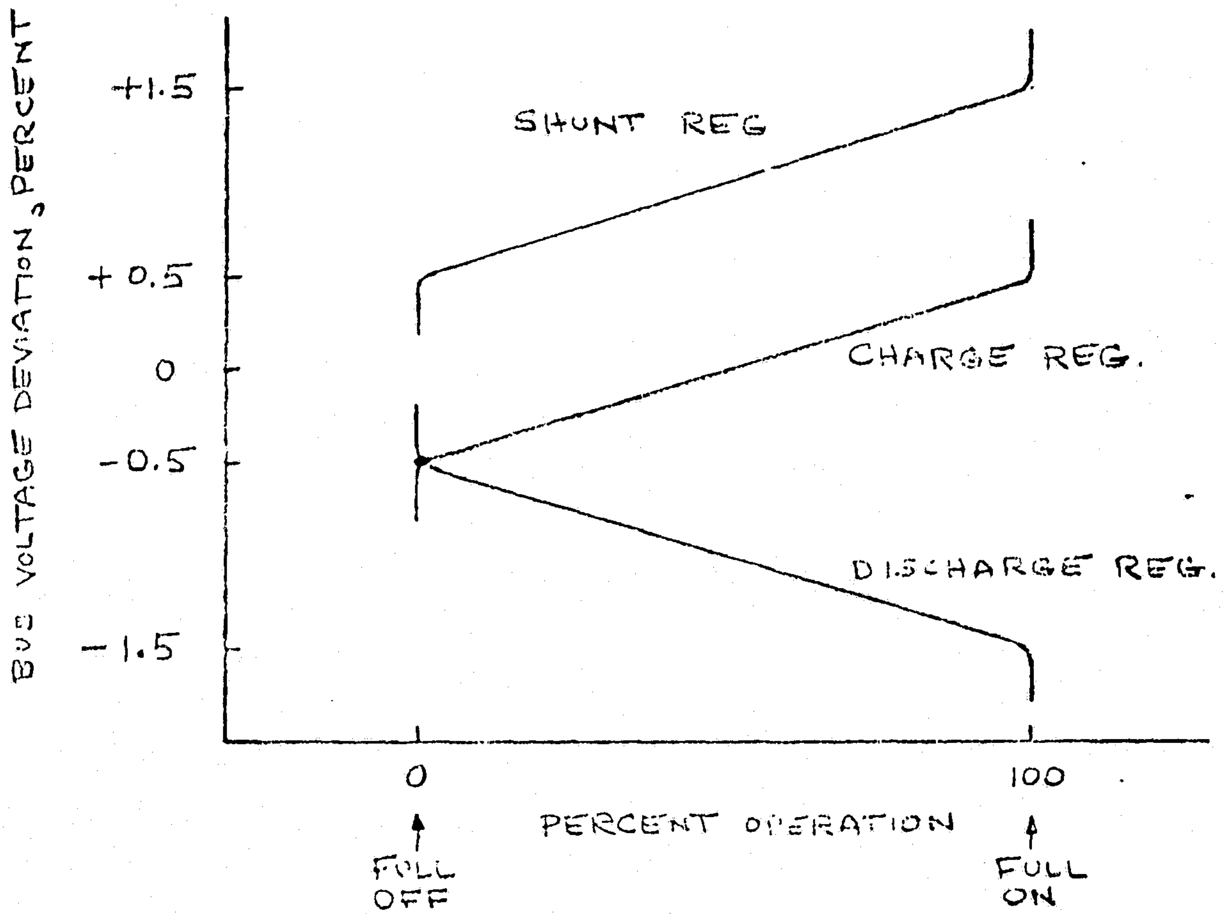


FIGURE 1 DIRECT ENERGY TRANSFER (DET) SYSTEM

3.2 Interfaces

3.2.1 Regulated Bus

The central control shall operate battery discharge regulators, chargers, and shunt dissipators to provide the following:

Voltage:	28 vdc ± 2%
Ripple:	100 mV P-P max
Dynamic Impedance:	200 m to 100 KHz

3.2.2 Discharge Regulator

The central control shall provide an analog signal to control operation of the discharge regulator. A decreasing voltage will require an increase in the percentage of discharge regulator operation. This signal shall be capable of driving 3 parallel discharge regulators.

3.2.3 Battery Charger

The central control shall provide an analog signal to control operation of the battery charger. A decreasing voltage will permit an increase in the percentage of charger operation. This signal shall be capable of driving 14 parallel battery chargers.

3.2.4 Shunt Dissipator

The central control shall provide an analog signal to control operation of the shunt dissipator. An increasing voltage will require an increase in the percentage of shunt dissipator operation. This signal shall be capable of driving 8 parallel shunt dissipators.

3.3 Fault Protection

No single piece part failure shall prohibit any function of the central control. Majority voting and quad redundancy shall be used where appropriate to meet this requirement.

