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SECOND INTERIM TECHNICAL REPORT

Project A-1251

SPACE VEHICLE ELECTRICAL POWER SYSTEMS STUDY

S. L. Robinette, G. W. Bechtold,
and G. W. Spann

CONTRACT NAS8-25192

22 January 1971

Prepared for
National Aeronautics and Space Administration
George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama

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GEORGIA INSTITUTE OF TECHNOLOGY
Atlanta, Georgia

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GEORGE C. MARSHALL SPACE FLIGHT CENTER
MARSHALL SPACE FLIGHT CENTER, ALABAMA

FOREWORD

This report was prepared at the Electronics Division of the Georgia Institute of Technology under Contract NAS8-25192. The work was performed within the Communications Branch under the general supervision of Mr. D. W. Robertson, Head of the Communications Branch. The report covers the activities and results of the Phase II effort on a project to aid in the development of electrical power systems for future space vehicles.

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ABSTRACT

Previous designs of electric power distribution systems for spacecraft have been based on predetermined power sources. Design studies centered on sources rather than on load requirements have been widely at variance in categorizing distributed power requirements, and excessive conversion equipment had to be included to condition the distributed power for load utilization.

In this study the design begins with load power categorizations and proceeds to suggested models for distribution systems. Load requirements included in the categorizations are: (1) percent of load power utilized as ac, dc, and either ac or dc; and (2) load power which is essential, non-essential, and emergency. Categorizations indicate that load power utilization in Space Station/Base would be almost evenly divided between ac and dc. When the power quality requirements of loads are examined, however, it is seen that motors, solenoids, heaters, and lights can utilize unconditioned power. Ac motors would be preferred over dc motors, and it would be desirable to supply all the raw-power loads without power conversion. These load requirements favor ac distribution. Two design factors which favor dc distribution are (1) the ease of paralleling power sources, and (2) the simplicity of dc components for multiplexed power control by data bus.

A review of two recent, large scale studies of Space Station/Base power distribution systems indicated that insufficient attention was given to corona onset voltage. Recommended voltages were high, and weight penalties for pressurization of conduits were not applied in these trade off analyses.

In this report two alternative distribution systems are modeled for Space Station: first, a 110 Vdc system, with conversion of some power to 115 Vac, 400 Hz, and to 28 Vdc; second, a 115/200 Vac, 400 Hz system with conversion of some power to 115 Vdc and 28 Vdc. The latter distribution system is recommended for Space Station with power supplied by an isotope/Brayton source.

A 115/200 Vac, 400 Hz system is recommended for Space Base, and consideration of isotope/Brayton sources for both Space Station and Space Base is urged.

A dual 115 Vdc and 115/200 Vac, 400 Hz, 3 Φ distribution system is recommended for the orbiter, and a 115/200 Vac, 400 Hz, 3 Φ distribution system is recommended for the booster.

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I. INTRODUCTION

A. Statement of the Problem

The purpose of this program was to aid in the development of potential concepts for electrical power systems (EPS) for future spacecraft, including Space Shuttle, Space Station, and Space Base. One aim has been to examine the state-of-the-art of advanced EPS components and to suggest ways of incorporating desirable advanced components in spacecraft power systems. The second aim has been to determine whether other recent EPS studies have taken into account environmental constraints such as corona onset voltage. The third aim was to reexamine recent EPS models by beginning the modeling process with load requirements rather than with the characteristics of a power source. Finally, the fourth aim has been to delineate areas where moderate research might be expected to produce large improvements in EPS.

B. Background

Phase I of this project was an extensive search of the literature for information related to the design of the electrical power systems for Space Shuttle and Space Station/Base vehicles. The search was not limited to devices and techniques which have been used on spacecraft but was extended to such areas as military and commercial aircraft, hydrofoil craft, and rapid transit vehicles. Among the sources of information were reports from NASA and DOD, proceedings of conferences which dealt with electrical power systems, and articles published in technical journals. A summary of the literature search was presented in an Interim Report [1]. The effects of environmental conditions on electrical subsystems were reviewed. Various estimates of power system load requirements for Space Shuttle and Space Station/Base were compared. Major power requirements were analyzed to determine a basis for specifying operating parameters, e.g., voltage, frequency, peak power, etc. A number of power distribution systems were examined, and the schemes which offered the best alternatives were identified.

The Interim Report also examined possibilities of transferring the advanced technology which is being developed for aircraft, hydrofoils, and other vehicles. The status of multiplex power distribution systems and solid state switching devices for aircraft electrical systems was reported. Trade off analyses of solid state power controllers (circuit breakers) on naval ships were studied. Techniques for reducing weights of electrical power systems in aircraft and in hydrofoil craft were reviewed.

The literature survey led to the identification of several areas in which further study could be profitable. Substantial disagreement had been found on the potential hazards associated with corona and arcing. There were also conflicting data about the current-carrying capacity and the minimum voltage drop of solid state switches. The inclusion of corona effects and solid state switch characteristics in design studies could have a direct influence on the choice of voltage levels and frequencies for spacecraft electrical systems. The reliability and efficiency of the power distribution system will depend strongly on the absence of corona, and on the operational characteristics of solid state switches.

After the Interim Report was published, additional papers were found which dealt with arcing and corona. Arcing was reported in a 28 Vac, 400 Hz, 3 ϕ power system during second stage separation of Agena rockets. The plasma generated by retrorockets was suggested as a contributing cause [2]. Electrical breakdowns at field strengths of only 100 volts per centimeter were also reported [3].

Based on information gained in the Phase I literature search, the types of load requirements of spacecraft are categorized in this report (Chapter II) so as to indicate what percentages of the power need to be ac, dc, essential, nonessential, and emergency. The categorization is accomplished for the Space Shuttle by assuming some functional similarities to aircraft. The categorization of load requirements for Space Station/Base is based on a recently reported tabulation of power requirements for environment control and life support (EC/LS) and other subsystems.

In order to identify candidate electrical power systems for spacecraft, a number of energy sources, power conversion methods, conditioning techniques, and distribution schemes are examined in later chapters. A listing of alternatives for these subsystems is in Table I. The choice of which combination would make a "best" electrical power system would be revealed by trade off analysis. As an example of alternative combinations for trade off studies, generation/distribution schemes might be:

1. Dc transmission with dc source.
2. Dc transmission with ac source.
3. Ac transmission with ac source.
4. Ac transmission with dc source.

In the evaluation of possible combinations, several criteria should be considered. Some of the more important are:

1. Safety.
2. Reliability.
3. Maintainability.
4. Low cost.
5. Low weight.
6. Artificial gravity compatibility.

Safety criteria include the safety of the passengers and crew during flight and the safety of persons involved in preparation and launch. If nuclear fuel is used, safety should also include disposal of radioactive wastes and emergency procedures in case of accidents. Safety procedures depend to a large extent on a reliable electrical power system; the electrical system must be designed for rapid maintenance and repair so that reliability is not jeopardized.

Requirements for artificial gravity operations of Space Station/Base add operational difficulties for the electrical power system. Slip rings have been suggested in the event that large, flat-panel, solar cell arrays are chosen for the primary power source. Slip rings would permit the panel to maintain a sunward-facing orientation while the vehicle rotates. Slip rings would not be needed if rotation could be about an axis pointed toward the sun. One problem with solar arrays is that during artificial gravity the panels would have to be tethered in order to counteract centrifugal forces.

TABLE I

ALTERNATIVE ELECTRICAL POWER SUBSYSTEMS

<u>Energy Sources</u>	<u>Power Conversion</u>	<u>Conditioning Equipment</u>	<u>Distribution Methods</u>
Reactor	Batteries	Transformer/rectifiers	Hard wiring
Chemical	Fuel cells	dc/dc converters	Dedicated multiplex
Radioisotope	Thermoelectrics	Inverters	Integrated multiplex
Battery	Brayton cycle	Battery boosters	Electromechanical relays
Solar	Rankine cycle	Cycloconverters	Solid state switches
	Solar cells	Transformer	Centralized switching
	Chemical dynamic	Regulators	Distributed switching
	Mercury Rankine	Battery chargers	
	Organic Rankine	Load sharing devices	

In addition to the obvious limitations of cost and weight, several other constraints on the electrical system should be considered. Some of the more important of these are:

1. Voltage is limited by corona and arcing.
2. System lifetime should be compatible with mission requirements.
3. If a nuclear system is chosen, there should be an assured availability of fuel.
4. The electrical system of Space Shuttle should interface with the electrical system of Space Station/Base.
5. The Space Station electrical system should be capable of being integrated with the Space Base system.

Literature sources reveal divergence of opinions and conclusions concerning these constraints and their implications. Specific detailed approaches to the problems outlined above are discussed in later chapters.

C. Method of Approach

In the design of a spacecraft electrical system, the loads, distribution network, and power sources should all be considered as parts of the same system. It is not sufficient to analyze only the loads, the "best" generation method, the most efficient power conversion system (PCS), or the "optimal" distribution system without also considering the interrelated effects of all electrical subsystems. The approach chosen for this report is:

1. Analyze spacecraft loads to determine needed power consumption.
2. Describe power sources.
3. Examine some of the distribution systems which have been recommended.
4. Model power distribution designs which would be based on load requirements.
5. Evaluate source candidates for "best fit" to the distribution systems.

The next logical step (which is beyond the scope of this report) would be an analysis of the system as a whole, leading to adjustments in the subsystems to achieve a "best" total electrical power system. Such a study of trade offs for Space Shuttle and Space Station/Base

should perhaps be delayed until more definitive load analysis data and subsystem characteristics are available.

The load analyses in this study are not based on definitive data because information in the form required was sketchy. Much extrapolation has been necessary. The load analysis for the Space Station/Base was dependent primarily on a thorough examination of the environment control and life support (EC/LS) system of Space Station [4]. Since the EC/LS system is one of the largest users of electrical power it provides a useful starting point. The data resulting from the study of EC/LS components, when combined with an analysis of the other subsystems, provide the basis for load type categorization. The analysis of Space Shuttle loads was based on preliminary estimates for the Shuttle and also on data available on electrical systems for large aircraft. The Shuttle's electrical power requirements can be compared to those of a large aircraft to some extent because both the orbiter and the booster will operate like aircraft during their re-entry and landing operations.

II. LOAD PARAMETERS

A. Introduction

In two recent, major studies of Space Station electrical power requirements, one report [5] suggested a power distribution system supplying 86 percent dc and 14 percent ac; the other [6] suggested 90 percent ac and 10 percent dc. Some loads can be powered by either ac or dc; e.g., incandescent lamps and heaters. Other loads, such as motors and fluorescent lights, use unregulated ac power advantageously. Most electronic and instrumentation circuits use dc power and require close regulation of voltage. Ac powered instrumentation may require regulation of both frequency and voltage. The amount and quality of ac and dc power needed by electrical loads should be a determinant in trade off analyses used to select the generation and distribution subsystems. If some of the raw power can be used directly, the costs and weights of regulation and conditioning equipment are reduced. Conversely, generation of power of one type for loads that require power of another type will necessitate power conditioning units.

It is not sufficient to choose generation and distribution subsystems on the basis of load analysis alone. The criteria and constraints listed in Chapter I, together with other criteria and constraints which may be found necessary, should be applied to the entire electrical power system. This implies that a "best" system might well incorporate a subsystem which would not be rated "best" if it were evaluated without regard to its fit within the total system.

Most models of spacecraft electrical power systems have been based on trade off analyses of subsystems. The tacit assumption has seemed to be that if the voltage and frequency of an "optimum" power generator do not match load requirements, power conditioning equipment can be employed to convert the generated power to the types needed by the loads. In the present study, another approach is taken. First, the load requirements are analyzed, then distribution systems are designed to fit the loads, and finally generation schemes are evaluated

relative to the distribution requirements. This method is thought to be feasible because there are only two major power source contenders. The first, isotope/Brayton, generates ac; and the second, solar array/battery, produces dc. Source trade off studies have indicated that the choice between them is very close.

The remainder of this chapter is devoted to evaluating the Space Shuttle and Space Station/Base electrical power load requirements. No exact categorization by type of load can be given because of lack of published data, and because either ac or dc can be used by some loads. What can be determined are ranges of load requirements. Though imprecise, these estimates provide a logical basis for evaluating the electrical power sources and distribution systems. Further load categorization analysis would be needed to define the load requirements of Space Shuttle and Space Station/Base more exactly. Such a study could provide information for overall system trade off studies.

B. Space Station/Base

Any categorization of the electrical load requirements of Space Station/Base can only be considered to be preliminary. Not only are the forecasts of total load requirements subject to change, but the various constituents of the total load may change in relative magnitudes. However, the methods used for classifying loads should remain valid even if the values themselves change.

Tables II [5] and III [7] present electrical load breakdowns for Space Station by functions, as estimated by McDonnell Douglas and North American Rockwell, respectively. The two tables show that the breakdowns by functions are reasonably close in the two studies (differing only in allotments for experiments) although as pointed out above they differ widely in their recommendations for division of power into ac and dc distributions. Table IV [5] is a further breakdown of the functional power requirements listed in Table II. The functional loads are divided into categories of ac or dc, and into various voltage and frequency levels, but Table IV does not indicate that some loads may be operated either by ac or dc power.

TABLE II

DISTRIBUTION OF SPACE STATION LOAD POWER BY CRITICALITY CATEGORIES
12 MAN STATION -- ZERO-G -- NORMAL OPERATIONS [5]

Subsystem	Average Load Power			
	Emergency (watts)	Essential (watts)	Nonessential (watts)	Total (watts)
Environment control and life support (EC/LS)	475	9,286	7	9,768
Stability-attitude control and guidance-navigation (S/AC,G/N)	-	979	240	1,037
Propulsion and reaction control system (P/RCS)	-	520	275	795
Communications	21	-	225	246
Data management	220	1,952	490	2,662
Information display and control (I/D and C)	550	2,045	330	2,925
Onboard checkout system (OCS)	110	501	256	867
Crew	158	482	15	655
Lighting	565	1,142	553	2,260
Logistics	-	-	-	-
Experiments	-	-	712	712
Total	2,099	16,725	3,103	21,927

TABLE III
SPACE STATION ELECTRICAL POWER SYSTEM [7]

<u>Subsystem/Equipment</u>	<u>Load Power</u>		
	<u>Average</u> (watts)	<u>Backup</u> (watts)	<u>Emergency</u> (watts)
CO ₂ management	5604	5604	0
Atmospheric storage	700	110	0
Atmosphere control	3464	3464	1610
Thermal loop	500	500	350
Water management	939	939	0
Waste management	108	108	26
Hygiene	86	86	0
Food management	950	950	0
Communication/tracking	1810	721	310
Data management	977	501	255
Display and control	1155	650	350
	470	470	0
Navigation	139	128	98
	161	161	158
Reaction control system	7	7	7
Lighting	3600	1500	500
Experiments	6000	800	0
Total	26,670	16,699	3,664

TABLE IV

SPACE STATION LOAD POWER REQUIREMENTS
BY POWER FORM AND OPERATING VOLTAGE
12 MAN STATION [5]

Subsystem ⁽⁶⁾	Unregulated 22-31 Vdc ⁽¹⁾ (watts)	Unregulated 236-263 Vdc (watts)	Regulated 25-29 Vdc (watts)	Square Wave ⁽⁷⁾ 108-118 Vac 394-404 Hz (watts)	Sine Wave ⁽⁷⁾ 108-118 Vac 394-404 Hz (watts)
EC/LS	2,687	4,039	189	2,808 (3Φ)	45
S/AC and G/N	-	-	1,037	-	-
P/RCS	-	500	245	50 (1Φ)	-
Communications	-	50	196	-	-
Data management	-	-	2,662	-	-
I/D&C	2,050	-	800	-	75
OCS	-	-	711	-	76
Crew	240	-	8	340 (1Φ)	67
Lighting	2,210	-	-	-	50
Logistics	-	-	-	-	-
Experiments	-	-	712 ⁽³⁾⁽⁴⁾	-	-
Total ⁽⁵⁾	7,187	4,589	6,640	3,198	313

1. The values given in column headings are limits of steady state operating voltage at loads.
2. Thermal load.
3. Average power available for Zero-G experiments.
4. NASA Experiment Spec imposes limits of 24-32 Vdc at the load.
5. Distribution losses not included.
6. Abbreviations same as Table II.
7. Ac voltages are line to grounded neutral.

In order to categorize needed power by types, the constituents of the functional divisions in Table III have been examined in detail; the resulting reclassification is shown in Table V. It is assumed that motors and fluorescent lighting require ac, electronics and instrumentation require dc, but heating can utilize either ac or dc. In the absence of better definition, experiments are assumed to be loads that require 50 percent ac and 50 percent dc. This leads to the conclusion that load requirements for Space Station are:

- 36 percent ac power,
- 42 percent dc power, and
- 22 percent either ac or dc power.

It appears that the power distribution for Space Station loads could range from 36 to 58 percent ac and from 42 to 64 percent dc. If the load requirements of experiments do not divide equally between ac and dc, the range estimates would change accordingly.

In choosing an electrical generation/distribution system for Space Station, the power requirements of Space Base should also be considered, since Space Station may ultimately be enlarged into Space Base. Table VI [5,8] gives a comparative analysis by functions of the power requirements of Space Station and Space Base. For some functions the power requirements remain unchanged, while for others there are large increases in required power.

The methods used to classify power requirements for Space Station can be extended to Space Base. Because of incomplete information, the ratios of power requirements for motors, heaters, etc., in the EC/LS system of the Space Base are assumed to be the same as for Space Station, with extrapolations on a per man basis. The results of this analysis are found in Table VII. Making the same assumptions about classification as with the Space Station leads to the following breakdown for Space Base:

- 37 percent ac power,
- 44 percent dc power, and
- 19 percent either ac or dc power.

TABLE V
SPACE STATION LOAD ANALYSIS*

<u>Load</u>	<u>Average Power Required (kilowatts)</u>	<u>Type</u>
Motors	3.2	ac
Heating	6.0	ac or dc
Lighting	3.6	ac
Electronics and instrumentation	8.3	dc
Experiments	3.0	ac
Experiments	3.0	dc
Total	27.1	

*Based on load distribution given in Table II, but with 6 kW instead of 712 watts for experiments.

TABLE VI
SPACE STATION/BASE ELECTRICAL LOAD REQUIREMENTS [5,8]

Subsystem ⁽³⁾	Average Power	
	Station (kilowatts)	Base (kilowatts)
EC/LS	9.8	33.5 ⁽¹⁾
S/AC and G/N	1.0	1.0
P/RCS	0.8	3.5
Communications	0.2	1.2
Data management	2.7	2.7
I/D&C	2.9	2.2
OCS	0.9	0.8
Crew habitability	0.7	2.5
Lighting	2.3	6.6
Logistics		1.0
Subtotal at loads	21.3	55.0
Distribution losses at 4 percent		2.2
Total for Station operation		57.2 ⁽⁴⁾
Contingency (10 percent)		5.7
Available for User Facilities (average)	6.0	37.1 ⁽²⁾
Total at load buses	27.2	100.0
Peak load estimate	50.0	180.0

1. Includes 6.2 kW thermal power.
2. Includes distribution losses. The estimate of power required for user facilities is 40 kWe average.
3. Abbreviations same as Table II.
4. Emergency power (excluding user power requirements for Base:

1 hour	10.6 kWe
1-24 hours*	23.5 kWe
1-8 days**	25.9 kWe
Sustained emergency***	57.5 kWe

*After 1 hour, turn on essential loads in all subsystems except EC/LS.

**After 24 hours, turn on EC/LS CO₂ control and one condensate recovery system.

***After 8 days, turn on remainder of EC/LS loads.

TABLE VII
SPACE BASE LOAD ANALYSIS

<u>Load</u>	<u>Average Power Required (kilowatts)</u>	<u>Power Type</u>
Motors	10.1	ac
Heating	18.8	ac or dc
Lighting	7.2	ac
Electronics and instrumentation	23.7	dc
Experiments	20.1	ac
Experiments	20.1	dc
Total	100.0	

The estimates and the ranges of power requirements by types are close to those for Space Station. The same power distribution design approaches can, to the degree that they are based on load requirements for ac or dc power, serve the needs of both Space Station and Space Base. The divisions of power requirements for the experiments are again assumed to be equally divided between ac and dc.

C. Space Shuttle

As presently envisioned, the Space Shuttle will consist of two separate manned vehicles. The booster will be used to "launch" the orbiter and will have a flight time of about three hours for each mission. The orbiter will have a mission duration of approximately one week. Figure 1 illustrates a nominal mission profile for Space Shuttle [9].

Because the booster and orbiter have such varied electrical power requirements throughout the duration of their missions, the loads cannot be characterized as easily as those of the Space Station/Base. During the initial part of their missions, the booster and orbiter operate as spacecraft; during their entry and landing phases, they function as aircraft. The similarities between Space Shuttle and present commercial aircraft warrant comparisons of their electrical power needs. The electrical load profile of a large subsonic aircraft [10] was presented in the Interim Report [1]. Evaluation of the aircraft load profile data permits a breakdown by function, as shown in Table VIII. Note that the lights, which are predominantly incandescent landing lights, have been designated either ac or dc. The classification implies a load usage for aircraft of:

- 55 percent ac power,
- 7 percent dc power, and
- 38 percent either ac or dc power.