3.4 Detailed Requirements

3.4.1 Power

The central control power consumption shall be less than 4 watts.

3.0 Requirements

The shunt dissipator shall control the operating point of the solar array to maintain the array voltage at a constant value and hence develope a regulated bus.

3.1 Performance

The activation of the shunt dissipator is accomplished by the central control signal which is related to the regulated bus voltage error.

3.2 Interfaces

3.2.1 Solar Array

The solar array shall be segmented into circuits. Each circuit shall contain an upper and lower array. The shunt dissipator shall be designed for a solar array whose cell characteristics are as shown on Figure 1.

The total array circuit shall consist of no more than 5 parallel cells and contain between 75 and 85 cells in series. The upper array circuit shall have a minimum of 30 series cells.

A shunt dissipator will control the power provided by 14 solar array circuits by shunting the lower circuits of the solar array.

3.2.2 Central Control

An external analog signal from the central control circuitry will activate and control the shunt dissipator via the quad driver circuit.

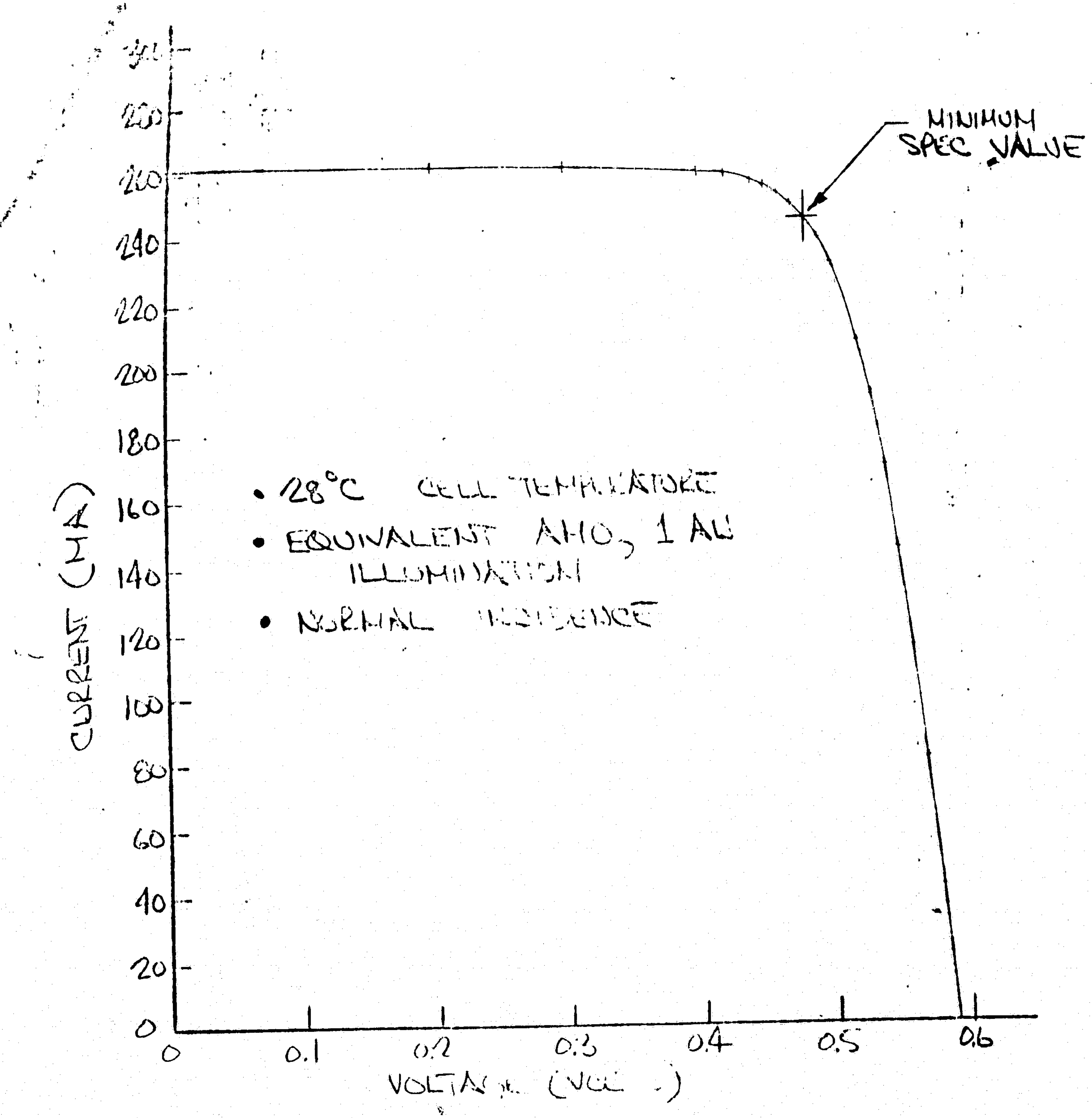


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Figure Minimum Lot Average Covered Solar Cell Performance

3.3 Detailed Requirements

3.3.1 Number of Shunt Elements

There shall be 12 Shunt Elements per Shunt Dissipator. The power provided by the 2 unshunted solar array circuits will be handled by a quad driver amplifier.