Based on the similarities between the Space Shuttle vehicles and large commercial aircraft, approximately the same load usage should apply to the booster and to the orbiter when they are operated in the aircraft mode.

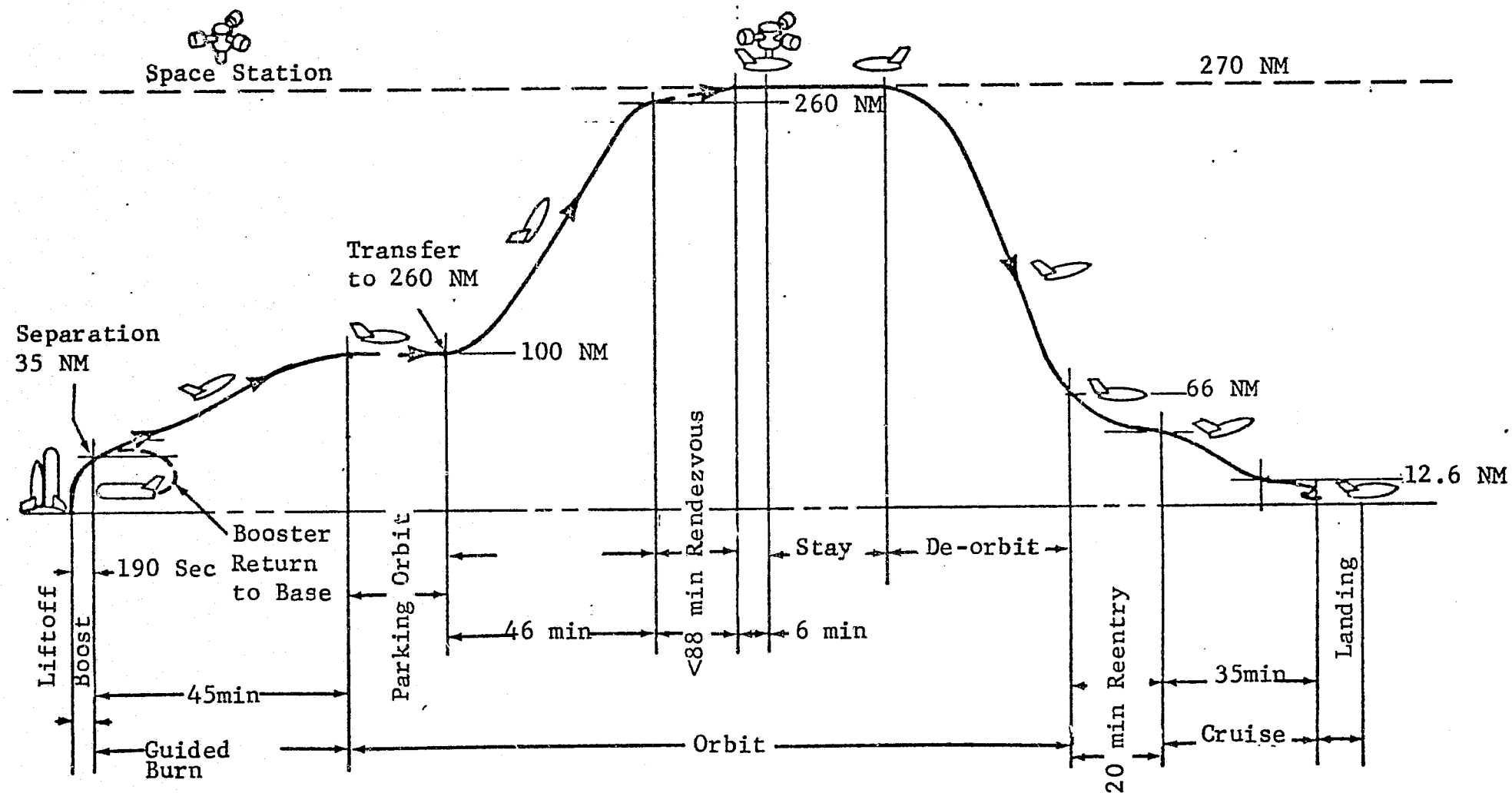


Figure 1. Nominal Mission Profile. From Digesu [9].

TABLE VIII
AIRCRAFT LOAD ANALYSIS [10]

<u>Load</u>	<u>Average Power Required (percent)</u>	<u>Power Type</u>
Motors	55	ac
Heating	25	ac or dc
Lighting	13	ac or dc
Electronics	5	dc
Relays and instruments	2	dc

One proposed configuration for the booster supports the extrapolation of aircraft estimates to Space Shuttle [11]. Estimates of booster generated power, including power for transformer rectifier units which supply dc loads, are shown in Figure 2. As would be expected, large peaks of electrical power are required for deploying and starting engines. Table IX gives a breakdown by subsystem of the booster electrical load, exclusive of transformer/rectifier requirements. From Table IX it can be seen that the peak requirement for the dc electrical power system occurs during ascent, whereas the peak electrical power requirement for the ac system occurs during the cruise and landing phase of the mission (the aircraft mode of operation).

During the cruise and landing phase of booster flight, power requirements as shown in Table IX are:

65 percent ac power, and
35 percent dc power.

During the ascent phase, power requirements for the booster are:

32 percent ac power, and
68 percent dc power.

The large peak demands for power (including intermittent hydraulic power), the short flight duration, and the dual mode of propulsion strongly influence the design of Space Shuttle electrical power systems. Choices of power sources and designs of distribution systems are described in Chapters III and IV.

The electrical power system for the orbiter is necessarily more complex than the booster because of differences in missions. The orbiter has a mission life of seven days of which five days is spent docked at the Space Station; but some power will be required even while the orbiter is docked. Figures 3 and 4 present projections of ac and dc power requirements of the orbiter as a function of mission elapsed time [12]. The ratios of peak power to average power during the mission are even larger for the orbiter than for the booster.

Analysis of the orbiter power profiles allows a determination of the average ac and dc power needed during the mission. The profile

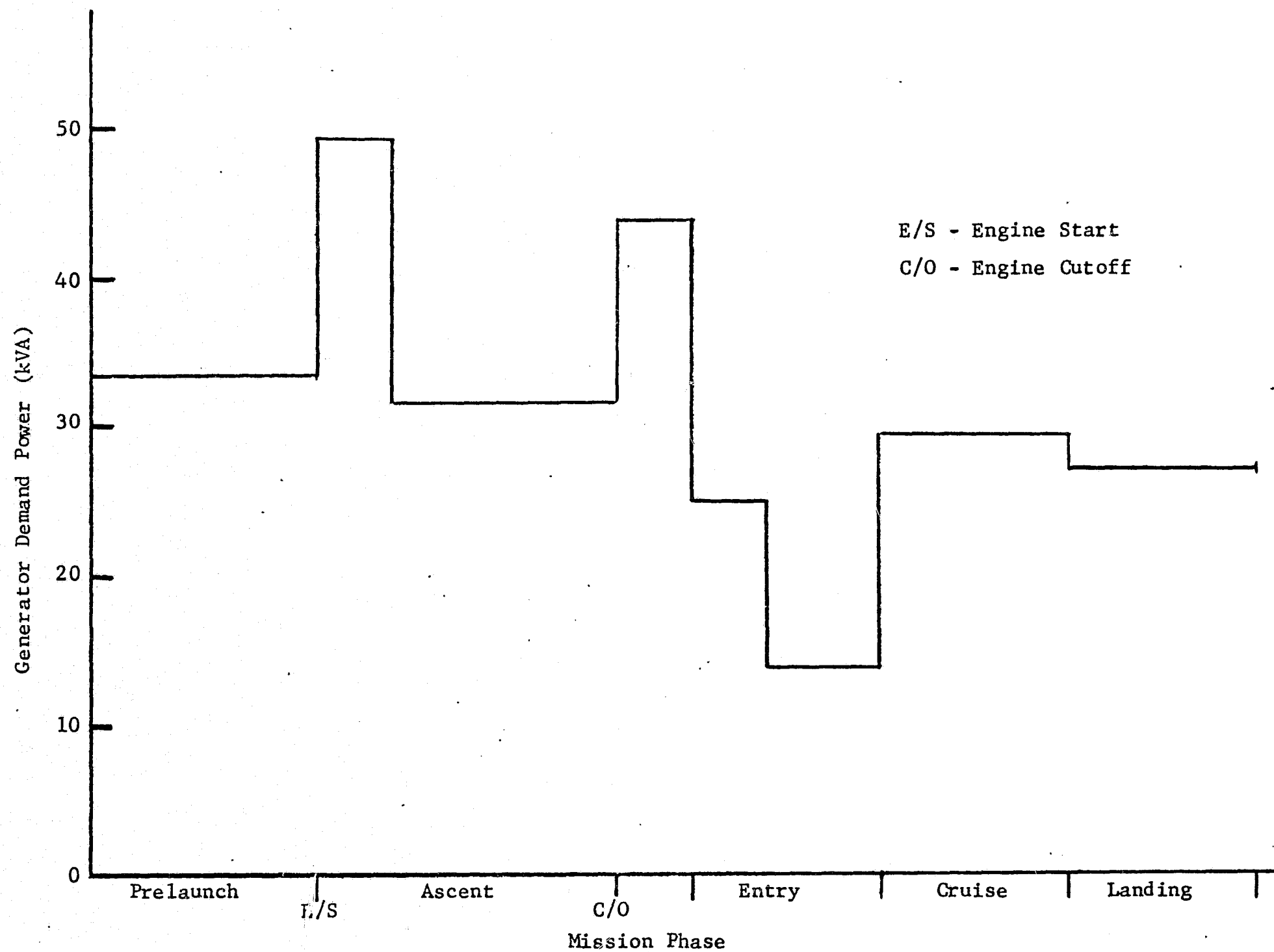


Figure 2. Booster Demand Power. From Walter [11].

TABLE IX

BOOSTER ELECTRICAL POWER REQUIREMENTS [11]

Subsystems	Power Required					
	Dc Power			Ac Power		
	Ascent (watts)	Cruise (watts)	Landing (watts)	Ascent (watts)	Cruise (watts)	Landing (watts)
Communications	290	190	190	-	356	356
Data and control mgt.	1797	1797	1797	-	-	-
Display and control	2155	2190	2190	-	-	-
Air breathing propulsion	155	155	155	30	15,150	15,150
Attitude control	1723	1175	1175	60	60	60
Main propulsion	8987	25	25	7700	-	-
Hydraulic	1008	1008	672	-	-	-
Auxiliary power unit	1800	1800	1800	-	-	-
Guidance, navigation and control	668	652	676	248	493	532
Lighting	-	-	-	230	630	630
EC/LS	-	-	-	320	320	320
Total	18,580	8,992	8,680	8,588	17,009	17,048

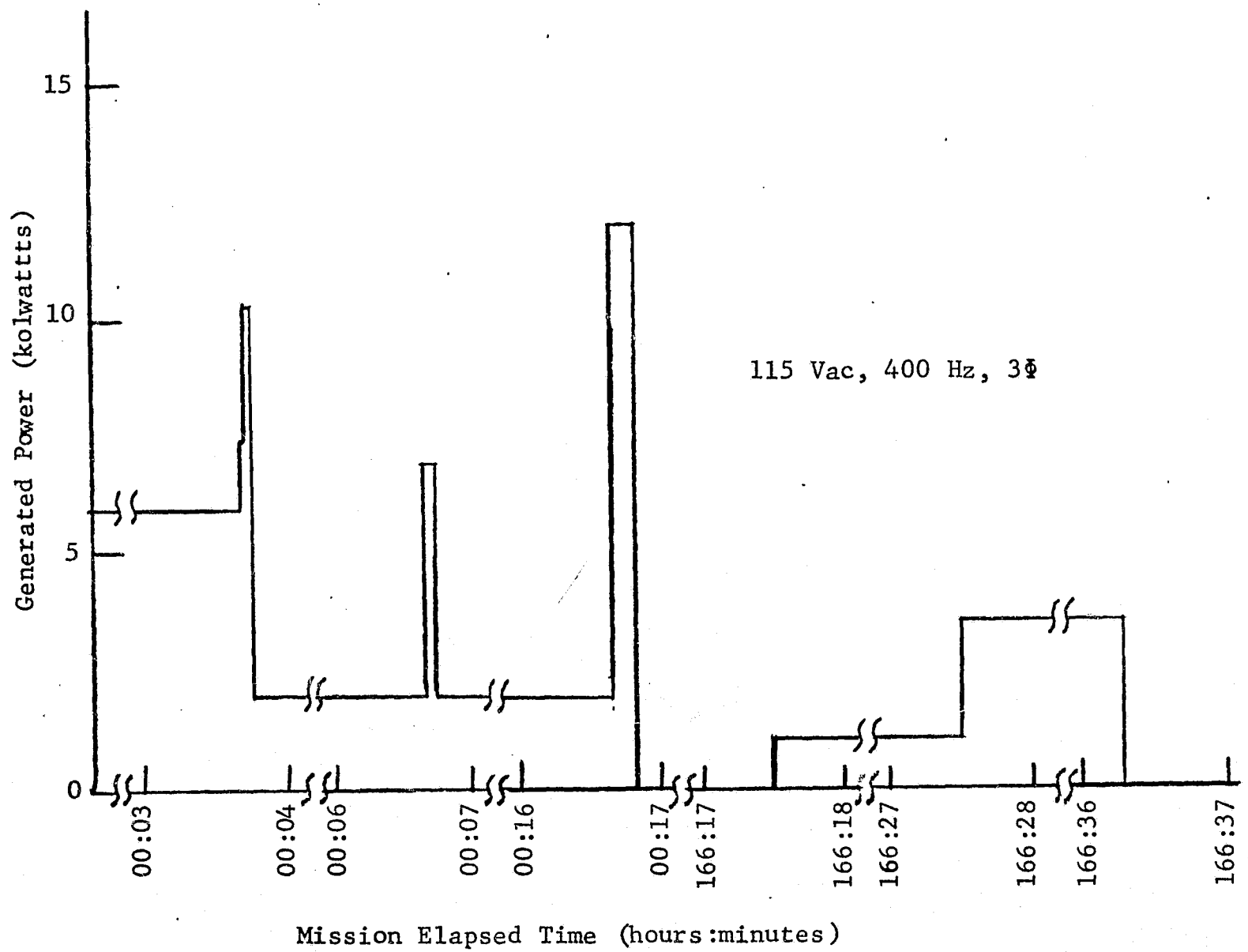


Figure 3. Space Shuttle Orbiter Power Profile
Alternator Subsystem. From MDAC [14].

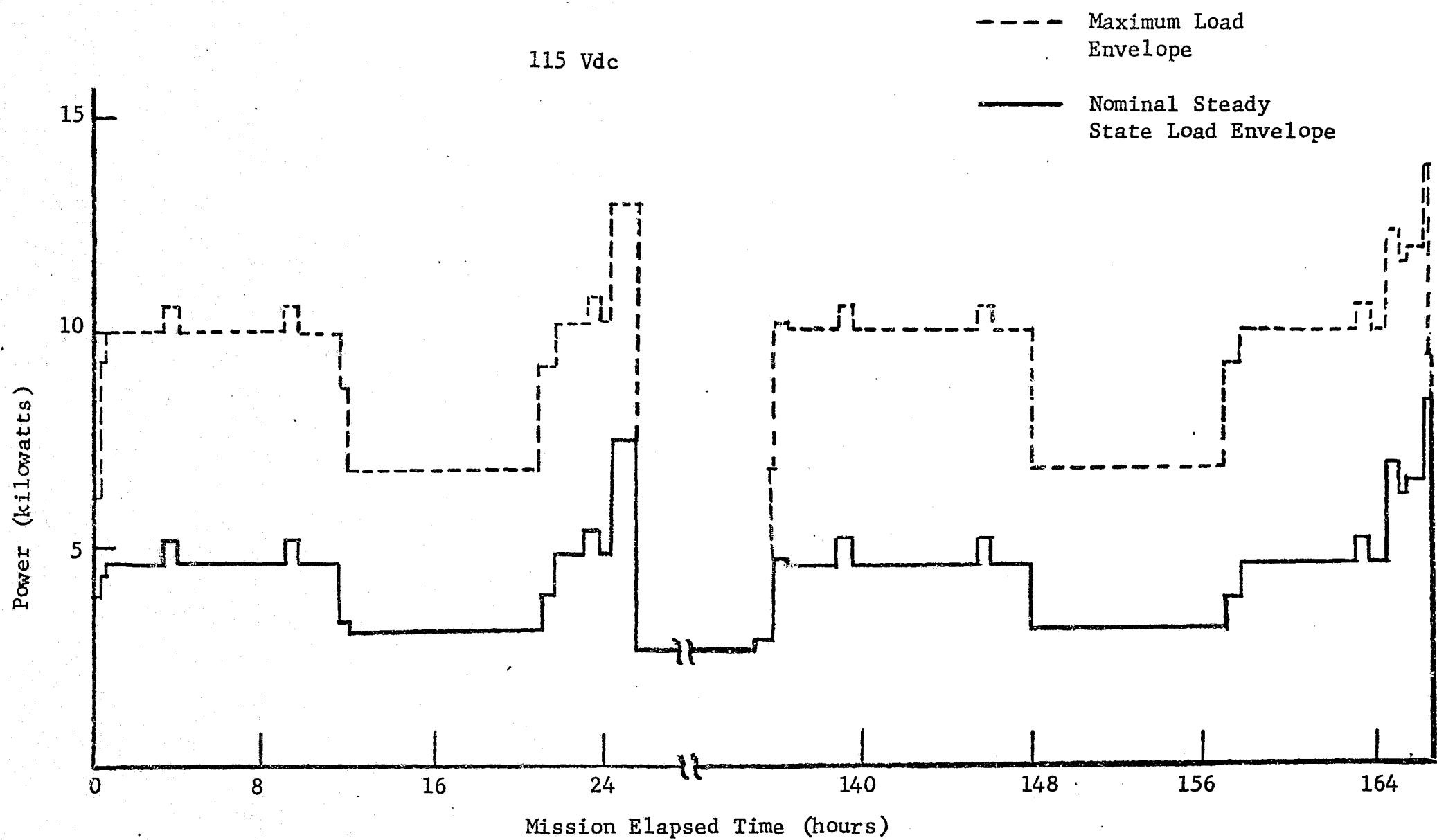


Figure 4. Space Shuttle Orbiter Power Profile, Fuel Cell Subsystem.
From MDAC [14].

in Figure 3 represents average ac power required. Note that ac power is required for only 37 minutes total during the seven-day mission. The breakdown shows further that ac requirements average 2.4 kW, and dc requirements average 3.1 kW. Based on these figures, the breakdown for the Space Shuttle orbiter is:

44 percent ac power, and
56 percent dc power.

There is certainly some power which could be either ac or dc, but the amount cannot be determined from present data.

The foregoing electrical power estimates for booster and orbiter are preliminary. Also, there is not enough information available on the Space Shuttle to give as complete a breakdown on power usage as can be done for Space Station/Base. It may be that present requirements will change substantially as a result of further investigation. In addition, there is a possibility that the entire operating concept will be revamped; that would necessitate a reevaluation of the electrical power needs.

D. Summary

The purpose of this section on load analysis was to establish the types and amounts of power required by the Space Shuttle (booster and orbiter) and by the Space Station/Base. From study of the electrical power requirements of the subsystems it was possible to divide the needed power into three categories -- ac, dc, and either ac or dc. It has also been possible to categorize some of the loads as essential, non-essential, and emergency. A summary of these analyses is given in Table X. Having the power requirements divided in such manner allows a rational approach to the design of the generation and distribution systems.

In general, higher voltage ac power permits weight savings in motors, relays, solenoids, and electromechanical actuators. A frequency of 400 Hz would permit the use of many off-the-shelf components. Fluorescent lights, which are more efficient than incandescent lamps, also benefit from higher frequency and higher voltage. Heaters can utilize either ac or dc power, and higher operating voltage leads to weight savings. Even the wire used for distribution of power weighs less in high voltage systems, and the inevitable cable voltage drop is smaller.

TABLE X
POWER USAGE CATEGORIZATION SUMMARY

<u>Category</u>	<u>Space Station</u> (percent)	<u>Space Base</u> (percent)	<u>Space Shuttle</u>	
			<u>Booster</u> (percent)	<u>Orbiter</u> (percent)
Ac	36	37	48	44
Dc	42	44	52	56
Ac or dc	22	19	-	-
Total	100	100	100	100
Essential*	62	48	-	-
Nonessential*	30	46	-	-
Emergency*	8	6	-	-
Total	100	100		

*Altering Table II by assuming 6 kW for experiments; and extrapolating Base requirements from Table VI.

Because most of the dc loads require many different values of voltage for end usage, many components for power conditioning will be needed, and any advantage of dc distribution is thereby weakened. A trade off study would compare weights of dc to dc converters with transformer rectifiers. Secondary penalty factors such as circuit isolation effectiveness should be included. The study should also take cognizance that dc distribution would necessitate inversion to ac power for motors, whereas ac distribution would allow the utilization of power by motors without conversion.