3.3.2 Element Current

Each shunt element shall be capable of shunting a maximum of 1.5 amperes.

The shunt element shall contain protection such that in the event of a failure of a solar array isolation diode, no more than 2 amperes ±20% can flow thru a shunt element from the solar array tap point.

3.3.3 Element Voltage

The maximum voltage across a shunt element (array tap point voltage) shall be 20 vdc.

3.3.4 Sequencing Control

The 12 shunt elements shall be arranged into 6 circuits each consisting of 2 parallel shunt elements.

As the quad driver current increases, the sequencer shall cause each circuit to turn-on in a manner that avoids simultaneous peak power dissipation in the 6 circuits.

3.3.5 Quad Driver

The shunt quad driver shall receive its operating power from the +28vdc bus. The maximum current drawn from this bus shall be 3 amperes.

3.4 Parallel Operation

The shunt dissipator must be capable of operation with as many as seven identical units in parallel with it . A common central control signal will operate all parallel shunt dissipators.

APPENDIX III-2

POWER MODULE THERMAL ANALYSIS

APPENDIX III-2

POWER MODULE THERMAL ANALYSIS

Introduction

A thermal evaluation of Power Modules for general application to a Satellite Bus was undertaken to verify an adequate thermal control concept. The thermal requirements include:

a) Battery temperature range $0^{o}C$ to $25^{o}C$.

b) $5^{o}C$ maximum temperature difference between batteries mounted to the same module.

c) $5^{o}C$ maximum temperature difference between batteries mounted in two modules.

d) Nominal temperature range (i.e. $0^{o}C$ to $40^{o}C$) for other module components.

One or two modules is required per spacecraft and the spacecraft mission is either a low earth altitude . or a geosynchronous altitude.

Discussion:

Layouts were completed for both a 48" x 48" panel module and a 40" x 48" panel module. Summary thermal data for the module components are presented in Table 1. The orbital geometry for the modules is presented in Figure 1 for both the low earth orbit and geosynchronous orbit configurations. Both modules are placed on the non-sun side of the spacecraft for the low earth orbit mission, and on the north side of the spacecraft for the geosynchronous orbit mission, thus minimizing the heat flux incident on the module radiating area. The thermal requirements for each size panel for only one module and for two modules is presented on Table 2. Per the module layouts, the number of each type component is presented along with the total power dissipation for both the low earth orbit and geosynchronous orbit mission, using the dissipation data from Table 1 and the component compliment defined in Table 2. Radiator requirements were then determined for each module for both orbital missions. For low earth orbit applications, where the orbital period of 100 minutes is small compared to the component thermal time constants, the module radiation areas required can be sized based on the average orbital environments, average orbital power dissipation and $10^{o}C$, the average desired battery temperature. For geosynchronous orbit applications, where the orbital period of 24 hours is large compared to the component thermal time constants and the sun angle is seasonal, module radiators must be sized for the maximum conditions using the summer solstice solar illumination and solstice power dissipation with a module radiator temperature of 20°C, near the maximum temperature allowed. The resulting radiator area requirements, shown on Table 2, indicate that adequate heat rejection area is available in all cases. Given the established design, the minimum average module temperature can be defined. For constant dissipations, the only change for the low earth orbit is due to the difference between degraded and undergraded thermal coating optical properties. With the module location defined (Figure 1) this change has a minimal effect and the minimum average temperature is 9°C, a 1°C range from the 10°C design point. For the geosynchronous orbit, the minimum case occurs at winter solstice when the sun is on the other side of the spacecraft. A minimum average temperature of -3.3°C results, indicated a need for heaters to maintain the required 0°C. The heater power requirements per module to hold 0°C are shown to be acceptably small. The average module temperature at equinox is shown to be somewhat higher since, at equinox, the average heat dissipation is higher, thus somewhat compensating for the lack of solar illumination.

The designs established are quite feasible based on average module conditions. In order to determine the battery to battery temperature differences and orbital transients, detailed transient thermal models were established for the 48" x 48" panel Case II designs defined for modules 1 and 2 for both the low earth orbit and geosynchronous orbit designs, as shown on Figures 2 and 3. Since the component layouts are thermally symmetrical, the required radiation areas were taken to be symmetrical and only one-half of the module was modeled. The analysis results are shown on Tables 3 and 4 for Modules 1 and 2 respectively. For Module 1, a single and double radiator thermal concept was evaluated to determine the sensitivity of the design to the high power discharge regulators. This concept was approximated by decoupling the conduction between panel elements along the double radiator boundary shown on Figure 2.