III. POWER SOURCES

A. Introduction

Many electrical power generation systems have been considered for future space application. The more promising candidates are listed in Table I, Chapter II. The relative merits of alternative power sources for Space Shuttle, Space Station, and Space Base missions are discussed in this chapter. Many are still in the development stage. Some of the critical parameters that affect the selection of a power source for a particular mission are:

1. Reliability.
2. Safety
3. Commonality.
4. Life.
5. Efficiency.
6. Development cost.
7. Weight.
8. Volume.
9. Availability.
10. Maintainability.
11. Artificial gravity compatibility.

B. Space Station Electrical Power Source

Two extensive studies, one by North American Rockwell [4,6,7], the other by McDonnell Douglas Astronautics Co., [5,8,12,13,14] have proposed electrical power sources for Space Station. North American Rockwell (NAR) has proposed a solar array/battery source for primary power, with either batteries or fuel cells for emergency power. McDonnell Douglas Astronautics Co., (MDAC) has proposed a radioisotope/Brayton system for primary power with batteries for emergency power.

1. A Solar Array/Battery Power Source

A proposed solar array/battery (SA/B) source is shown in Figure 5. It consists of four solar arrays, each of which has an area of 2500 square feet. Beginning of life (BOL) rating for each panel is 22.8 kW at 70°C. In the NAR system, each array supplies 112 Vdc power

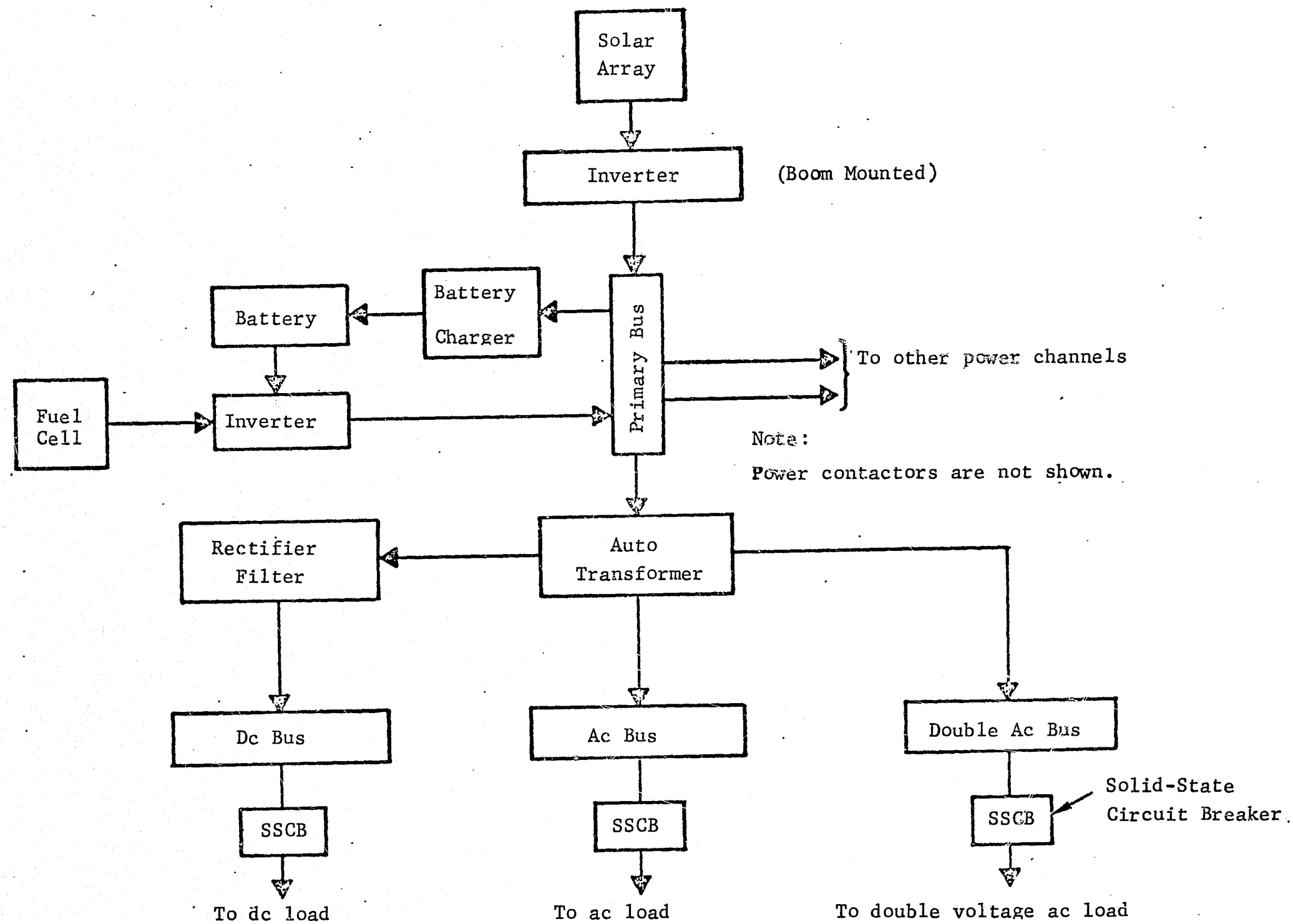


Figure 5. Solar Array/Battery Power Distribution System. From NAR [7].

to an inverter regulator whose output is 416 Vac, 400 Hz, 3 Φ . The outputs of the inverters are regulated to close frequency and voltage tolerance, so that the ac outputs of the four inverters operate in parallel for load sharing. The inverter/regulator outputs are distributed by a primary bus throughout the spacecraft to auto transformers which transform to 208 Vac for secondary bus distribution. Tertiary windings on the auto-transformers supply low voltage to rectifier/filters which furnish 56 Vdc for the dc loads.

The selection of this system for the Space Station was based on a number of factors. Solar array technology is well advanced, having been used on previous manned and unmanned space systems. Solar arrays deteriorate due to micrometeoroid collisions, radiation damage, cell failures, and misorientation. As shown in Figure 6, it is expected that the output power of the array will decrease to 64 percent of BOL after 5 years. The arrays will need to be replaced once or twice during a 10 year space mission. A summary of the solar array design parameters is given in Table XI.

There are several significant factors regarding the use of a solar array/battery source. It is characterized by large power output per unit weight, approximately twenty watts per pound; but the solar array can be deployed to only half its maximum length during the artificial gravity phase of the mission. Restricted deployment will reduce the available power by about one half. Another important factor is the requirement for an auxiliary source to supply power during the dark part of the orbit. Batteries are usually chosen for the auxiliary power source because they can be charged during the daylight and then used as a power source during the dark period. (Worst case dark time is 38 percent of the orbit.) In the NAR system, however, the batteries are also used during daylight hours to supply peak power. This permits a smaller solar array, because it must supply only average power rather than peak power during daylight hours. The design philosophy is illustrated for a typical 24 hour load profile in Figure 7 which shows that peak power periods exist during the crew day. Nickel cadmium batteries have been recommended for the auxiliary source because they

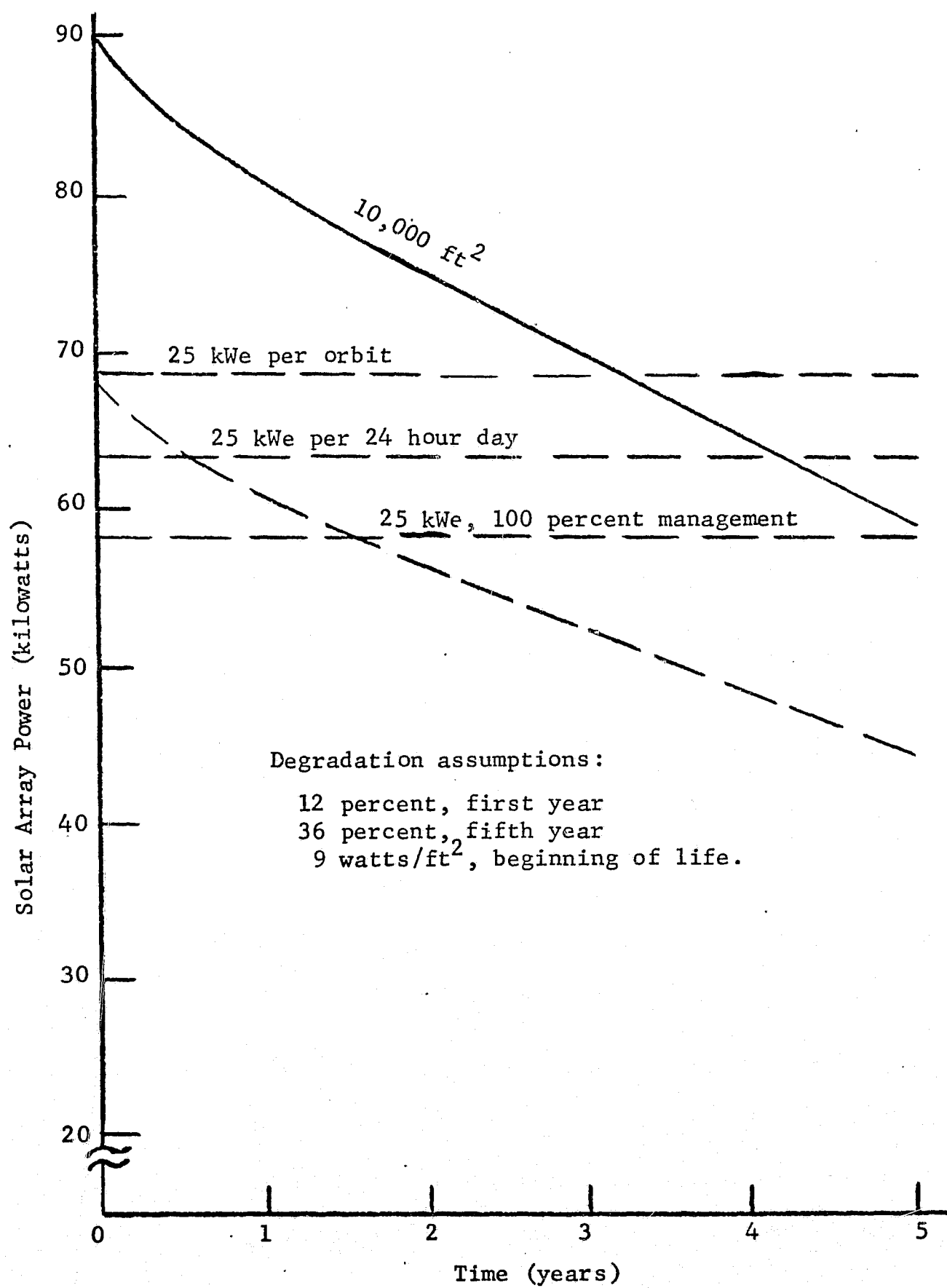


Figure 6. Solar Array Degradation Characteristics.
From NAR [7].

TABLE XI
SPACE STATION SOLAR ARRAY DESIGN PARAMETERS [7]

Nominal array unit size (four arrays required)	2,500 ft ²
Total solar cell module area	2,489 ft ²
Maximum array power	22.8 kW*
Maximum power voltage	126 Vdc*
Number of 2 x 6 cm cells	184,800
Total solar cell area	2,388 ft ²
Number of series cells/circuit	330
Number of parallel 2 x 6 cm cells/circuit	14
Fully deployed array length	96.3 ft
Total array width	31.1 ft

*Beginning of life (BOL), 70°C, 15° misorientation.

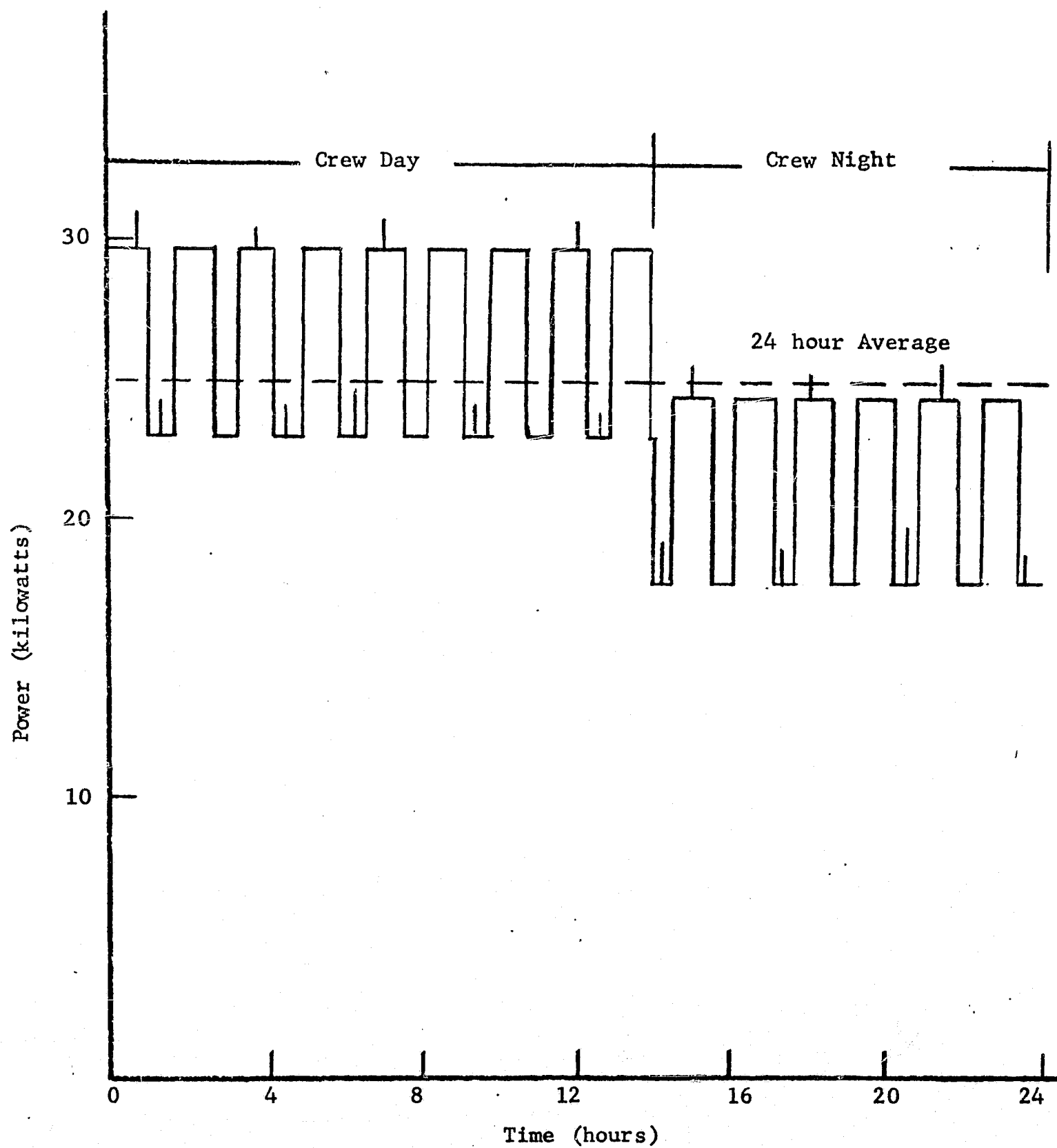


Figure 7. Space Station Typical Load Profile.
From NAR [6].

can survive a large number of charge/discharge cycles (25,000 to 30,000) and they have established good performance records in past programs.

Curves of array weight versus distribution power loss are shown in Figure 8. One array consists of 14 parallel cells per circuit, 330 series cells per circuit, and 40 circuits. This produces a total array voltage of 112 Vdc, and both polarities can be furnished with a three wire system. Figure 8 shows that 330 series cells yield a lower bus weight than 165 cells in series, and on this basis the array of 330 series cells was selected. Open circuit output is 126 Vdc.

In order to insure long life the auxiliary source batteries would be discharged to a maximum depth of 15 percent, and the batteries would be charged by high-amplitude pulses, with each battery cell monitored individually to insure proper charge. This could be controlled by the computer if data bus multiplex is employed to control load switching.

A weight penalty should be charged to the solar array/battery combination, because about 11,000 pounds of batteries are required. The total solar array/battery electrical power system weight breakdown is given in Table XII. Examination of this table shows that fuel cells have been included for emergency power. It is expected that the fuel cells would supply about 3 kW power and have an expected lifetime of 10 years because of their limited duty. The batteries cannot be relied upon at all times for full emergency power, because a catastrophic failure of the solar array might occur when the batteries are at their deepest discharge, e.g., at crew daybreak. In such case, the fuel cells would serve as a backup power source.

2. Isotope/Brayton

An isotope/Brayton electrical power system, which is like that recommended by MDAC, is shown in Figure 9. It consists of a radioisotope heat source, a heat exchanger, and power conversion equipment. A closed cycle gas loop drives a three-phase alternator, which generates 115/200 Vac, 1200 Hz, 3 ϕ . The rest of the thermal system includes the recuperator, compressor, and heat exchanger, which extract waste heat from the operating gas (helium/xenon).

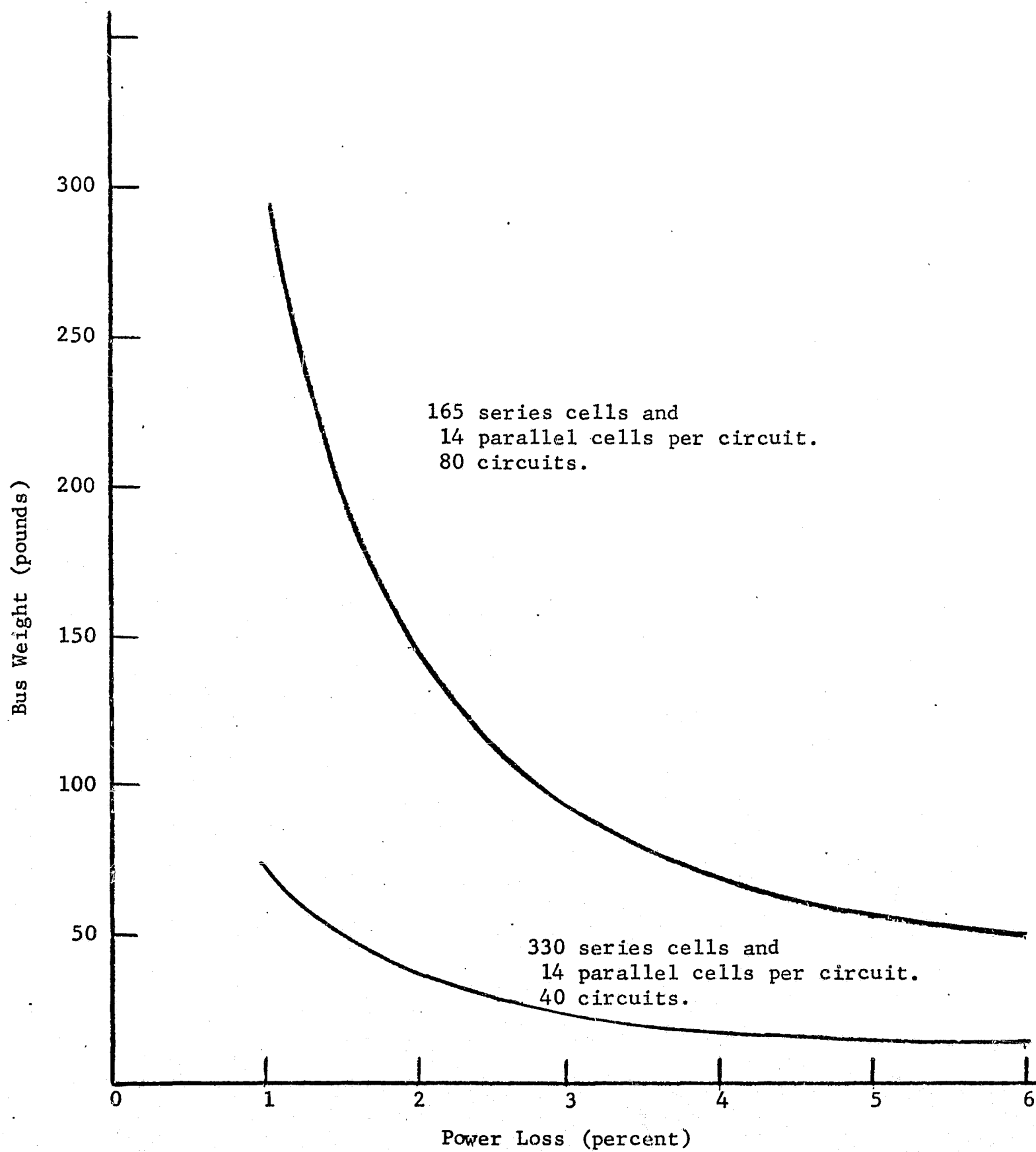


Figure 8. Solar Array/Battery Transmission Bus Weight Characteristics. From NAR [4].