Results

The results show that detailed optimization is required to meet all requirements, but that attaining these requirements is feasible with a single radiator concept. For module 1, the average module temperature level must be reduced with an average panel temperature reduced from 10°C to nearer 0°C which will result in an average battery temperature of about 15°C. The radiator area increase to 15 ft^2 at 0°C (from 12.33 ft) should favor the high power discharge regulators, thus simultaneously lowering their maximum temperatures at least 10°C. The module 1 geosynchronous orbit design must also have its maximum average temperature lowered to nearer 15°C. The radiator area increase to 5.17 ft^2 at 15°C (from 4.69 ft^2) will increase the required module heater power from 5.35 watts to 16.6 watts which is still considered to be acceptable. The module 2 design will meet all requirements with a slight tailoring of the radiation area location relative to the module components. Detailed analyses will be required to finalize any module design established.

Conclusions

The power modules evaluated can be designed for both low earth and geosynchronous orbit applications with a single radiator passive thermal design (coatings plus insulation) supplemented by heater power for geosynchronous orbit applications. The thermal analysis required to determine average design temperature levels and thermal coating patterns is very detailed, due to the large variation in day/night component dissipations and large component thermal masses, requiring a transient thermal model and multiple iterations.

Table 1

Summary Data

Component	Size (Inches)	Weight (#)	Thermal Capacitance (BTU/°F)	Maximum Power Dissipation - Watts: Low Earth Orbit Night (1/3)	Low Earth Orbit Day (2/3)	Low Earth Orbit Avg.	Geosynchronous Orbit Equinox Night (1.2 Hrs)	Geosynchronous Orbit Equinox Day (22.8 Hrs)	Geosynchronous Orbit Equinox Avg.	Geosynchronous Orbit Solstice
Battery	8x9x8	44.	6.6	35.	7.5	16.6	40.	6.8	8.5	6.8
Charger	6x9.5x8.5	9.	2.25	7.5	15.	12.5	9.	7.	7.1	7.0
Central Control	4x4.5x4	4.	1.0	1.	5.	3.7	1.	5.	4.8	5.0
Discharge Regulator	8.5x9.5x5	12.	3.0	17./* Battery	4.	-+	25./* Battery	4.	-+	4.0
Remote Decoder Mux.	6x4x2	2.	0.5	1.2	1.2	1.2	1.2	1.2	1.2	1.2

* 100.0 Watts Maximum per Regulator

\+ Function of No. of Batteries

THERMAL REQUIREMENTS

Parameter	48" x 48" Panel (15.16 ft^2)			40" x 48" Panel (12.5 ft^2)		
	Module 1		Module 2	Module 1		Module 2
	Case I	Case II	Case II Only	Case I	Case II	Case II Only
Number of Components						
Batteries	7	6	10	5	4	8
Charger	7	6	10	5	4	8
Central Control	1	1	-	1	1	-
Discharge Regulator	3	4	-	3	4	-
Remote Decoder Mux	1	1	1	1	1	1
Total Power Dissipation						
Low Earth Orbit						
Day (Watts)	175.7	157.2	226.2	130.7	112.2	181.2
Night (Watts)	418.7	529.2	426.2	299.7	376.2	341.2
Average (Watts)	254.2	280.0	292.2	186.5	199.3	234.0
Geosynchronous Orbit						
Equinox:						
Day (Watts)	114.8	105.0	139.2	87.2	77.4	111.6
Night (Watts)	520.2	696.2	491.2	372.2	498.2	393.2
Average (Watts)	135.1	134.6	156.8	101.5	98.4	125.7
Solstice: (Watts)	114.8	105.0	139.2	87.2	77.4	111.6
Radiator Area (ft^2)						
Low Earth Orbit At 10°C & Avg. Diss.	11.20	12.33	12.87	8.22	8.78	10.31
Geosynchronous Orbit at 20°C & Summer Solstice Diss.	5.13	4.69	6.22	3.89	3.46	4.98
Minimum Avg. Temperature (°C)						
Low Earth Orbit	9.0	9.0	9.0	9.0	9.0	9.0
Geosynchronous Orbit						
Equinox	7.7	13.8	4.7	7.2	13.3	4.8
Winter Solstice	-3.3	-3.3	-3.3	-3.3	-3.3	-3.3
Heater Power (Watts)						
For 0°C Min. Temp. at Geosynchronous Orbit Winter Solstice	5.85	5.35	7.09	4.43	3.94	5.68

Table 3

Module 1 - Case 2 Temperatures - °C

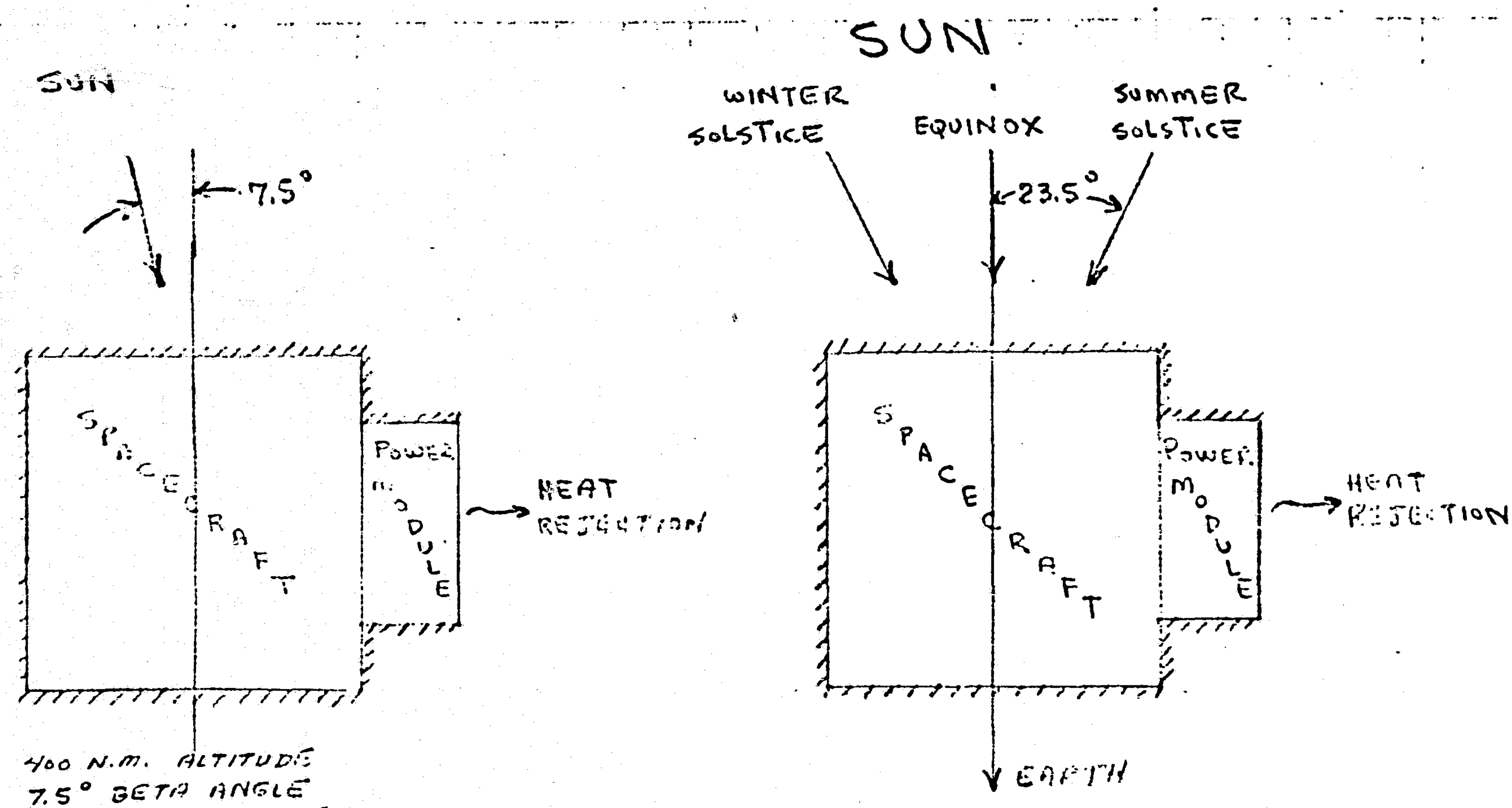
Component	Node No.	Low Earth Orbit						Geosynchronous Orbit									
		Single Radiator			Double Radiator			Single Radiator					Double Radiator				
		Tmax	Tmin	Tavg	Tmax	Tmin	Tavg	Solstice	Equinox				Sols	Equinox			
								Tmax	Start of Umbra Tmin	After 72 Min Umbra	Tmax	72 Min Umbra +4 Hr	Tmax	Start of Umbra Tmin	After 72 Min Umbra	Tmax	72 Min Umbra +4 Hr
Battery Charger	1	20.3	17.2	18.7	9.5	6.8	8.2	21.9	-1.4	1.7	9.5^{+}	9.5	27.0	4.4	7.6	11.0^{+}	11.0
Battery Charger*	2	32.6	29.6	31.1	50.4	46.8	48.5	21.4	-2.1	3.8	10.4^{+}	10.4	16.4	-7.8	-1.8	10.2^{+}	10.2
Discharge Regulator*	3	56.9	42.4	49.7	78.9	63.9	71.9	20.7	-2.5	56.2	56.2	20.9	13.3	-11.0	53.0	53.0	26.8
Battery	4	27.3	24.2	26.1	15.1	12.3	13.7	25.3	2.2	14.8	17.8	17.8	31.7	9.3	19.6	19.6	17.9
Remote Dec/Mux	5	28.5	26.6	27.4	17.9	16.6	17.2	28.3	5.1	11.1	20.1^{+}	20.1	34.3	11.8	17.1	20.8	20.8
Discharge Regulator*	6	57.5	43.0	50.3	78.8	63.9	71.9	21.8	-1.4	57.4	57.4	22.3	13.3	-11.0	53.0	53.0	26.8
Battery	7	28.6	25.5	27.4	17.4	14.6	16.1	27.6	4.6	17.1	20.1	20.1	34.6	12.4	22.7	22.7	21.5
Central Control	8	33.9	30.2	32.1	24.4	21.3	23.1	37.0	13.9	13.1	28.4^{+}	28.4	43.6	21.2	20.0	30.5^{+}	30.5
Battery Charger*	9	34.0	31.1	32.6	50.4	46.8	48.5	22.6	-.6	6.4	13.2	13.0	16.4	-7.9	-1.8	10.2^{+}	10.2
Battery	10	21.5	18.8	20.4	14.3	11.7	13.0	22.3	-.9	9.5	10.3^{+}	10.3	28.2	5.9	16.7	16.7	15.6
Battery to Battery ΔT Max				7.0			3.1	5.3	5.7	7.6	9.8	9.8	6.4	6.5	6.0	6.0	5.9
Module to Module 2 Battery to Battery ΔT Max.				15.5			6.9	10.4	8.5	10.6	16.8	16.8	17.4	16.3	16.2	16.2	18.2