TABLE XII

SPACE STATION SOLAR ARRAY/BATTERY ELECTRICAL POWER SYSTEMS
WEIGHT SUMMARY [7]

<u>Subsystems</u>	<u>Initial Launch Weight</u> (pounds)	<u>Orbit Weight</u> (pounds)
Solar Energy Conversion:		5,815
Solar cell rollup array	4,720	
Orientation mech (NR)	286	
Array support (boom)	374	
Mounts and supports	434	
Fuel Cells:		704
Fuel cells (4)	640	
Mounts and supports	64	
Energy Storage:		10,890*
Batteries (12)	4,950	
Mounts and supports	990	
Power Conditioning:		2,257
Chargers	320	
Controls	160	
Core Inverters	378	
Secondary bus	400	
Mounts and supports	168	
Boom Inverters	755	
Mounts and supports	76	
Distribution Control and Wiring:		4,051
Wiring	3,600	
Core Buses, contractors	58	
Mounts and supports	369	
Wiring	20	
Boom:Mounts and supports	4	
Lighting, including mounts	898	848
Total	19,665	24,615

*4,950 pounds of battery cells are supplied after initial launch and prior to 12-man station configuration.

$$\frac{12.5 \text{ kWe}}{48.5 \text{ kWt}} = 25.8\% \text{ Efficiency}$$

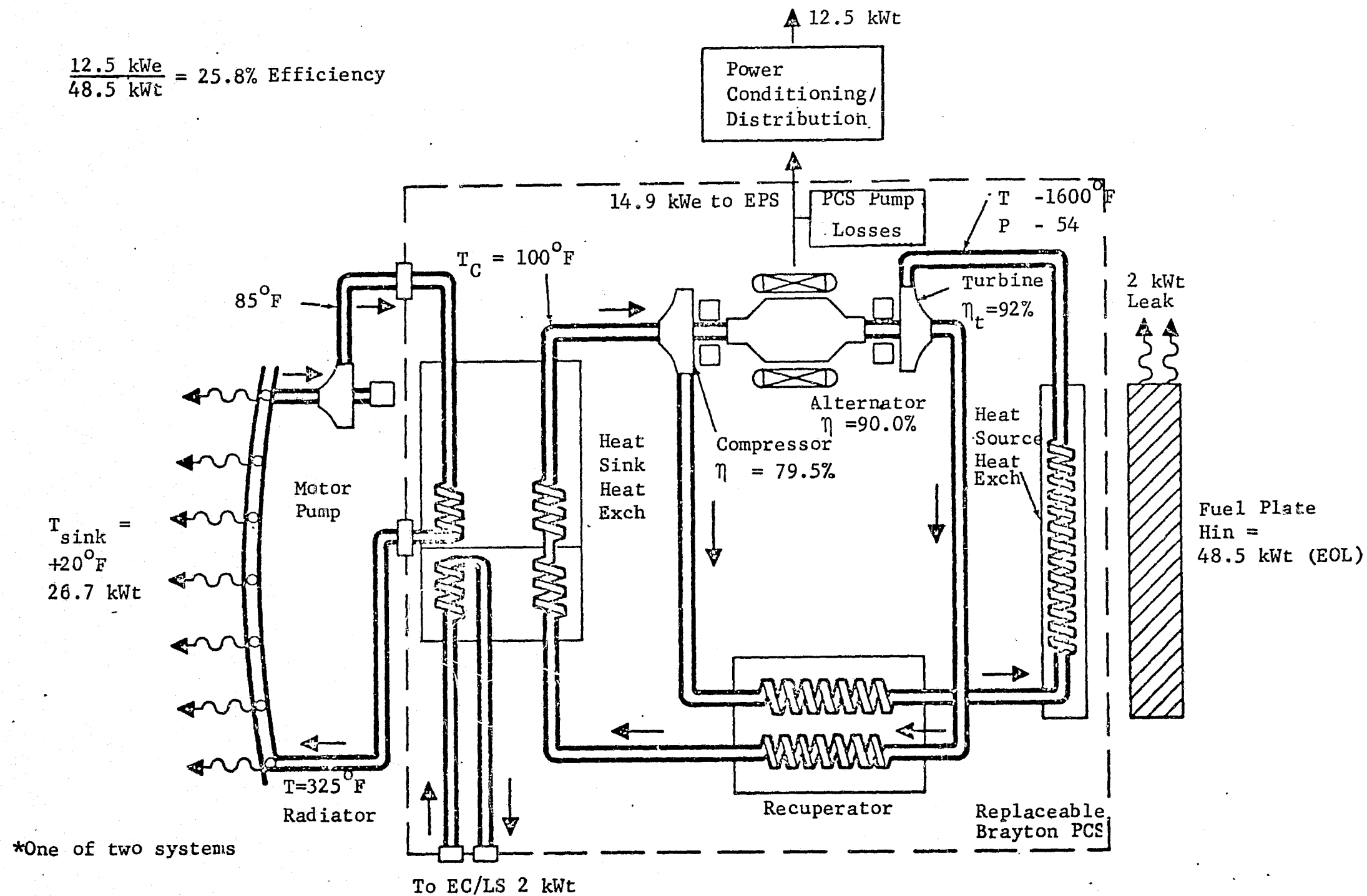


Figure 9. Isotope/Brayton System. From MDAC [13].

It is quite obvious that the problems associated with the isotope/Brayton system are quite different than those associated with the solar array power source. Operation of a radioisotope source has an inherent problem of safety to personnel because of radiation hazard. A radiation shield must be provided to limit the neutron and gamma radiation to less than 5 mrem/hr. The shield recommended for the system in Figure 9 consists of a thick layer of lithium hydride (8.5 inches) and a thin layer of stainless steel (0.25 inch). Shield weights for the power module vary with design; boom mounted shields are the lightest for a given radiation dose because radiation is omnidirectional and the boom increases separation between the source and the Station personnel. A boom mounted shield weighs 1210 lbs. for a radiation dose of 5.0 mrem/hr at a point 57 feet from the core module [13].

To make a valid comparison with the solar array/battery power source it is necessary to make the comparison on the basis of equal electrical power outputs. The following performance discussion will apply to an isotope heat source that will produce ≈ 25 kWe (kilowatts electrical) output, since this was the output of the solar array discussed in a previous section and is approximately the load requirement for the Space Station.

A typical power system thermal balance is shown in Figure 10 where it is seen that 56 kWt (kilowatts thermal) produces 13.65 kWe output. It would require two such isotope/Brayton modules to produce the required 25 kWe. The diagram shows that 41 kWt is waste heat that must be dissipated by the radiators. Some of this waste heat can be effectively utilized to heat the interior of the Space Station, and this must be considered when comparing it with the solar array/battery system. The performance data for an isotope/Brayton power system are given in Table XIII. Perhaps the outstanding characteristic of the isotope/Brayton is its high efficiency, but the 32 percent in Table XIII seems optimistic. About 27 percent is a more realistic efficiency rating. The isotope/Brayton system weighs 14,000 pounds, which is less than the solar array/battery system with its 11,000 pounds of batteries.

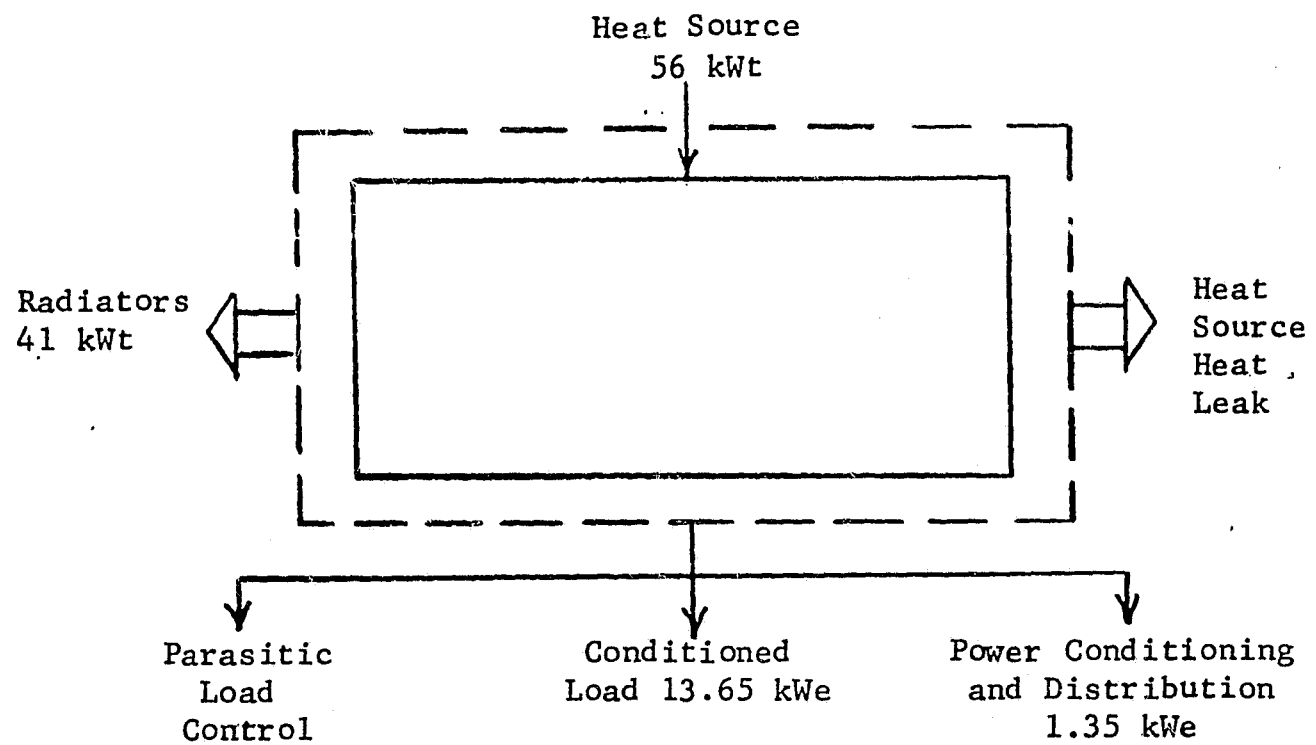


Figure 10. Power System Thermal Balance, Orbit Conditions. From NAR [6].

TABLE XIII

SPACE STATION ISOTOPE/BRAYTON SYSTEM CHARACTERISTICS [13]

Net conditioned power	25.0 kWe* at 1200 Hz
Conditioning and distribution loss	4.8 kWe
Net raw power	29.8 kWe
PCS module rating	14.9 kWe
Isotope heat source rating (2 at 48.5 kWt)	97.0 kWt end of life (EOL)**
-Cycle heat input	(46.5 kWt)
-Heat leakage	(2.0 kWt)
Power conversion efficiency (14.9 kWe/46.5 kWt)	32 percent
Radiator	1850 ft ²
Design sink temperature	+20°F
Power conversion system (PCS) working gas	Xe-He
PCS turbine inlet temperature	2060°R
PCS compressor inlet temperature	560°R
Isotope reentry vehicle diameter	87 inches present design, 92 inches final design projection
Crew dose rate***	25 rem/6 mo from Pu-238

*4.0 kWe bonus for experiments; waste heat provides 4.0 kWt to EC/LS.
 **Pu-238 age is 11.0 years, beginning of life (BOL) rating is 104 kWt.
 ***8/8/4/4 hour compartment residence schedule.

A potential difficulty with the isotope/Brayton power source is the possible limited supply of fuel (Pu 238). Figure 11 shows the projected fuel availability. Additional difficulties are the high fuel costs, \$540 per thermal watt [6], and methods of disposing of the power module at the end of life. Present planning calls for either launching the spent module into a separate orbit, or separating the isotope heat source as a reentry package to be reentered and recovered. Reorbit of the spent module is recommended at present because of the possible pollution hazard of a reentry operation. A further possible disadvantage of the isotope/Brayton power source is the limited life of the alternator. Although the alternator utilizes the heat transport fluid to cool the windings and lubricate the bearings, the expected life of an alternator is only two or three years. Thus, the power module will need to be replaced before the isotope heat source output has decreased enough to warrant its replacement.

The isotope/Brayton system was recommended by MDAC over the solar array power source mainly on the basis of projected growth of power needs and the requirement for integration with the Space Base power system.

C. Space Base Electrical Power Source

Space Base is an expanded version of Space Station and is scheduled for launch in the 1980 time period. A preferred Space Base configuration is shown in Figure 12 [8]. The crew complement of Space Base is thirty to sixty men, as compared to six to twelve men for the Space Station. Consequently the electrical power requirements and power source configuration for Space Base are both larger than and different from Space Station. The candidate systems that have been considered for possible use in Space Base are:

1. Thermoelectric.
2. Mercury/Rankine (reactor).
3. Reactor/Brayton.
4. Organic/Rankine (reactor).
5. Isotope/Brayton.

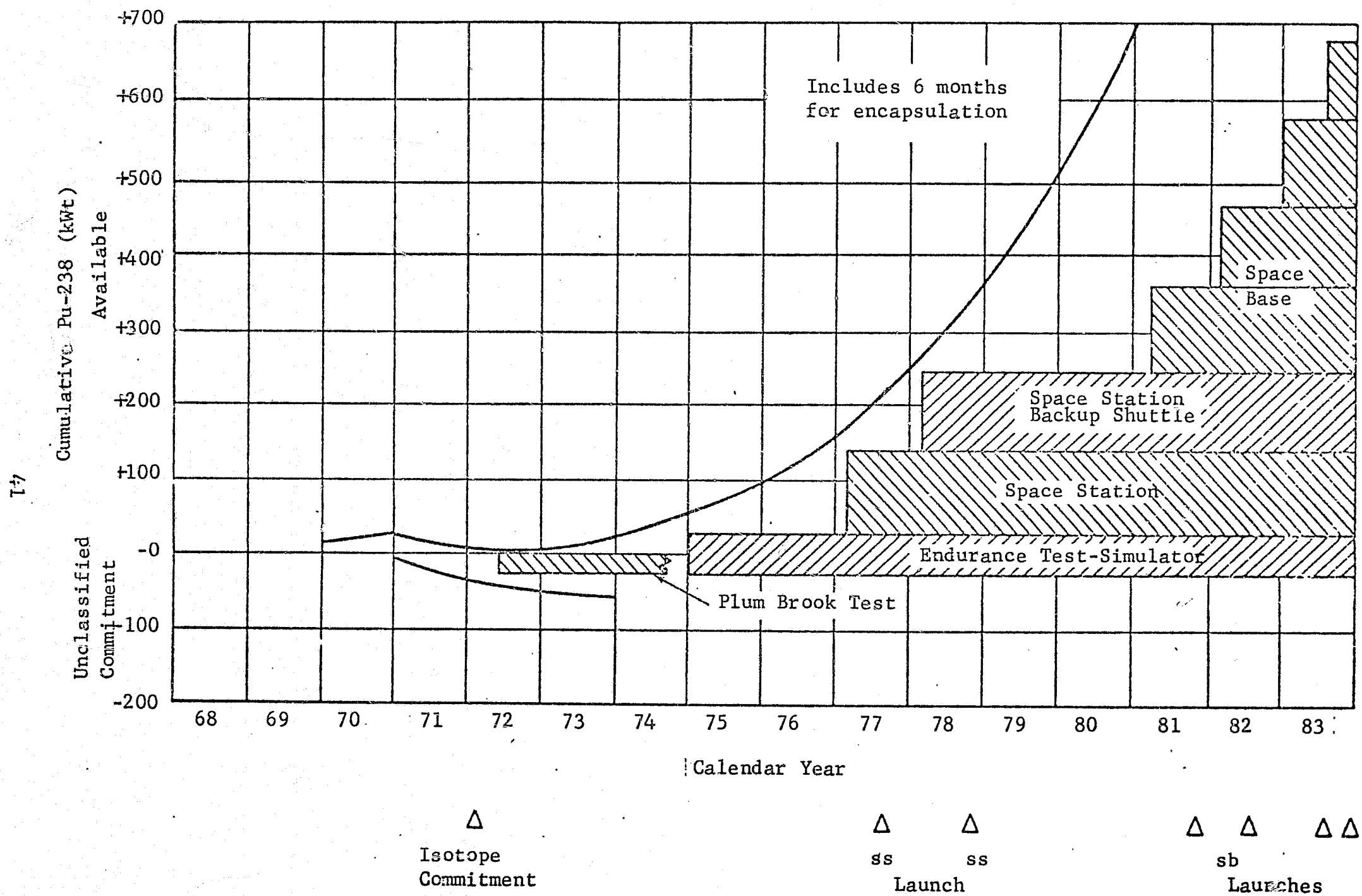


Figure 11. Pu-238 Availability. From MDAC [5].

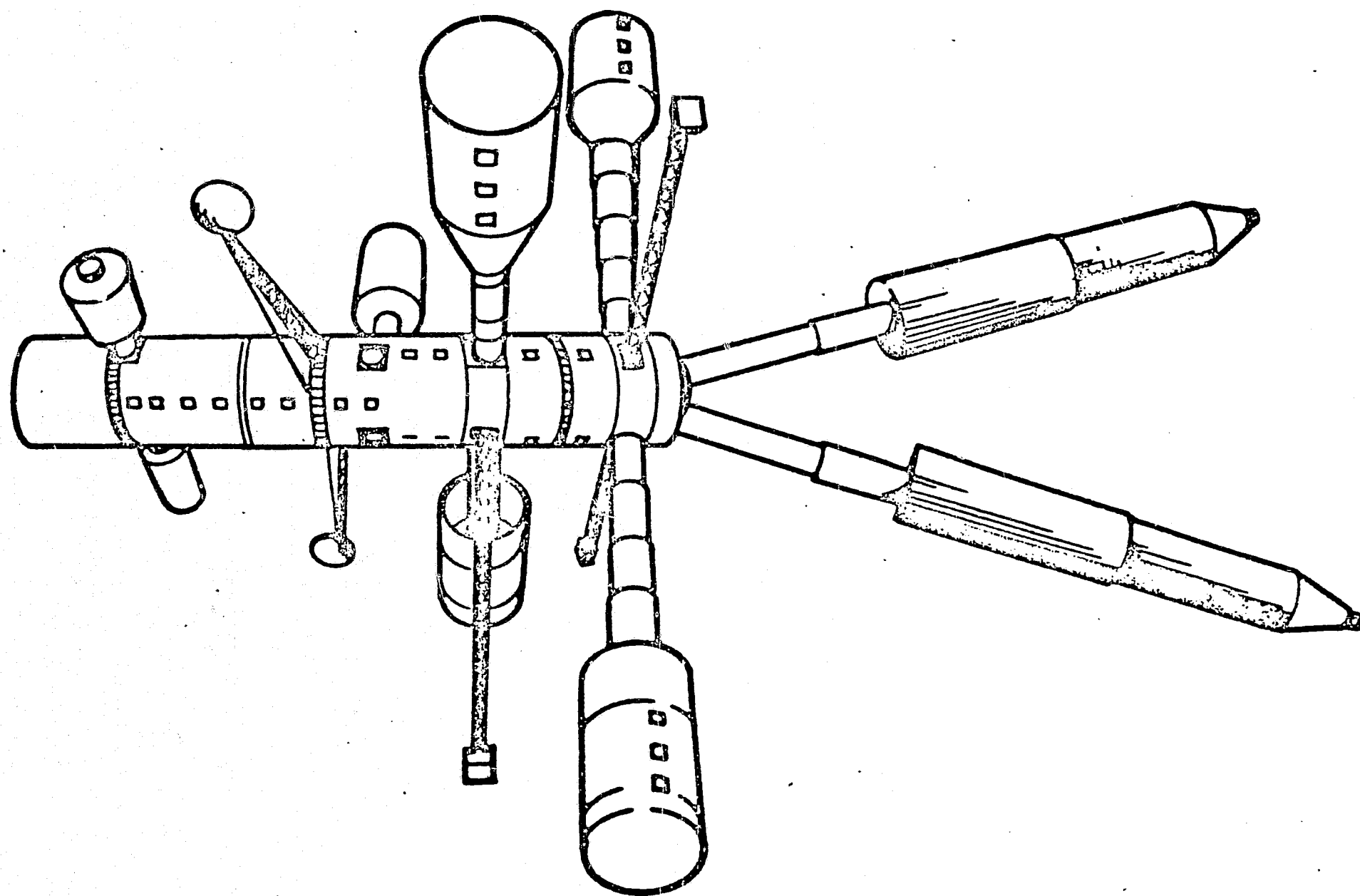


Figure 12. Preferred Space Base Configuration. From MDAC [8].

The solar array is not considered a serious candidate because of the extensive deployed area and the excessive weight of the batteries that would be required. The projected ultimate power requirement for Space Base is 100 kWe. Most of the candidate systems recommend reactor power sources; they differ only in methods of converting the thermal energy to electrical energy.

The thermoelectric generator would have the advantages of inherent ruggedness and no moving parts, but its low efficiency (about 7 percent) makes it presently noncompetitive. The lower efficiency has the dual disadvantages of requiring more power input and larger radiator weight per unit power output.

The mercury/Rankine system also has low efficiency and therefore is not considered to be a promising candidate, although much development work has been done to demonstrate system performance.

The reactor/Brayton power source is a leading candidate for use in the Space Base because it can produce the required power and has high efficiency (about 20 percent). A schematic of a reactor/Brayton system is shown in Figure 13. It includes a primary loop with sodium and potassium (NaK) as the vapor which conducts heat from the reactor to the heat exchanger. Heat is conducted from the heat exchanger to the secondary loop which also uses NaK. The secondary loop contains the alternator, compressor and waste heat rejection exchangers. The operating characteristics of the reactor/Brayton system are given in Table XIV and the weight breakdown is given in Table XV. Examination shows that although the system is capable of high efficiency it weighs about 124,000 pounds for a 100 kWe system. The power system would therefore have to be launched as separate parts and assembled in space. Shields make up about half of the system weight. Even with some 80,000 pounds of shield material, the two reactors must be deployed on booms about 200 feet forward of the crew's compartment to limit the radiation to the 25 mrem per day, which is considered safe dosage.

There are several disadvantages inherent to a reactor/Brayton power system. In addition to requiring large radiation shields, the system produces radioactive products, principally gamma rays and neutrons

Notes:

1. Auxiliary heat-rejection loop not shown.
2. Operating temperature = 80°F in and 240°F out.
3. Heat rejected = 17.6 kWt.

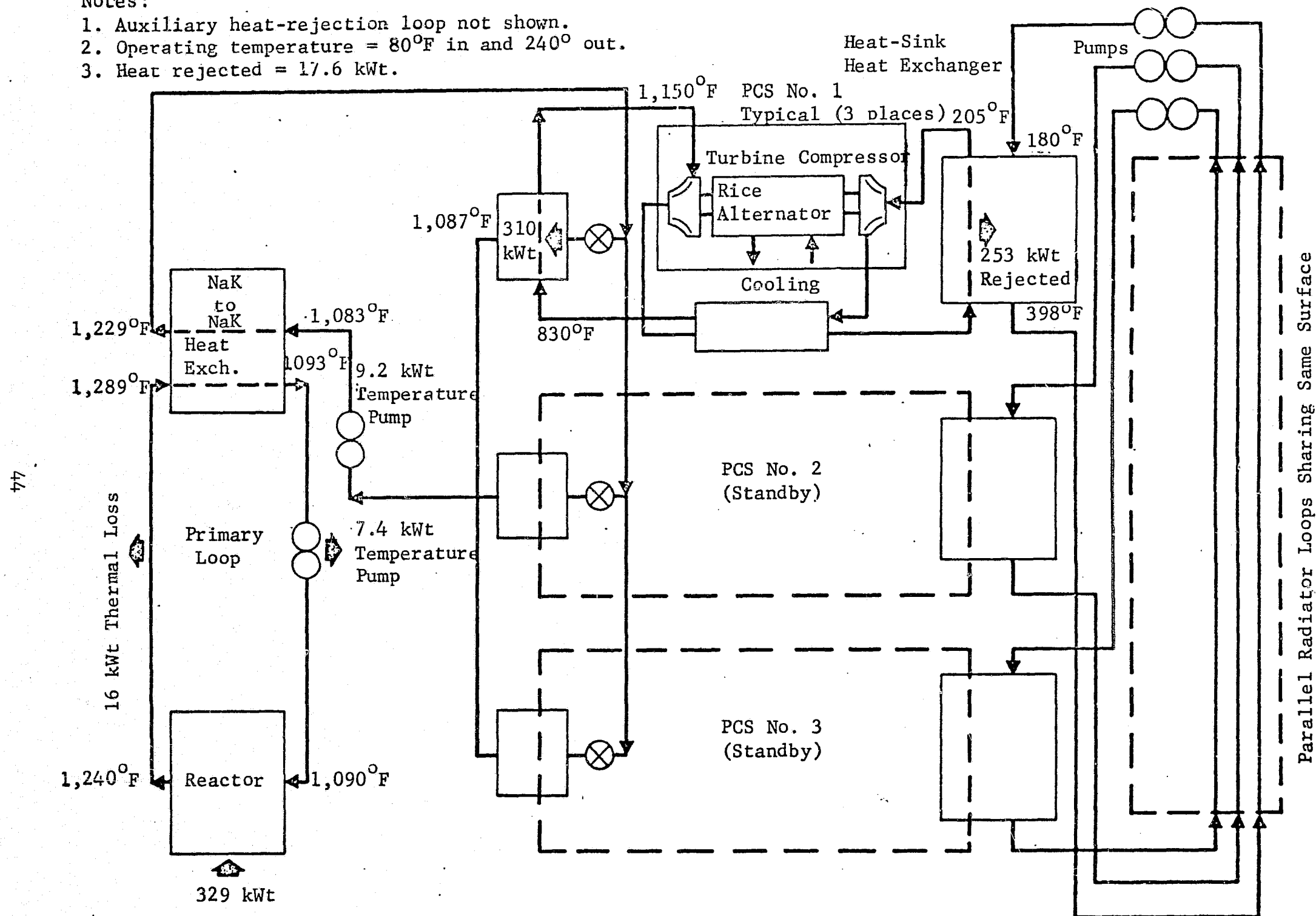


Figure 13. Schematic of Reactor/Brayton-Cycle System. From MDAC [8].

TABLE XIV

SPACE BASE REACTOR/BRAYTON POWER MODULE CHARACTERISTICS [8]*

Net conditioned output power (kWe)	50.0
Conditioning and distribution losses (kWe)	<u>8.2</u>
Net raw output power (kWe at 400 Hz)	58.2
Parasitic pumping and control losses (kWe)	<u>7.8</u>
Gross raw power rating (kWe)	66.0
PCS module rating (gross raw - kWe)	66.0
Installed PCS modules (active/standby)	1/2
PCS design wearout life goal (yr)	2.5
Reactor thermal power (kWt)	329
Prime radiator heat rejection (kWt)	(253)
Auxiliary (low-temperature) radiator rejection (kWt)	(17.6)
Conversion efficiency percent	17.7
Radiator area (ft ²)	2600
Prime	(2400)
Auxiliary	(200)
Average system life (yr)	5
Reactor temperatures (inlet/outlet - (°F))	1,090/1,240

* Space Base will require two modules for a total power of 100 kWe.

TABLE XV
SPACE BASE REACTOR/BRAYTON POWER MODULE WEIGHTS [8]*

Subsystem	Weight (pounds)
Reactor	1,830
Primary loop	655
Shield	40,600
Intermediate loop	750
Power conversion systems (3 at 1358 lb each)	4,074
Turbine-alternator-compressor	437
Heat source heat exchanger	201
Recuperator	268
Heat sink heat exchanger	118
Miscellaneous	334
Total (each)	<u>1,358</u>
Electrical/electronics (3 at 493 lb each)	1,479
Heat rejection loop	592
Radiator/radiator structure	3,750
Structural cone and mounting brackets	1,000
Thermal shroud, tracks, and actuators	1,300
Docking equipment	300
Disposal propulsion	<u>8,450</u>
Subtotal	64,780
Contingency (10 percent)	<u>6,478</u>
Total	<u>71,258</u>
*Two such modules will be required for Space Base.	

which contaminate the primary loop vapor (NaK) and ultimately make it unusable. The reactor vessel is also adversely affected by radiation. It is expected that reactor life could be about five years, but the alternator, compressor, and pumps will require replacement after two or three years. Present plans call for replacement of the power module by Space Shuttle.

The organic/Rankine system employs the same type of heat source as the reactor/Brayton and therefore has the same problems. Since the secondary loop material of the organic/Rankine could be Dowtherm A, it is expected that the overall life of its heat source would be longer because it operates at a lower outlet temperature. Lower outlet temperature increases lifetime and yields higher thermal energy. For these reasons, the organic/Rankine system possesses great potential for space application. However, it has not been developed and tested as extensively as the reactor/Brayton and mercury/Rankine systems.

Table VI in Chapter II shows a load breakdown of the 100 kWe required for Space Base [5,8]. It is seen that 57.2 kWe power is required for operation of the base, 5.7 kWe is allocated for contingency, and 37.1 kWe is for experiments and distribution losses. In the case of reactor failure, or during reactor replacement, an emergency power source will supply power until full power is restored. In some concepts, Space Base is an enlargement of Space Station and therefore the original Space Station power system would be part of the Space Base system. The Space Station power sources (probably isotope/Brayton sources) could serve as emergency power backup during replacement of one of the Space Base power modules for a period up to eight days. Implementation of this concept would, of course, require continuation of the resupply schedules of radioisotope fuel over the life of Space Base.

D. Space Shuttle Power Source

Space Shuttle consists of a booster and an orbiter. The original concept of Space Station/Space Base flight missions would employ Space Shuttle for crew and equipment transfers. In addition, Space Shuttle

would transport parts of the massive Space Base power system to the Space Base docking orbit. More recent concepts suggest that Space Shuttle also could be designed for commonality with Space Station/Space Base launch, with a common booster for both the Shuttle and the Station/Base missions.

The power sources for the booster and orbiter will be markedly different from the Station/Base sources, because the missions of Space Shuttle are quite different. The Space Station power source requires long life (two to five years) therefore system efficiency is important. The Space Shuttle power system must supply large peaks of power for short periods of time, so high efficiency is less important. Both systems require high reliability, but the specifications for the power source for the booster/orbiter have been based on a "fail operational, fail operational, fail safe" philosophy. This would mean that first one and then a second system failure would not degrade operational capability, and that while a third system failure would degrade operation, a safe return to earth could still be made.

Candidate power sources that have been recommended are fuel cells, auxiliary power units (APU's), and batteries. All of these sources can deliver large amounts of power for short periods of time and thus fit the basic load requirements of the booster and the orbiter. The selection of optimum power sources for the booster and the orbiter is complicated, however, by the fact that each vehicle must operate both as a spacecraft and as an aircraft. This requirement can be met by using fuel cells, batteries, and/or auxiliary power units for the spacecraft parts of the flight, and APU's for the aircraft part of the flight.

1. Booster Power Source

A booster power profile is shown in Figure 2, Chapter II. There is a large peak demand of about 48 kW for the three-minute boost, and an average power demand of about 25 kW for the remainder of the flight. Since the lift off flight time of the booster is short and there is a high peak load demand, batteries have been proposed in one study [9]. Batteries appear to be a natural choice for the booster primary power source because of mission time, availability, greater

landing safety, replaceability, and low cost. Silver-zinc, silver-cadmium, and nickel-cadmium are the three most promising types of batteries. A comparison of battery characteristics is shown in Figure 14, in which can be seen the clear superiority of silver-zinc batteries in energy output per pound weight.

In addition to batteries for the primary power source, chemically driven turbo-alternator APU's have been proposed to deliver the high peak power required in the reentry and landing phase of the booster mission [11,12]. During 6 minutes of the reentry and landing phase, 567 horsepower is required by the hydraulic system to deploy and position control surfaces. The turbines that drive the hydraulics would also drive alternators which would deliver the electric power peak demand.

The results of a recent booster power source study are shown in Table XVI. Battery, fuel cell, APU, combined APU/jet engine, and combined battery/jet engine sources are compared. The APU was recommended because of its low weight and its suitability for the large, short-time, hydraulic load during reentry and landing. Relative system weights of batteries, turbo-alternators and fuel cells are shown in Figure 15, where it is seen that turbo-alternators have a clear weight advantage over batteries for mission durations in excess of ten minutes; this is longer than the booster's liftoff phase. There was a tacit assumption in the study [12] that there would be no additional development costs for turbine or alternator. It was expected that such costs would be absorbed in off-the-shelf APU costs, but this is doubtful because the power requirements of the booster are much larger than those of current aircraft. Further study should be made to clarify the relative cost and reliability of batteries versus the APU for the primary power source.

2. Orbiter Power Source

Orbiter power profiles are shown in Figures 3 and 4, Chapter II. The maximum power required by the orbiter is 19 kW for 1 second during boost. In addition to primary electrical power, peak power of 190 horse-

Battery Type

Nickel-cadmium
Silver-cadmium
Silver-zinc

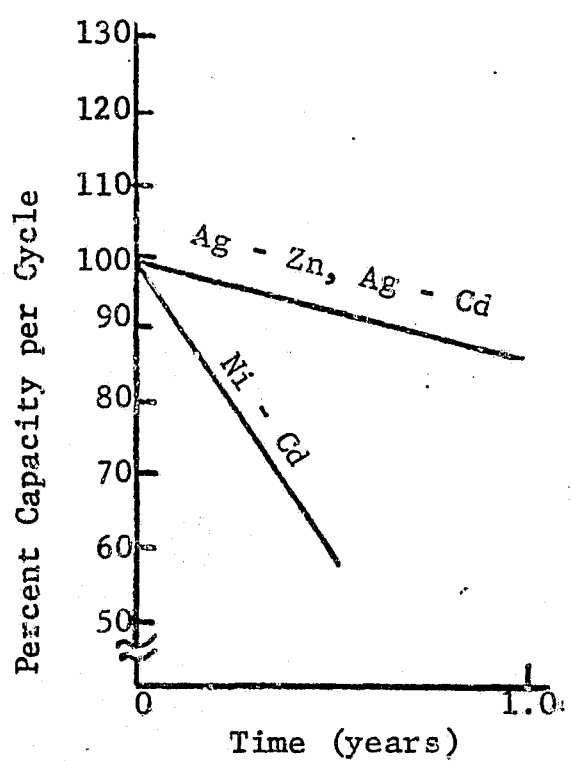
Energy Density

(watt hours/pound)

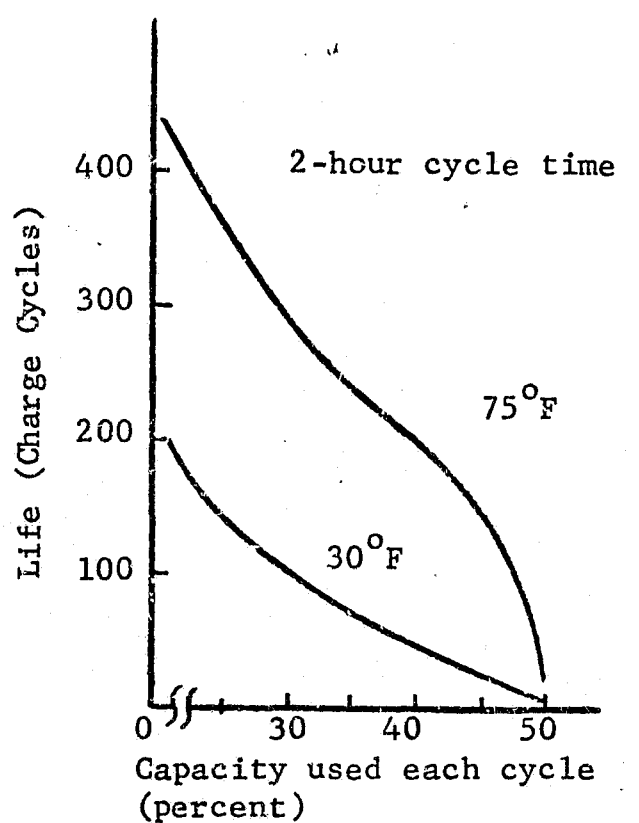
10 to 15

25 to 30

30 to 75



(a) Charging Capacity



(b) Cycle-Life Performance

Figure 14. Typical Battery Characteristics.
From Digesu [9].

TABLE XVI

BOOSTER POWER SOURCE EVALUATION [12]

Source	Weight (pounds)	Cost		Weight	Rating	
		Hardware (dollars/ vehicle)*	Development (dollars)		Hardware	Development
Battery*	9000	2.6×10^6	2×10^5	6	3	1
Fuel cell 20 units, 5 kW each	3000	8.0×10^6	Accomplished	4	6	5
4 units, 25 kW each	2000	3.0×10^6	25×10^6	3	5	6
APU**	1500	1.14×10^5	2.5×10^5	1	1	2
APU/jet ⁺ engine	1750	1.8×10^5	4.6×10^5	2	2	3
Battery/ jet engine ⁺	5700	1.4×10^6	6.1×10^5	5	4	4

*Vehicle life base: 250 missions. Ag/Zn batteries have minimum weight, life of 50 missions; replacement costs are included.

**APU is required for hydraulics in all cases; costs and weights are for alternator, transformer/rectifier, gear reducer, and coolant pump.

⁺Jet engine requires constant speed drive, in addition to the alternator, T-R unit, gear reducer, and coolant pump.

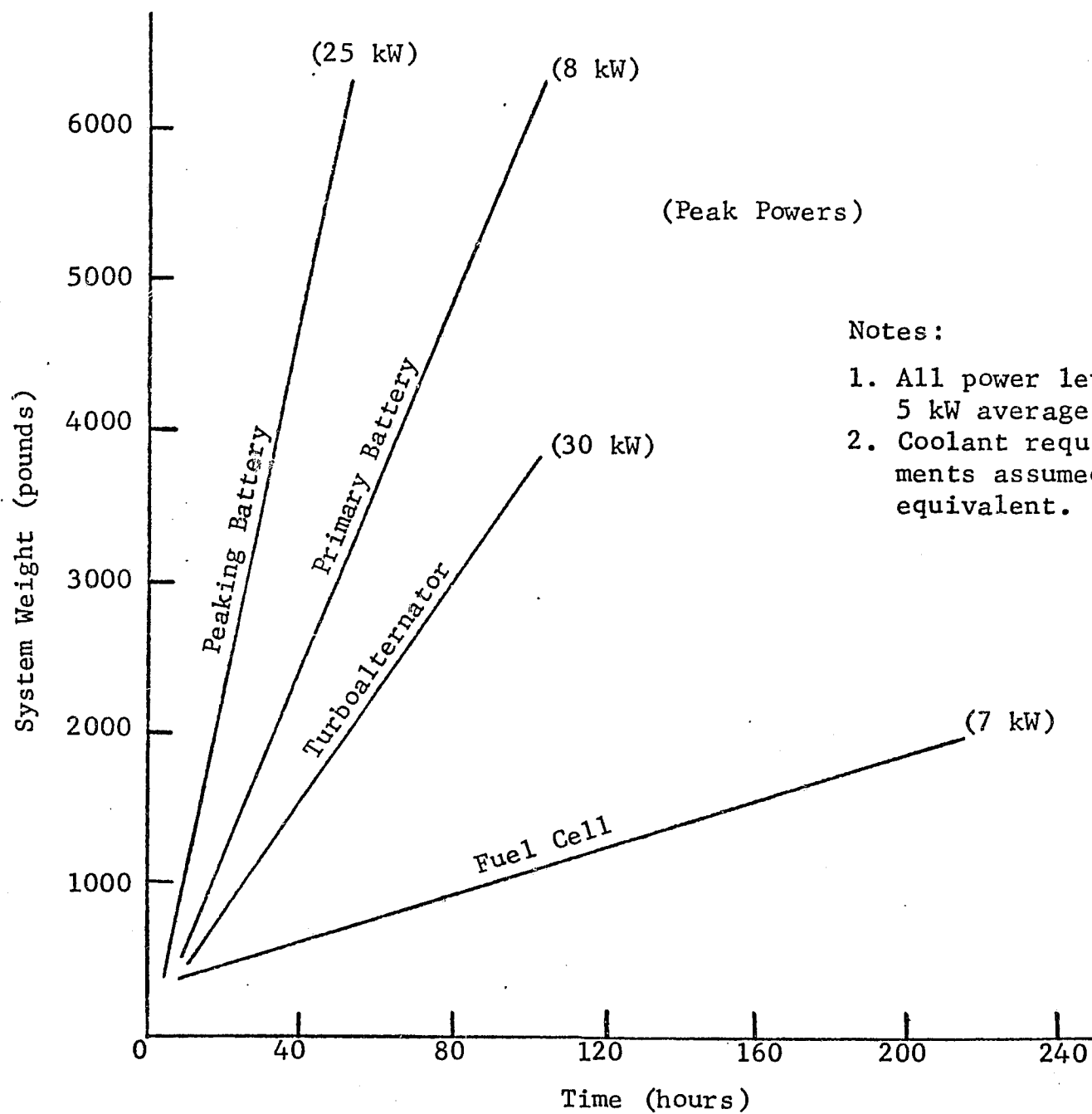


Figure 15. Comparative Weights of Power Systems.
From Digesu [9].

power must be supplied to the hydraulic system during reentry and landing. Fuel cells weigh the least per watt of power, as shown in Figure 15, and have been tentatively suggested [15] for primary electrical power. Chemically driven turbines would furnish both hydraulic power and aircraft mode electrical power [12].