*Components on Second Radiator in Double Radiator Design

Table 4

Module 2 - Case 2 - Temperature - °C

Component	Node No.	Low Earth Orbit			Geosynchronous Orbit				
					S. Solstice	Equinox			
		T_{max}	T_{min}	T_{avg}	T_{max}	Start of Umbra T_{min}	After 72 Minute Umbra	T_{max}	72 Minute Umbra Plus 4 Hours
Battery Charger	1	11.5	9.2	10.4	17.4	-3.8	- .5	2.1	2.0
Battery	2	13.9	11.1	12.5	17.2	-3.9	6.5	6.5	3.7
Battery	3	20.1	17.3	18.8	20.1	-0.7	9.4	9.4	6.2
Battery	4	19.3	16.4	17.9	20.1	-0.8	9.6	9.6	6.9
Battery Charger	5	24.6	22.0	23.3	24.0	3.3	6.1	8.3^{+}	8.3
Battery Charger	6	23.8	21.3	22.6	23.7	3.1	6.9	9.5	9.3
Battery Charger	7	15.9	13.5	14.8	20.4	-0.5	2.1	4.0^{+}	4.0
Battery	8	18.3	15.5	16.9	20.2	- .7	9.3	9.3	6.1
Battery Charger	9	10.2	7.8	9.1	17.5	-3.8	-1.1	.8	.8
Battery	10	13.3	10.5	11.9	17.3	-3.9	6.5	6.5	3.3
Battery To Battery ΔT Max		6.8	6.8	6.9	3.0	3.2	3.1	3.2	3.6



400 N.M. ALTITUDE
7.5° BETA ANGLE
SUN - SYNCHRONOUS
97° INCLINATION

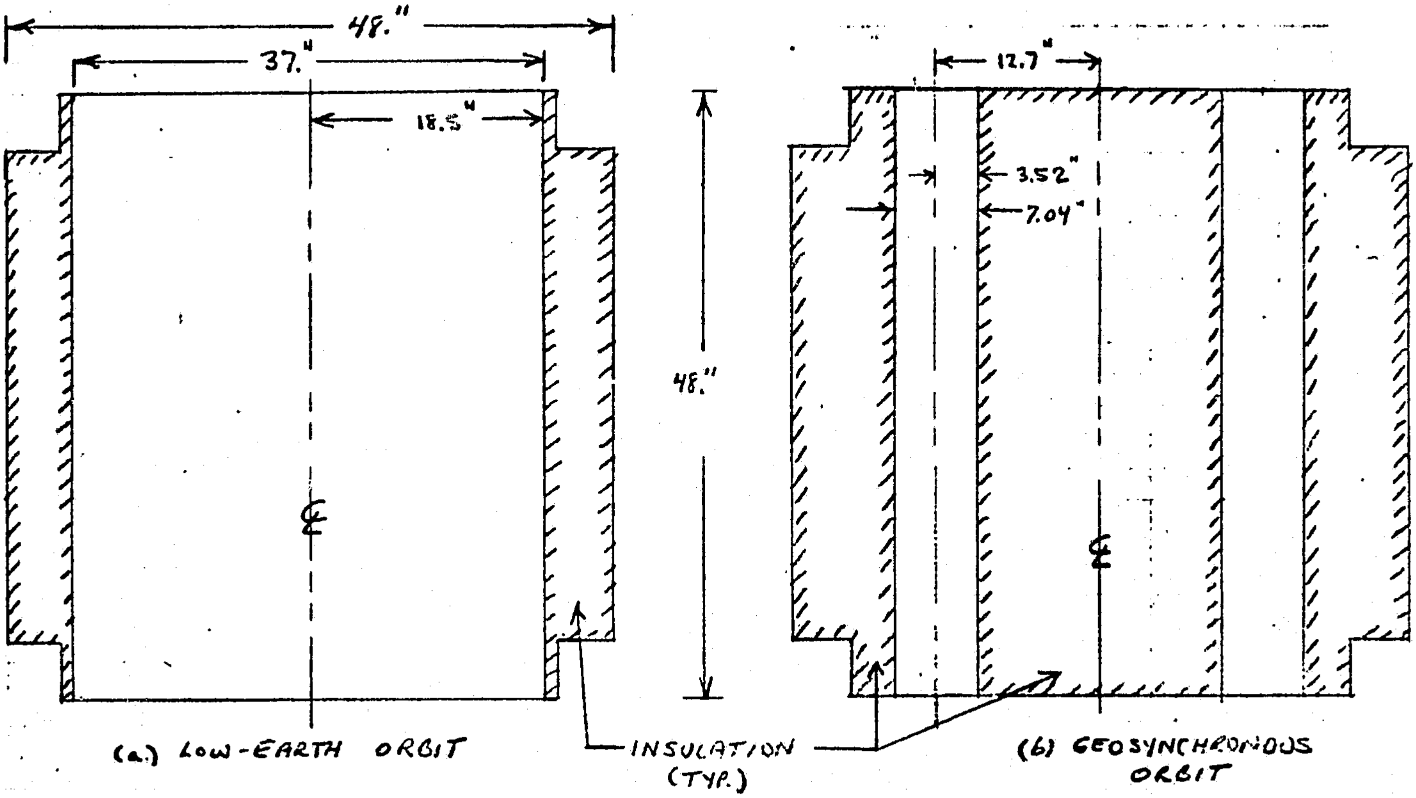
AVG INCIDENT SOLAR FLUX = 0.0
AVG INCIDENT ALBEDO FLUX = 9.2 BTU/hr ft^2
AVG. INCIDENT EARTH FLUX = 17.0 BTU/hr-ft^2