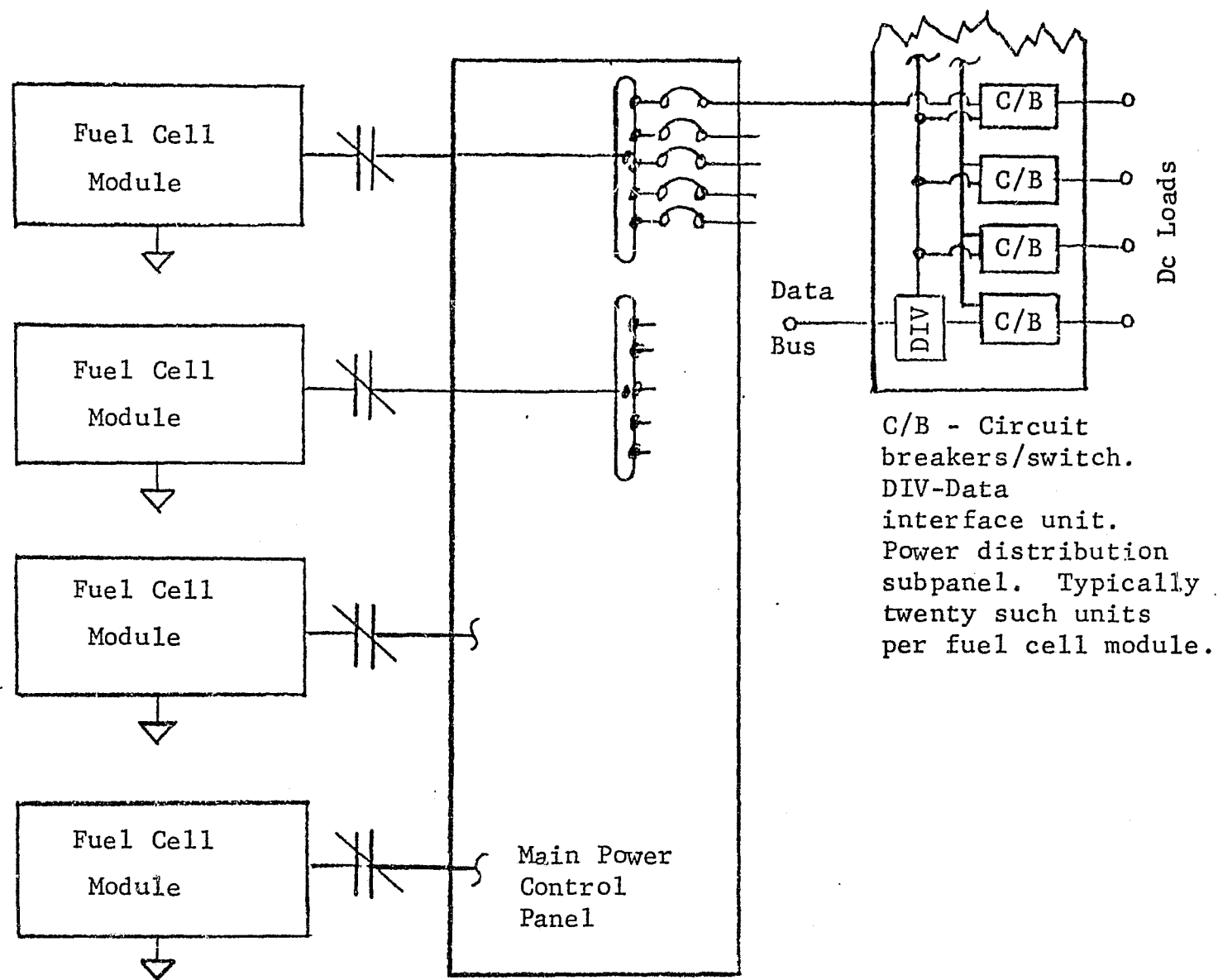
Fuel cells convert chemical energy into dc electrical power directly, with energy densities of 25 W-hr/lb. Fuel consumption of about 0.85 lb/kW-hr is practical, with efficiencies of about 60 percent. All of the fuel cell characteristics are impressive when compared with the lower efficiency and the higher weights of thermodynamic systems. Fuel cells do have disadvantages of low power capability (7 to 10 kW per unit) and short life, and thus are not good candidates for long duration missions such as Space Station. Present-day fuel cells have only 24 Vdc output, and are not adaptable to series stacking for a high voltage dc system. This is a disadvantage because a high voltage system is desirable for spacecraft power transmission to reduce wire weight. Comparative characteristics of fuel cells are given in Table XVII. Although this table shows the power output of the fuel cells is about 5 kW, they will drive much larger short-duration loads. For example, one of the Pratt and Whitney cells can supply a load of 20 kW for an hour. The fuel cell modules could provide all the electrical power required by the vehicle loads except for main propulsion engine pumps, windshield anti-icing, landing lights, and hydraulics. These power requirements are supplied by turbo alternator APU's.

The MDAC study has proposed the use of four hydrogen/oxygen fuel cell modules, each rated at 7.5 kWe. Since the average load is about 5 kWe, this system would meet the requirement of "fail operational, fail operational, fail safe," because one operating module could supply enough power for a safe return. A schematic diagram of a fuel cell/APU power source is shown in Figure 16. The three alternators in Figure 16 would be driven by turbines which would also drive hydraulic pumps. The alternators would supply electrical power during the aircraft mode, and the hydraulic system would deploy motors and position control surfaces. The estimated hydraulic power requirement for the orbiter is

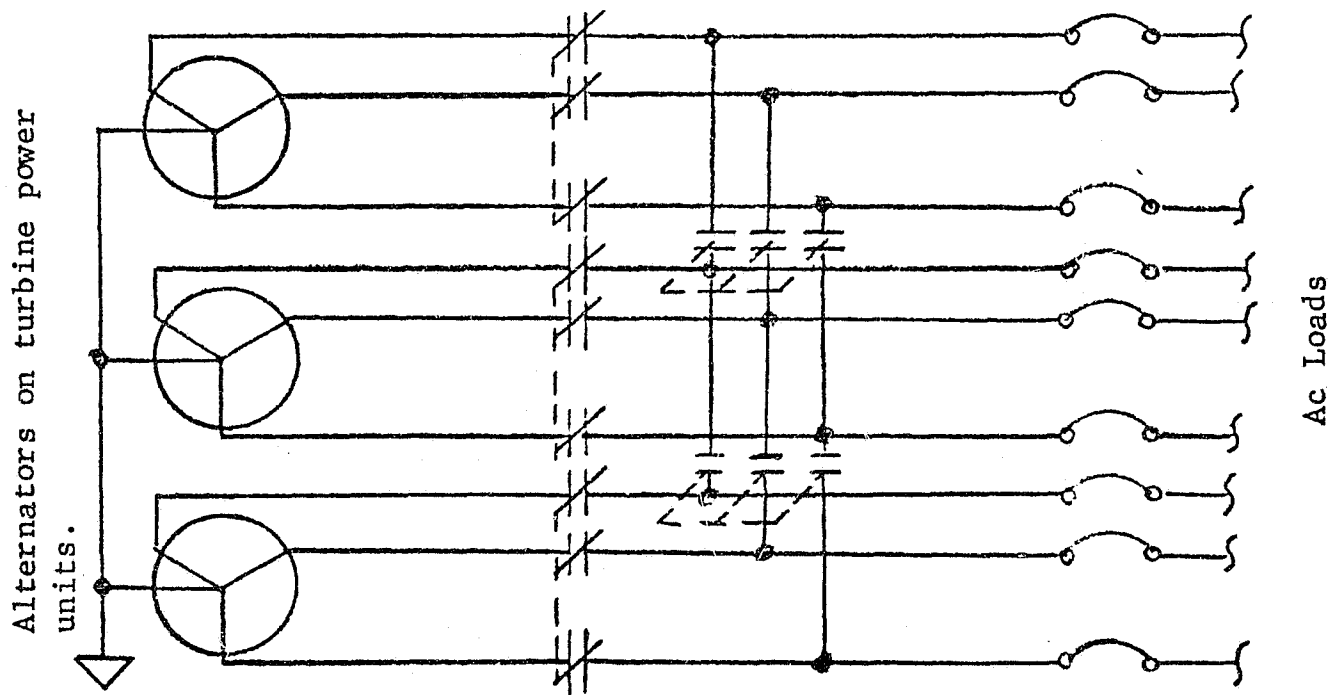
TABLE XVII

FUEL CELL SYSTEMS COMPARATIVE DATA [9]

Type	A.C. Electronics Div. Cells		Pratt and Whitney Cells	
	Helium Cooled	Liquid Cooled	PC8B-3	PC8B-3 Modified
Power (kW)	2.0	5.0	4.5	4.5 to 20*
No. of cells	31	30	31	31
Cell area (sq ft)	0.4	1.0	0.5	0.5
Fuel consumption (lb/kW)	0.8 @ 1 kW	0.8 @ 1 kW	0.86 @ 1 1.4 kW	0.86 @ 1 kW
Max sustained load (kW)	3.2	5.0	4.5	20 (1 hr)*
Radiator required	Yes	Yes	Yes	Yes
System weight (lbs)	190	170	105	132
Design life (days)	62	125	90	90
Operating temperature (^o F)	195	195	190	190
Life test (hrs)	3000	None	2100	None
Cumulative test (hrs)	35000	None	8100	None
Dimensions (in)	35 x 21 x 16	11 x 11 x 32	18 x 18 x 20	20 x 20 x 18



(a) Dc Subsystem



(b) Ac Subsystem

Figure 16. Orbiter Electrical Power System.
 From MDAC [12].

190 horsepower for 27 minutes. The alternators would supply electrical power for the main propulsion engines, fuel boost pumps, windshield anti-icing, and landing lights. The proposed alternators would be 115 Vac, 400 Hz, 3 ϕ machines. A load profile of the alternator loads is shown in Figure 3, Chapter II. Large transient power loads during engine ignition and shutdown are apparent in the load profile. Parallel operation of the alternators would not be possible because frequency synchronization is difficult with the large, short-duration loads. The switchgear indicated in Figure 16 is arranged to allow load transfer if any of the alternators fail; all of the loads could be supplied from one alternator in case the other two fail.

A trade off study was made by MDAC, to determine the optimum power source for supplying power to the orbiter main propulsion engines. It was concluded that the ac alternator was preferred because it was available off-the-shelf, whereas a 120 Vdc fuel cell, although competitive on a weight basis, would require considerable expense for further development. In another trade off study to select a power source for all loads except the main propulsion engines, it was determined that fuel cells have a clear weight advantage over ac alternators, even if the alternators are already part of the system. The reason for this is that the turbines which drive the alternators would have very low fuel efficiency if they were used only to drive the alternators without also driving the hydraulic system. The penalty associated with the additional fuel weight dictated a choice of fuel cells for the light-load phases of the mission. Unfortunately, the study did not account for the distribution bus weight penalty which would be associated with the 28 Vdc fuel cell system.

E. Summary

Previous studies have indicated that if Space Station is not considered in context with Space Base, solar array/battery sources offer some competition to isotope/Brayton sources for Space Station. When all factors are considered, however, including the weight penalties associated with preventing corona, an isotope/Brayton source is preferred.

An excellent 1200 Hz alternator has been developed for the isotope/Brayton source. Unfortunately, most off-the-shelf utilization equipment uses 400 Hz power. The choice of a 1200 Hz alternator would require added power conversion equipment.

For Space Base, only reactor/Brayton sources have been seriously considered. Projected isotope fuel supply appears to be adequate for isotope/Brayton power sources for Space Base as well as Space Station. A great savings in shield weights and development costs could be realized by using a radioisotope heat source instead of a reactor.

The durations and the natures of the dual spacecraft/aircraft modes of the Space Shuttle orbiter and booster influence the choices of Shuttle power sources. Combined ac and dc sources have been recommended in recent studies. APU's would furnish electrical peak power demands, but APU operation would only be feasible during the time that the driving turbines are also furnishing power to the high-demand hydraulic systems.

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IV. DISTRIBUTION SYSTEM

A. Introduction

Historically, electrical power distribution systems for spacecraft have been modeled after aircraft power systems which usually have been 115 Vac, 400 Hz, 3 ϕ , and/or 28 Vdc systems. Most of the aircraft circuit breakers, switches, and displays have been located in the cockpit. In a typical 28 Vdc distribution system for aircraft, the wire weight constitutes about 60 percent of the total distribution system weight because low dc voltage requires high current. In future spacecraft it would be desirable to reduce the weight of distribution wire. It would also be desirable to eliminate the electromechanical relays because they require high operating power and have limited reliability and lifetime. Solid state switching and multiplexing techniques could eliminate the electromechanical relays and reduce wire weight. Distribution at higher voltages could also reduce wire size. The employment of multiplexed power control on the proposed vehicles of the 1977-1987 decade -- Space Shuttle, Space Station, Space Base -- requires consideration of many interacting factors:

1. Development cost of new switching and distribution devices.
2. Costs of departure from present load equipment which is designed for 115 Vac and 28 Vdc.
3. Corona problems at high voltages.
4. State-of-the-art limitations on maximum operating voltages of solid state switches.
5. Limited overload capability of solid state devices.

The penalties of these factors would have to be offset by the following inherent advantages of a multiplexed distribution system employing solid state switches:

1. Lower weight.
2. More reliability.
3. Flexibility for future growth.
4. Built-in test capability.
5. Longer life expectancy.

This section of the report will consider various aspects of the power distribution system. Several alternate distribution systems which have been proposed for possible use in Space Station, Space Base, and Space Shuttle will be discussed.

B. Space Station Power Distribution System

In Chapter III candidate power sources for Space Station were discussed in some detail. One study (NAR) tentatively recommended a solar array power source with 112 Vdc fed directly to inverter regulators, while another (MDAC) has recommended an isotope/Brayton power source with an alternator generating 115/200 Vac, 1200 Hz, 3 ϕ . The choice between these two sources is not obvious but depends on such undetermined factors as system growth and system integration requirements and the availability of isotope fuel. The following discussion will review two power distribution systems which have been proposed [7,13]. Recommendations will then be made for power distribution systems based on load requirements rather than on power source characteristics.

1. A Distribution System for Solar Array/Battery Power

The NAR recommended solar array power source consists of four solar array panels with 22.3 kWe output each at 112 Vdc. The power conditioning and distribution system is shown in Figure 5 of Chapter III. Each of the four solar array panels in Figure 5 is connected to an inverter which would convert the 112 Vdc to 240/416 Vac, 400 Hz, 3 ϕ . Voltage and frequency are regulated to close tolerances to permit parallel operation of the outputs of the inverters. Nickel cadmium batteries serve as backup sources and drive separate inverters which supply power to the primary power buses. The batteries furnish power for peak loads during daylight and also supply all the power during the dark part of the orbit. The 416 Vac from the inverters is transmitted to load center autotransformers which convert power to 208 Vac and (after rectification) to 56 Vdc. The loads are supplied from either a single ac bus (208 Vac), a double ac bus (416 Vac), or a 56 Vdc bus. Solid state switching is planned for the load

switching at 208 Vac and 56 Vdc with transistors switching the dc loads and SCR's switching the ac loads. The solid state switches will be computer controlled for power routing and circuit protection. A multiplex "data bus" system is planned to control the load switching. The multiplex system provides the following advantages:

1. Decreased wire weight.
2. Built-in test.
3. Ease of system growth.
4. Lower weight.
5. Higher reliability.

The rationale for recommending high ac voltage for the system are:

1. Reduced transients by switching at zero voltage crossover.
2. Many loads are ac.
3. Lower voltage drop across the solid state switch.
4. Reduced wire weight.

The load breakdown between dc and ac is somewhat arbitrary, since heaters, etc., can operate on either dc or ac. In the study cited (NAR), it is proposed to operate fluorescent lighting on 208 Vac, the molecular sieve on 416 Vac, and the galley oven on 416 Vac. These loads can be designed specifically for the Space Station to fit the operating voltage. Most of the experiment loads, however, have not been clearly defined as to their requirements for dc or ac power.

Possible corona problems with the dc systems have been mentioned, but there are also potential corona problems with ac systems. The primary bus voltage of 416 Vac in the NAR system would have 587 V peak, which would greatly exceed the corona onset voltage in unpressurized areas. To guard against corona it would be necessary to pressurize or heavily insulate both the wire and the connectors, which would increase the system weight. Studies have shown that a 416 Vac system would not weigh much less than a 208 Vac system. It is even questionable whether 208 Vac should be used for power distribution, since the peak voltage (293 V peak) would also exceed the corona onset voltage in unpressurized areas, and it would exceed the breakdown voltage of present-day solid state power controllers. (It would be possible to switch SCR devices at zero crossover voltage so that the on-off transition would

not occur at high voltage, but the switch would have to survive the total line voltage when it is opened. Information furnished by vendors indicates that SCR's can operate at 220 Vac, however.)

Power distribution at 110 Vac would be feasible, although a penalty of approximately 1500 pounds in wire weight would ensue. The 416 Vac system was recommended by NAR to reduce the penalties of wire loss and wire weight and to obtain high efficiency in conversion equipment. The solar array/battery system of MDAC, described in Chapter III, would transmit power at 112 Vdc, the array output voltage. Because of the low voltage and correspondingly high current, the losses in the transmission line between the solar array and the inverters, which would be located inside the spacecraft, would be quite high. It has been estimated that the transmission cable loss would be 758 watts and the weight of the cable would be 158 pounds. The NAR approach was an attempt to reduce this loss and this weight. If the spacecraft lights, the molecular sieve, and the galley oven could be powered at high voltages, as recommended by NAR it would reduce the load on conversion equipment and would permit some reduction in wire size.

The power distribution system recommended in the NAR study included parallel operation of the four inverters for load sharing, but 416 Vac for transmission must be considered too high for present state-of-the-art solid state switches. Since it is proposed to mount the inverters on a boom, the transmission cabling between the inverters and the source bus would have to be pressurized to prevent corona breakdown. It is not clear that the weight penalty for pressurization was considered. If a commutator coupling is required to permit spinning the spacecraft for artificial gravity, it would require special design to prevent electrical breakdown.

A trade off study of the NAR solar array/battery power system [7] is shown in Figure 17 for comparison with other recommended systems. (Figure 17 was derived from Figure 5.) Excluding wire losses, subsystem losses in the solar array/battery system would be 15 kW.

2. A Distribution System for Isotope/Brayton Power

One major study (MDAC) has recommended the power distribution system shown in Figures 18, 19, and 20 [13]. Two isotope/Brayton sources,

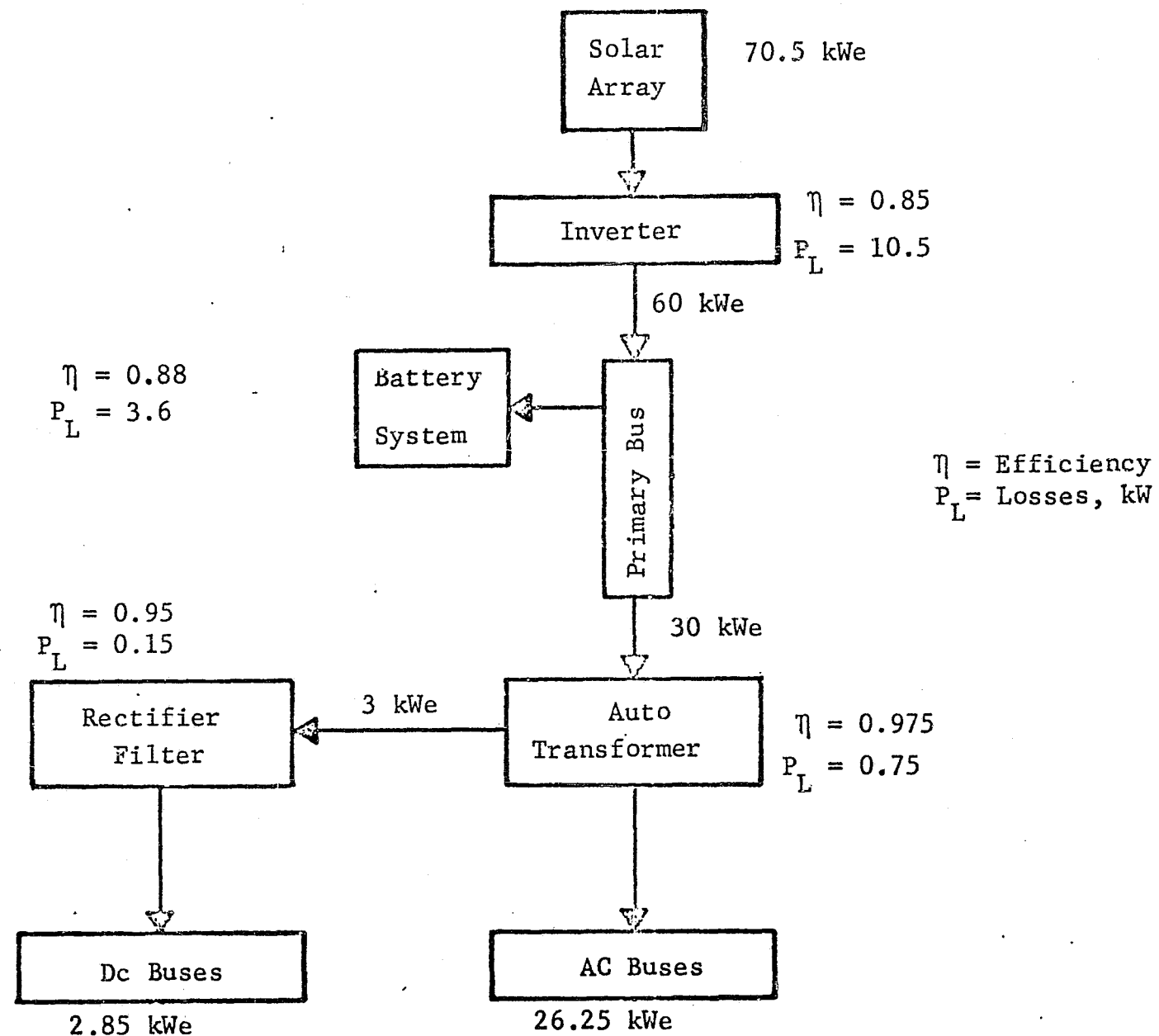


Figure 17. Solar Array/Battery Trade Off Study. From NAR [7].