SOLAR CONSTANT = 419. BTU/hr ft^2

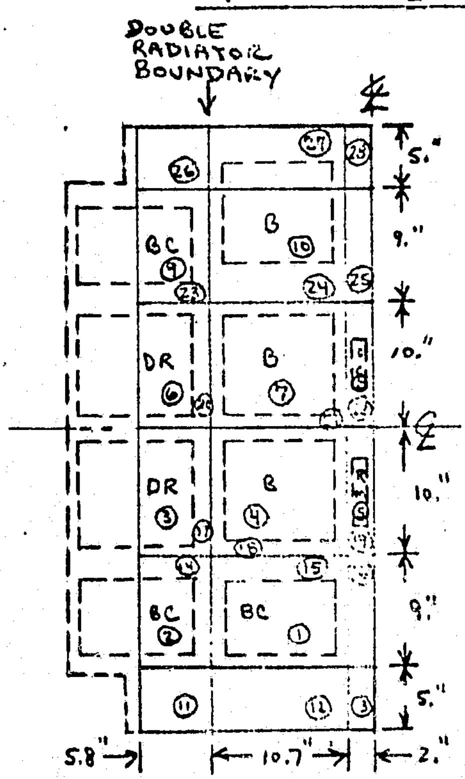
(a) LOW EARTH ORBIT

(b) GEOSYNCHRONOUS ORBIT

FIGURE 1 - ORBITAL GEOMETRY



MODULE #1 - CASE II THERMAL COATING PATTERN



28 NODE MODEL
NODAL DEFINITION:
1-10 - COMPONENTS
11-28 - BASE PLATE

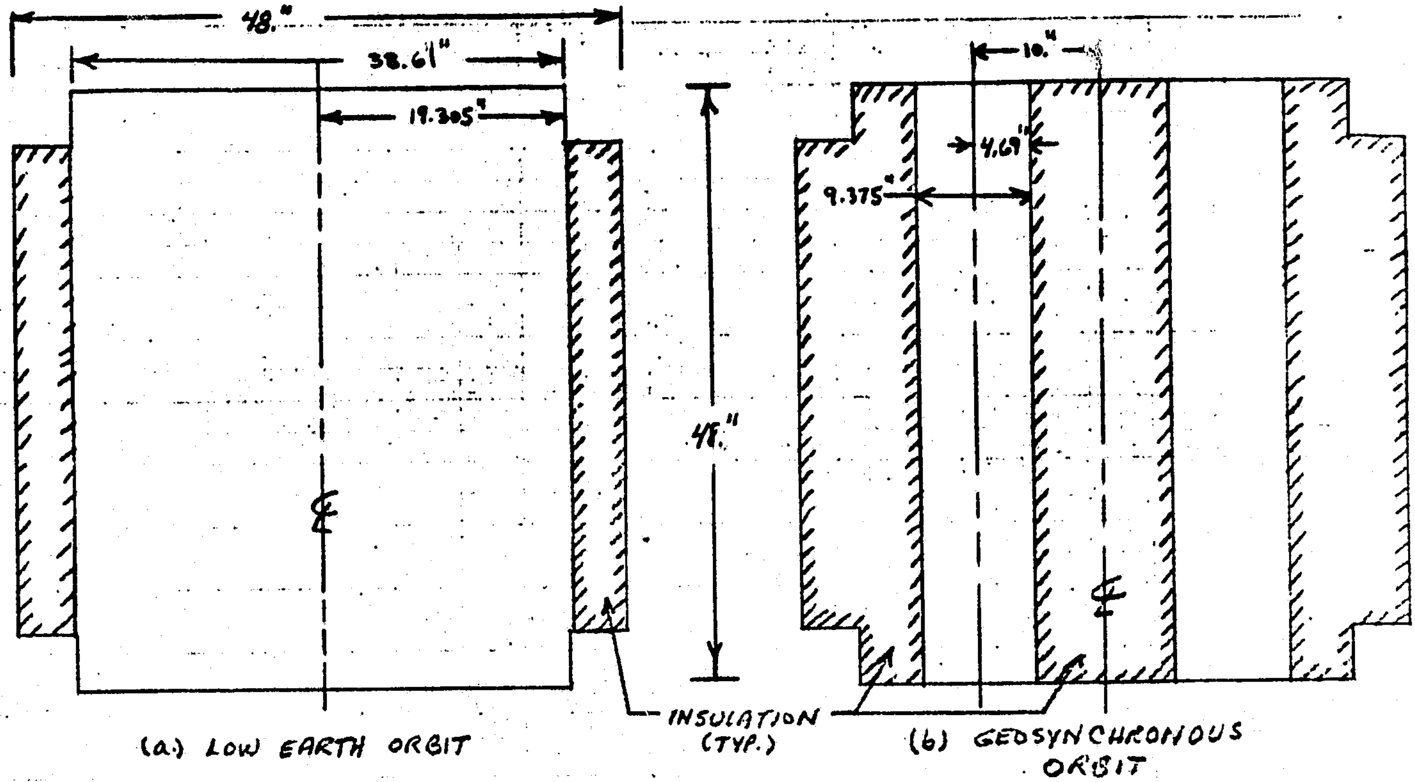
THERMAL COATING:
5 MIL TEFLON/SILVER
α/ϵ from $0.08/0.80$ to $0.18/0.80$

PANEL
EFFECTIVE THICKNESS = 0.06"
CONDUCTIVITY = 100 BTU/hr ft °F
COMPONENT INTERFACE CONDUCTANCE
= 325. BTU/hr ft²°F

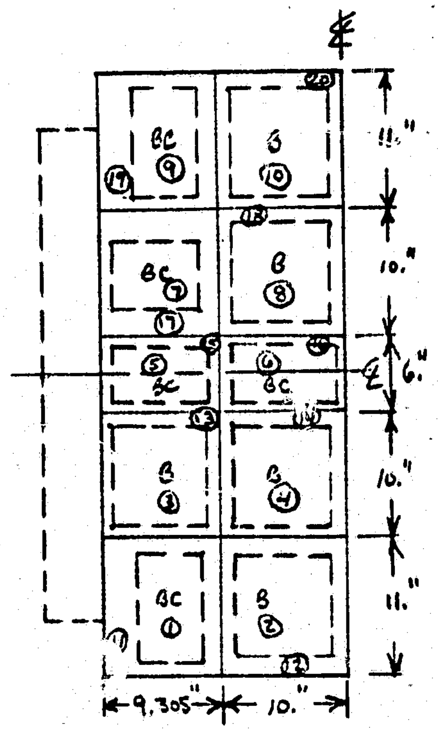
INSULATION ADIABATIC
NO APPENDAGE BLOCKAGE

FIGURE 2

MODULE #1 THERMAL MODEL - 1/2 SYMETRICAL



MODULE #2 - CASE II THERMAL COATING PATTERN



20 NODE MODEL
NODAL DEFINITION
1-10 - COMPONENTS
11-20 - BASEPLATE

FIGURE 3

MODULE #2 THERMAL MODEL - 1/2 SYMETRICAL