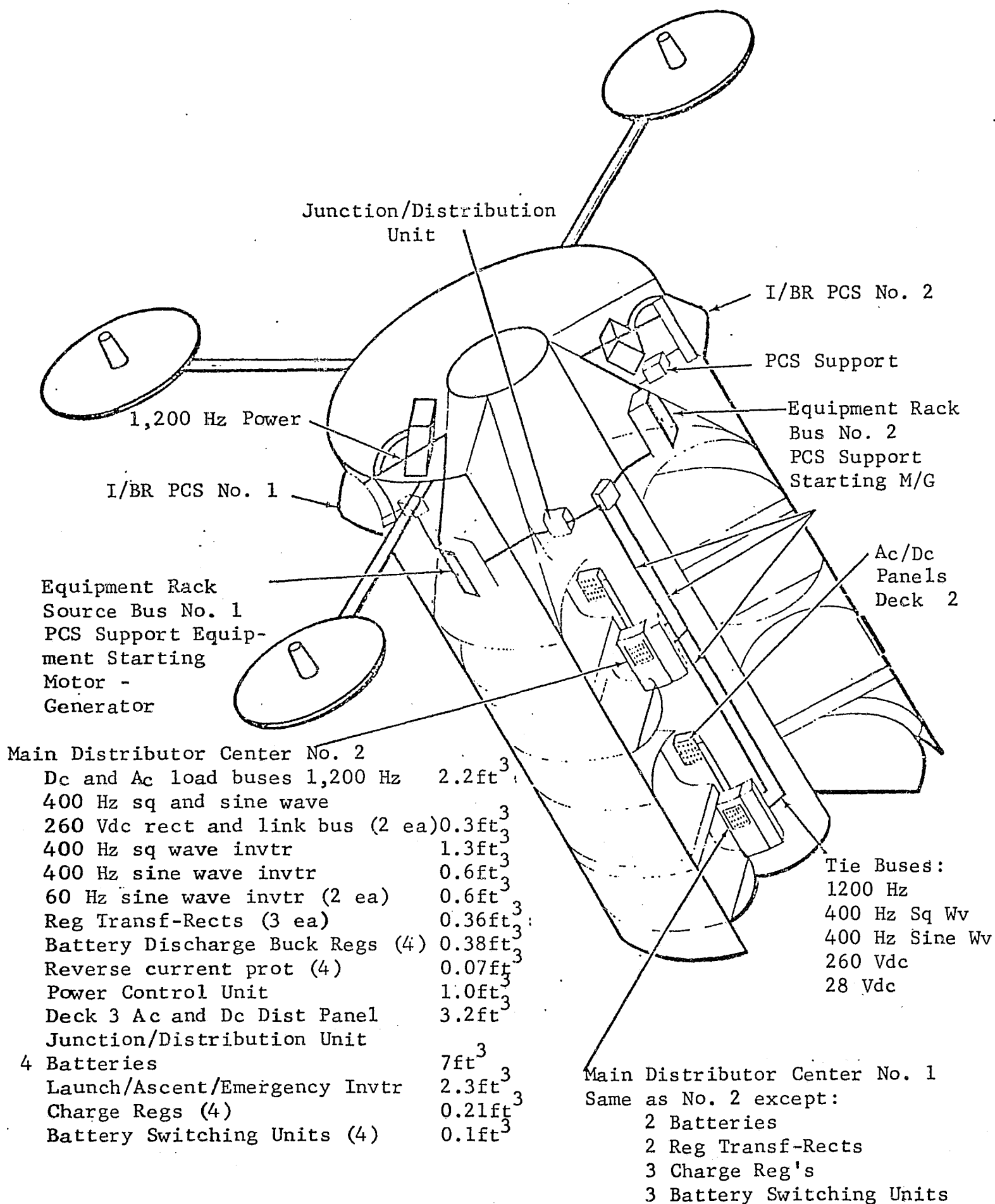


Figure 18. Space Station Power System Layout.
From MDAC [13].

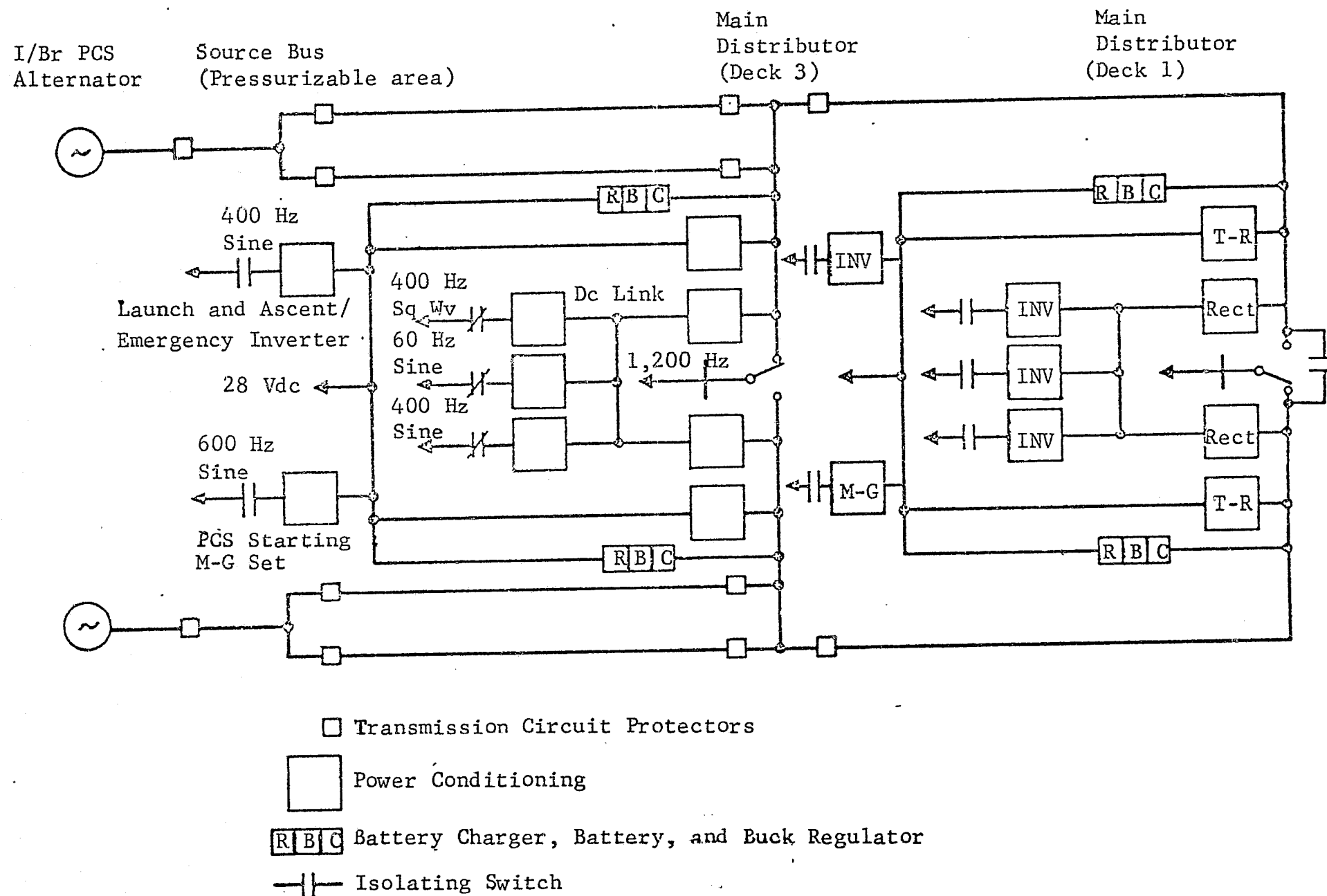


Figure 19. Selected System for Space Station. From MDAC [13].

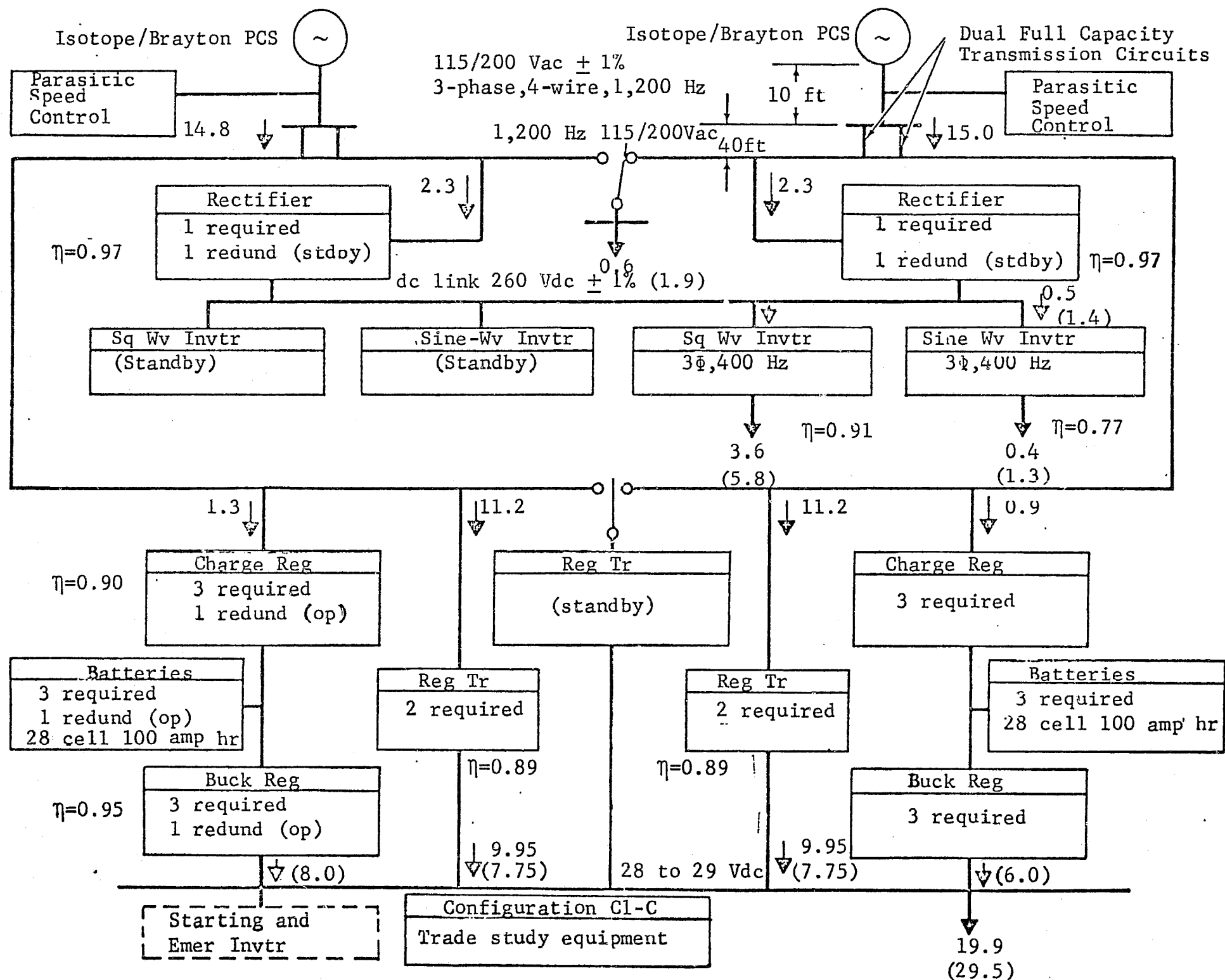


Figure 20. Isotope/Brayton Power Distribution System. From MDAC [5].

as shown in the figures, drive two alternators each of which has about 25 kVA capacity. Each alternator output, 115/200 Vac, 1200 Hz, 3 Φ , is conducted (by a pair of transmission cables) to two main distribution centers. The interconnections shown in Figure 20 indicate that at each distribution center the two 1200 Hz inputs are rectified and then tied together on a common 260 Vdc bus. The 260 Vdc tie bus then powers three inverters. One inverter produces a 115 Vac, 400 Hz, 3 Φ sine wave; the second produces a 115 V peak, 400 Hz, 3 Φ square wave; and the third produces a 115 Vac, 60 Hz, 3 Φ sine wave. The two 1200 Hz primary buses at each distributor center also power two transformer rectifiers, which produce 28 Vdc. The 28 Vdc outputs are connected to a common bus which distributes most of the spacecraft power. The 28 Vdc bus drives, among other loads, a 115/200 Vac, 400 Hz, 3 Φ sine wave inverter and a 115/200 Vac, 600 Hz, 3 Φ sine wave motor/generator set, both of which are used for emergency operations. Each of the two 1200 Hz primary buses also powers a battery charger/regulator at each distribution center. The battery modules at each center serve as backups for the 28 Vdc power.

The distribution voltages available at each of the two main distribution centers shown in Figure 19, are:

1. 115/200 Vac, 1200 Hz, 3 Φ from a transformer.
2. 115/200 Vac, 400 Hz, 3 Φ square wave.
3. 115/200 Vac, 400 Hz, 3 Φ sine wave.
4. 115/200 Vac, 60 Hz, 3 Φ sine wave.
5. 28 Vdc, with battery backup.
6. 115/200 Vac, 400 Hz, 3 Φ sine wave power (inverted from 28 Vdc) for launch and ascent/emergency.
7. 115/200 Vac, 600 Hz, 3 Φ sine wave power from a motor generator set (driven from 28 Vdc) for starting the isotope/Brayton alternator.

The distribution scheme of Figures 19, 20 and 21 has the advantage of high voltage, high frequency transmission, which would minimize the weight of the transmission cable. Distribution at 28 Vdc and at 115 Vac, 400 Hz, would accommodate standard electrical loads. Rectification to dc makes load sharing easily possible since the outputs of the transformer/rectifiers can be tied together. The 260 Vdc tie buses could have corona troubles unless precautions are taken to insulate wiring and connectors

and to pressurize the high voltage parts of the system. A trade off study has been performed to determine the relative advantages of candidate distribution systems [5]. The transmission and conditioning configurations were compared quantitatively for weight and efficiency and qualitatively for such factors as relative complexity, component technology requirements, and operational flexibility. Some of the ground rules of the study are given in Table XVIII. The most important ground rules are the nominal system load of 25 kWe and the weight penalty of 350 lb/kW for power losses. The trade off study block diagram, shown in Figure 20, includes the 260 Vdc tie bus and the 28 Vdc distribution bus, but the 60 Hz bus was omitted. The 260 Vdc tie bus supplies power to drive the inverters, which are more efficient with a high input voltage. Most of the load (19.9 kW) is supplied at 28 Vdc. A small amount of power (2.2 kW) charges the battery backup sources. The comparison of two of the power transmission conditioning systems in the trade off study is given in Table XIX. The recommended isotope/Brayton distribution system is configuration C-1C. From the table it is seen that the equivalent distribution system weight is 3633 lb including weight penalties for power losses. The required input to the distribution system is 30 kWe.

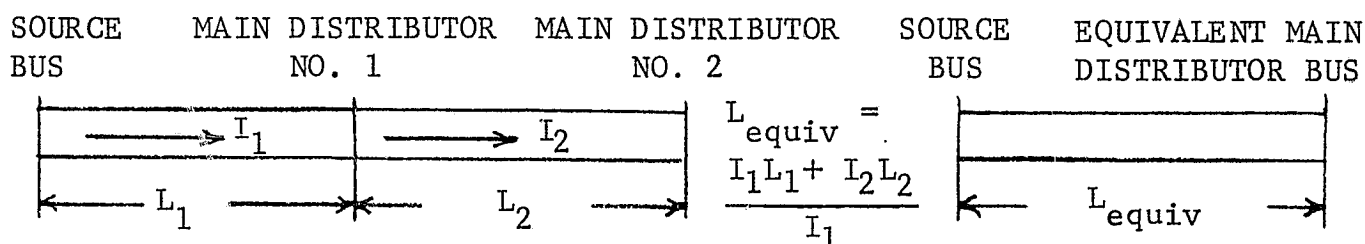
A trade off study of a solar array/battery, 112 Vdc distribution system is shown in Figure 21 [5]. A comparison of the isotope/Brayton system with the solar array/battery system is of interest because a second major study, described in the next section of this report, has recommended a solar array/battery system over the isotope/Brayton system. Reference to the trade off study configuration A-2 in Table XIX shows that the solar array/battery distribution system has an equivalent weight of 7276 lb. It also would require about 65 kWe input power. The recommended isotope/Brayton system, configuration 1-C1, has a much smaller weight, 3633 lb. It also would require less input power, only 30 kWe.

The advantages of the isotope/Brayton source over a solar array/battery source are due first to shorter transmission cable, which means less wire weight, and second to a lower penalty for transmission losses. Most of the transmission loss of the solar array/battery source is attributable

TABLE XVIII

TRANSMISSION/CONDITIONING -- ASSUMPTIONS [5]

1. Nominal system load is 25 kWe average at the load buses (ac and dc combined).
2. Power required at the source is given in net kWe (power for source parasitic loads and controls is not included).
3. Transmission distance from source bus to equivalent main distributor bus is determined as follows:



4. Weights include all power transmission, conditioning, and distribution equipment from power source terminals through power protection and distribution panels on individual decks. Batteries and source parasitic loads and controls are not included. Power instrumentation and control wiring is not included.
5. Weights do not include spares, but do include redundant transmission circuits from source to source bus.
6. Losses are calculated assuming, for configuration C-1C, that one parallel wire in one of the two transmission circuits is open.
7. All power interfaces with experiments are at 28 Vdc.
8. Power conditioning equipment is represented by lumped equivalents.
9. Specific weight of power conditioning equipment is for rated power.
10. Rated power is based on maximum steady-state demands.
11. Conductor weight is for round aluminum wire except for use of copper from solar array terminals to source bus.
12. Wiring from each solar array panel is redundant, including blocking diodes.
13. Neutral wire in 3-phase, 4-wire ac transmission circuit is same gage as phase wires.
14. Common wire in 3-wire dc circuit is same gage as positive and negative polarity wires.
15. Losses in 3-wire dc and 4-wire ac circuits are calculated assuming balanced loads.
16. Wire losses are calculated for resistance at 100°C. Losses in ac transmission circuits are calculated for 0.8 power factor line current.
17. Weight penalty factor for losses is 350 lb/kW.
18. Transmission circuit conductors are sized to minimize sum of wire weight and weight penalty for losses at 350 lb/kW.
19. Calculations for weights and losses reflect use of electromechanical switches, relays, and circuit breakers. Solid state switches, relays and circuit breakers are recommended for the Space Station, contingent on development of required low-loss, low-weight, high-power devices.

TABLE XIX
SPACE STATION POWER TRANSMISSION SYSTEM
WEIGHT SUMMARY [5]

Subsystems	Trade Off Study Configuration	
	Solar Array Battery, A-2 (pounds)	Isotope/Brayton C-1C (pounds)
Load bus conditioning	948	590
Source bus to main distributor buses	286	106
Source to source bus	354	16
Protection & control, distributors, panels, etc.	1759	1257
Related mechanical and support struc- ture	669	394
Total system weight, subtotal	4016	2363
Weight penalty for system power losses ⁽¹⁾	3260 ⁽²⁾	1270 ⁽³⁾
System weight plus loss penalty, total	7276	3633

1. The weight penalty is 350 lb/kW.
2. Power losses in the solar array/battery transmission system are 9.31 kW.
3. Power losses in the isotope/Brayton transmission system are 3.62 kW.

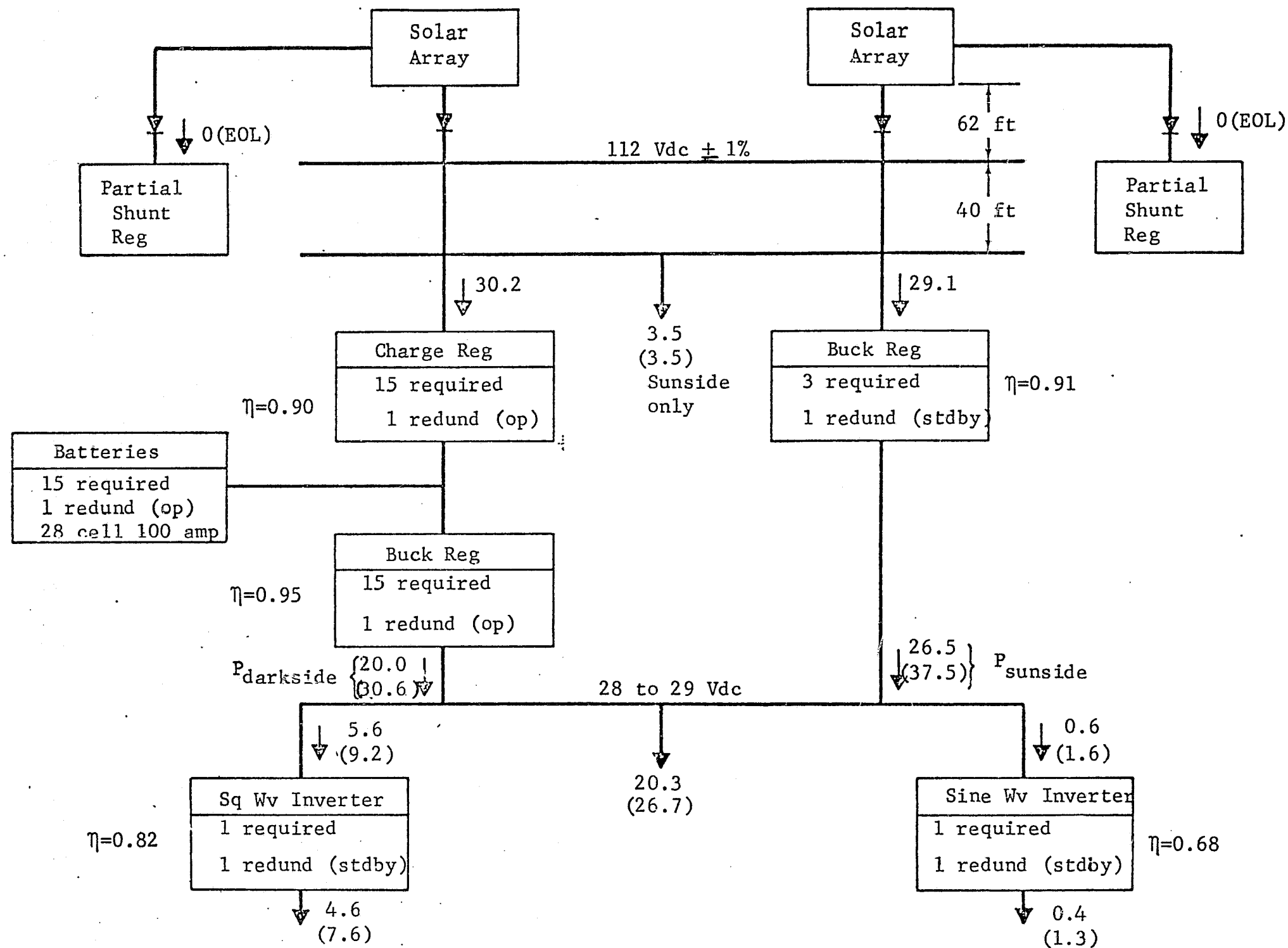


Figure 21. Solar Array/Battery Power Distribution System, Trade Off Study. From MDAC [5].

to the additional power that must be generated and transmitted during daylight hours to charge the batteries for night usage. Still another advantage of the isotope/Brayton system is that about 4 kWt waste heat from the source can be used to heat the spacecraft. With the solar array source, the 4 kWt would have to be supplied electrically.

There are, however, several problems with the isotope/Brayton power system. As shown in Figure 18 the proposed mounting of the isotope/Brayton sources would be at an extremity of the Space Station. If the transmission lines from the sources are routed through unpressurized sections of the vehicle, pressurized conduit would be needed to prevent breakdown with 200 Vac (282 V peak). Transmission at 1200 Hz could increase the dangers of corona. The trade off study should include penalties for the weight of pressurized conduit. Also, in the proposed system the 260 Vdc tie bus at the distribution center could have corona problems. Special precautions with wire and connector insulation might be needed to prevent breakdown. Another problem with the system of Figure 19 is the proposed use of solid state switches to replace electromechanical circuit breakers. Higher voltage solid state devices would have to be developed since the present state-of-the-art is about 100 Vdc. Devices which will switch 200 Vac or 260 Vdc for the proposed power system cannot be assured for the near future. In fact, further development would be required to produce suitable electromechanical switches.

C. Space Base Power Distribution Systems

The recommended power sources for Space Base, which were mentioned in Chapter III, were all nuclear reactors, differentiated by various conversion equipment configurations. Because of the large power output of each generator (50 kWe) the nuclear reactor thermal power required would be quite high; and it would be necessary to mount the reactors on booms, remote from the load centers, in order to reduce the radiation level at the crew location. The distances from the sources to the distribution centers would be in excess of 200 feet.

Two alternate power distribution systems have been proposed for Space Base [8], with the choice depending on whether an isotope/Brayton or a solar array/battery is the power source for Space Station. (Since the Space Station power system will furnish backup power for the Space Base, the Station and Base distribution systems must be compatible.) Both Space Base systems would use reactor/Brayton primary power sources. If a solar array/battery source powers the Space Station, a 260 Vdc distribution system would be recommended for Space Base. If an isotope/Brayton source powers the Space Station, a 115 Vac, 400 Hz distribution system would be recommended for Space Base. Even if the Space Station source is isotope/Brayton, several different methods of interconnecting power are possible for Space Base. The recommended system, shown in Figure 22, utilizes parallel outputs of both the reactor/Brayton and the isotope/Brayton systems. With the configuration shown, one reactor/Brayton source could be shut down for maintenance or replacement while the other reactor/Brayton and the backup isotope/Brayton sources would supply the required power. Table VI, Chapter II, shows that 57.3 kWe would supply all power needs except for the experiments, and this amount of power could be furnished by one reactor/Brayton with one isotope/Brayton backup source.

In a study on conceptual designs for Space Base power distribution systems, four candidate power systems were evaluated to determine their relative merits [15]. The conversion systems were:

1. Cycloconverter.
2. Dc link.
3. Dc distribution system.
4. 1200 Hz system.

The results of the evaluation are tabulated in Tables XX and XXI. It can be seen that a 1200 Hz transmission system has lower conversion system weight and lower losses than the other systems evaluated. Despite the indicated advantages of 1200 Hz distribution, a reactor/Brayton source with a 400 Hz output was strongly recommended [15]. A 400 Hz system would require less frequency-conversion equipment than a 1200 Hz system, because 400 Hz motors, solenoids, etc., could be driven directly from the distribution buses. Furthermore, much 400 Hz utilization equipment is available off-the-shelf.

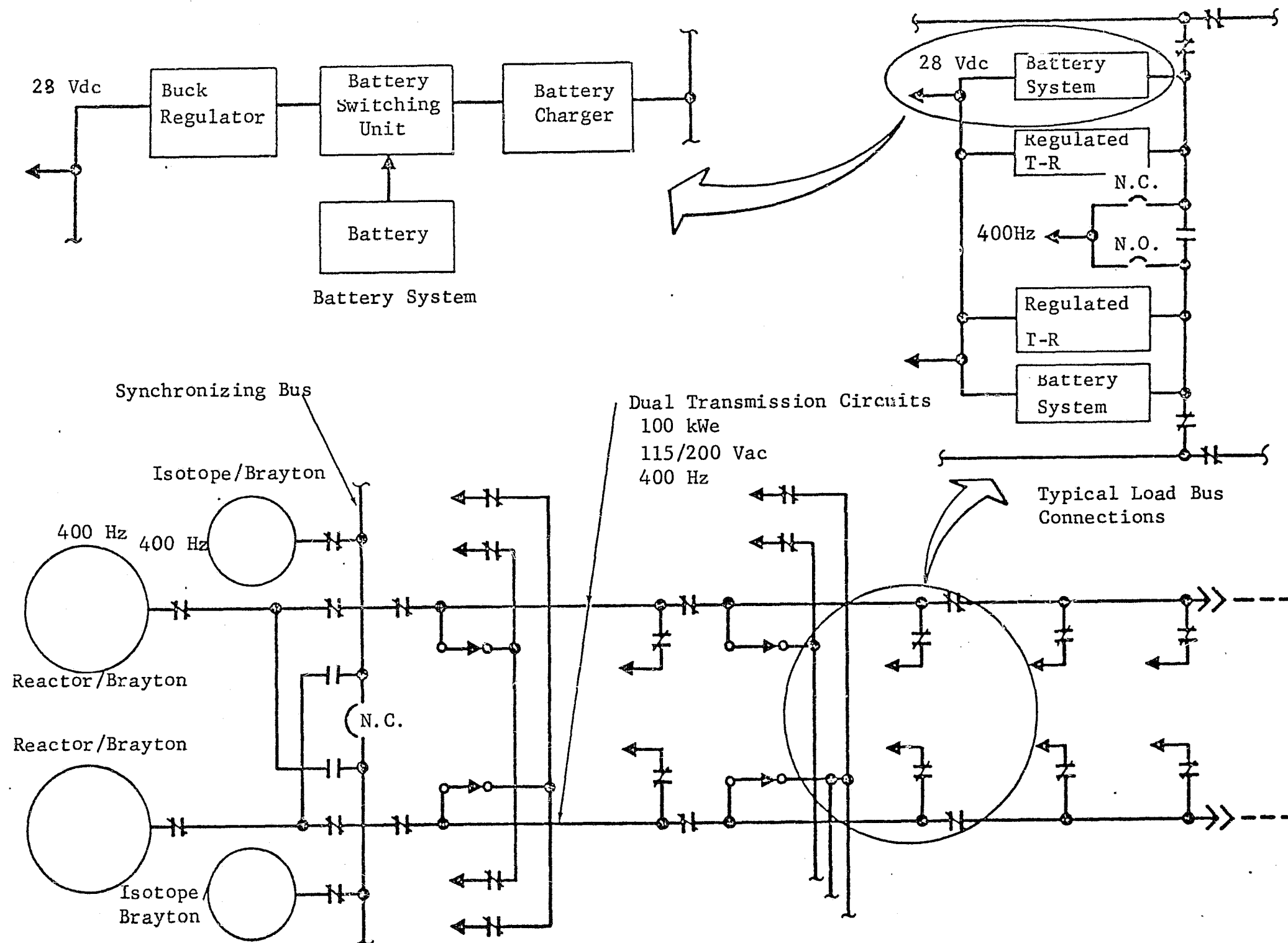


Figure 22. 400 Hz Reactor/Brayton Primary with 400-Hz Isotope/Brayton Backup for Space Base.
From MDAC [8].

TABLE XX

WEIGHT OF CONVERSION EQUIPMENT
SPACE BASE POWER SYSTEM [15]

<u>Subsystems</u>	<u>Cycloconverter System</u> (pounds)	<u>Dc Link System</u> (pounds)	<u>Dc Dist. System</u> (pounds)	<u>1200 Hz System</u> (pounds)
Frequency conversion	400	1000	1000	100
Rotating transformer	210	210	-	140
Rectifier and regulator	400	400	-	400
Slip ring	-	-	350	-
Inverter	-	-	680	-
Dc to dc (250 Vdc to low voltage)	-	-	600	-
Total	1010	1610	2630	640

TABLE XXI

SYSTEM LOSSES AND POWER FACTOR PENALTY
SPACE BASE POWER SYSTEM [15]

<u>Subsystems</u>	<u>Cycloconverter System</u> (kilowatts)	<u>Dc Link System</u> (kilowatts)	<u>Dc Dist. System</u> (kilowatts)	<u>1200 Hz System</u> (kilowatts)
Frequency converter	2.80	18.0	8.0	3.24
Rotating transformer	3.54	4.3	-	5.25
Hub module rectifier	0.64	0.8	-	1.06
AGM rectifier	1.40	1.7	-	2.00
Slip ring	-	-	1.05	-
Inverter (400 Hz)	-	-	6.00	-
HV dc to LV dc	-	-	4.50	-
Total losses	8.38	24.8	19.55	11.55
Power factor penalty	30.00	-	-	-
Power available (assuming 100 kWe raw power)	61.62	75.2	80.45	88.45

One way to save weight is to use high voltage to supply loads that are not critical of voltage level. Such loads as the molecular sieve, galley, heating, and lighting could be designed specifically for the Space Base and could be powered directly by the transmission voltage, eliminating the need for some conversion equipment. An estimate of the total power rating of these loads is shown in Table XXII. About 20 percent of the total Space Base power could be supplied at the transmission voltage.

The weight saving effects of higher transmission and distribution voltages have also been evaluated and are tabulated in Table XXIII [15]. For this study, weight penalties were assessed for transformers and other conversion equipment needed to supply power at the proper voltage levels to the loads.

It can be seen that 220 Vac for both transmission and distribution yields the lowest weight. It is of interest to note that 110 Vac for both transmission and distribution requires about 1800 lb more wire weight than the 220 Vac system. An important point apparently overlooked was the weight penalty of about 350 lb/kW for losses. If this penalty were included, the 440 Vac transmission system would be the leading candidate.

TABLE XXII
SPACE BASE LOADS WHICH COULD BE DESIGNED
FOR UNCONDITIONED POWER

<u>Loads</u>	<u>Power</u> (kilowatts)
Galley	1.0
Lighting	6.6
Thermal	6.2
Molecular sieve	5.6
Total	19.4

TABLE EXLII

SPACE BASE TRANSMISSION AND DISTRIBUTION SYSTEM
WIRE WEIGHTS [15]

Voltage Level		Line Weight (pounds)	Transformer Weight ⁽¹⁾ (pounds)	Distribution Wire Weight (pounds)	Transformer Weight ⁽²⁾ (pounds)	Wire Weight Plus Transformer Penalty (pounds)
<u>Transmission</u> (volts)	<u>Distribution</u> (volts)					
110	110	2070	None	1150	None	3220
220	110	810	600	1150	None	2560
220	220	810	None	450	155	1415
440	110	315	600	1150	None	2065
440	220	315	600	450	155	1520

1. Transformer between transmission line and distribution system.
2. Transformers between distribution system and loads.

D. Space Shuttle Power Distribution Systems

The Space Shuttle includes a booster which has a three hour mission and an orbiter which has a seven day mission. The candidate power sources for the booster and the orbiter were discussed in Chapter III. This section of the report will treat the power distribution systems for both the booster and the orbiter. As mentioned in Chapter III both the booster and the orbiter must have the capability of performing as a spacecraft and as an aircraft since these vehicles must fly back to a base, and therefore the proposed power systems contain features of spacecraft and aircraft power systems.

1. Booster

The booster will be manned by a small crew and will be launched with the orbiter attached. As shown in Figure 1, Chapter II, the orbiter will detach from the booster at an altitude of 186,000 feet and the booster will then fly back to its base. The electrical power system for the booster is influenced by the following factors and criteria:

1. Both spacecraft and aircraft type loads.
2. "Fail operational, fail operational, fail safe" criterion.
(See Section D in Chapter III for explanation.)
3. Power system must fly 50-250 missions before replacement.
4. High peak transient loads (propulsion engines).
5. Cost to launch is a critical parameter (\$5000/lb).
6. Commonality of power distribution system with the orbiter.

Two types of power sources, batteries and alternators, have been proposed for use in the booster. The alternators would be driven by chemically fueled turbines which would also supply hydraulic power for the aircraft control surfaces, the landing gear, and motor deployment. A discussion of these power sources is given in Chapter III. This section will discuss distribution techniques for the booster power system.

A power distribution system suggested by MDAC for the booster is shown in Figure 23. It consists of distribution at 115/200 Vac, 400 Hz, with conversion of part of the power to 56 Vdc. The main propulsion engine pumps, and some of the EC/LS equipment could be driven directly

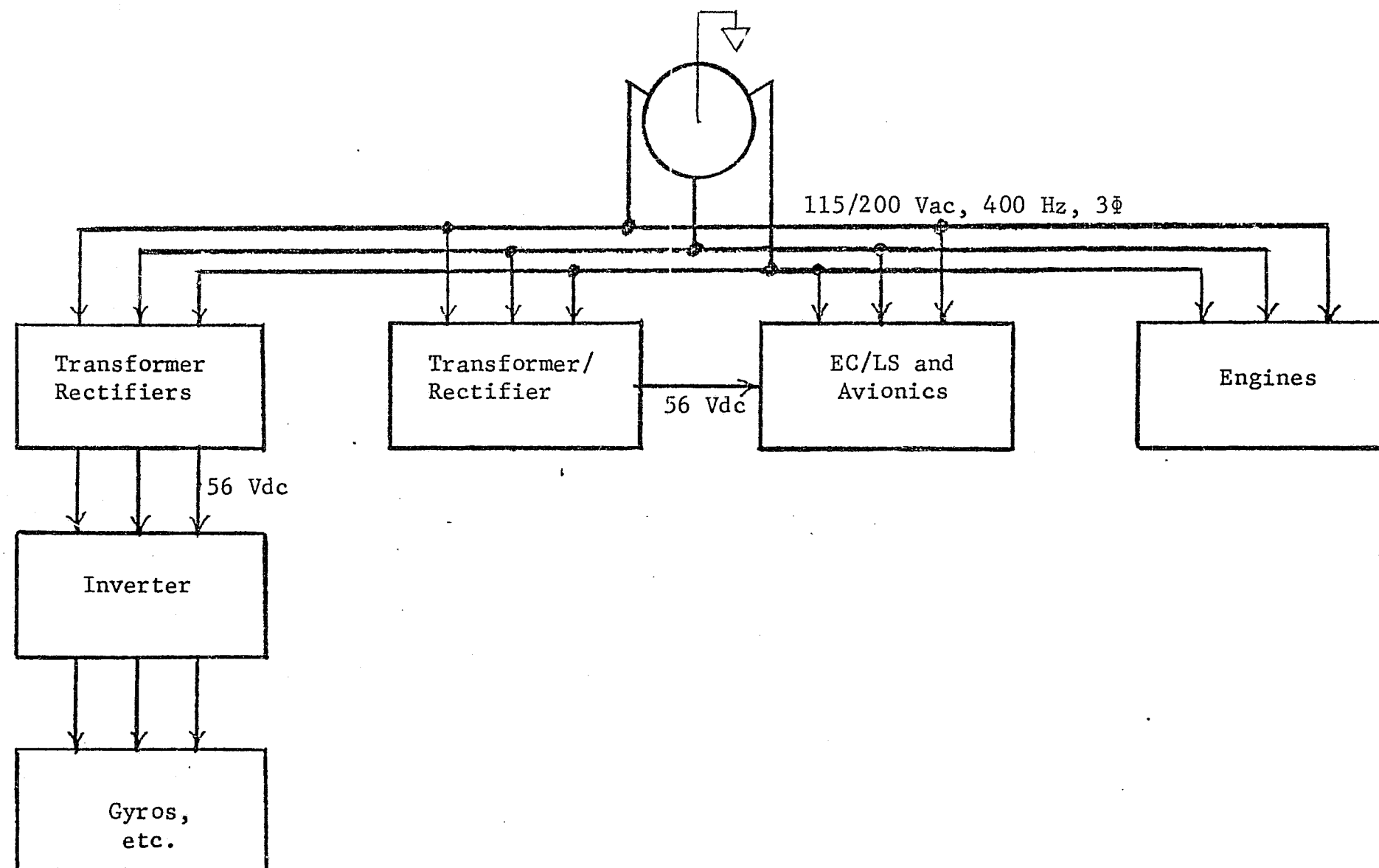


Figure 23. Booster Power Distribution System.
From MDAC [17].

from the alternator. The avionics and the remainder of the EC/LS system would be supplied by 56 Vdc. The gyros and other sensitive ac loads would be supplied by a closely controlled sine wave inverter or a cycloconverter.

Decentralized distribution would permit voltages such as 28 Vac, 6 Vac, etc., to be produced at local load centers. A trade off study has shown that a decentralized distribution system would yield a large weight savings. Solid state switches will probably be developed in time to allow multiplexing the dc power in the system of Figure 23. Switching of the 115/200 Vac could be accomplished with SCR power controllers [16].

In a trade off study of voltage levels and types in a power distribution system for the booster, 115/200 Vac and 56 Vdc ranked first. A 115/200 Vac, 115 Vdc system ranked second. It was concluded that \$32 million could be saved by using a 115/200 Vac system instead of a 28 Vdc system [17]. Unfortunately, the study did not separate the booster costs from the orbiter costs. The latter should have included the development costs for a 120 Vdc fuel cell in the evaluation of the second-ranked system.

2. Orbiter

The orbiter flight profile is also shown in Figure 1. The orbiter requirements are similar to the booster requirements, since it too has both spacecraft and aircraft modes of operation. The criteria and constraints listed in the previous section will apply to orbiter also. In addition, the orbiter must rendezvous and dock with the Space Station. During the docking phase the orbiter will receive electrical power from the Space Station so provisions must be made for interface. After detachment from the Space Station the orbiter will reenter the atmosphere and return to base as an aircraft. As mentioned in Chapter III, fuel cells could supply dc electrical power for the orbiter, and APU's could supply ac power for the following loads:

1. Main propulsion engine pumps.
2. Air breathing engine pumps.

3. Window de-icing.
4. Heaters.
5. Landing lights.

Since the orbiter must be propelled into orbit, its weight penalty is about five times that of the booster, or about \$25,000 per pound of launched weight [17]. For this reason weight saving on the orbiter is even more critical than on the booster. This implies an increased desirability of high distribution voltages. However, state-of-the-art fuel cell technology is limited to 28 Vdc output.

A power distribution system suggested by MDAC for the orbiter is shown in Figure 24. (See also Figure 16.) Solid state switching could be incorporated into a multiplexed power distribution system with a master control unit for all the switching functions. The system shown in Figure 24 would require development of 115 Vdc fuel cells, however, and the development of a high voltage fuel cell would be costly. As an alternative, a dc to dc converter could be used to increase the voltage with only a modest penalty in power system weight (4 lb/kW). A design study has shown that a 28 Vdc distribution system (5 kWe average) would weigh 527 lb more than a 120 Vdc system, so the dc to dc converter weight would be more than offset by converting fuel cell output to a higher voltage.

E. Multiplex Control of Electrical Power Distribution

A multiplex system for electrical power control on Space Station, Space Shuttle, and Space Base has been discussed in an earlier report [1]. The advantages cited for multiplex power distribution were:

1. Reduced weight (fewer wires and shorter cable runs).
2. Flexibility which enables system expansion.
3. Redundancy for reliability.
4. Built-in test (maintainability).
5. Low cost.
6. Parity checks for system self-check.

A proposed multiplex system (SOSTEL II) designed for an A-7 aircraft is shown in Figure 25 [18]. Redundant master control units drive remote output units which in turn supply control signals to the power controllers.

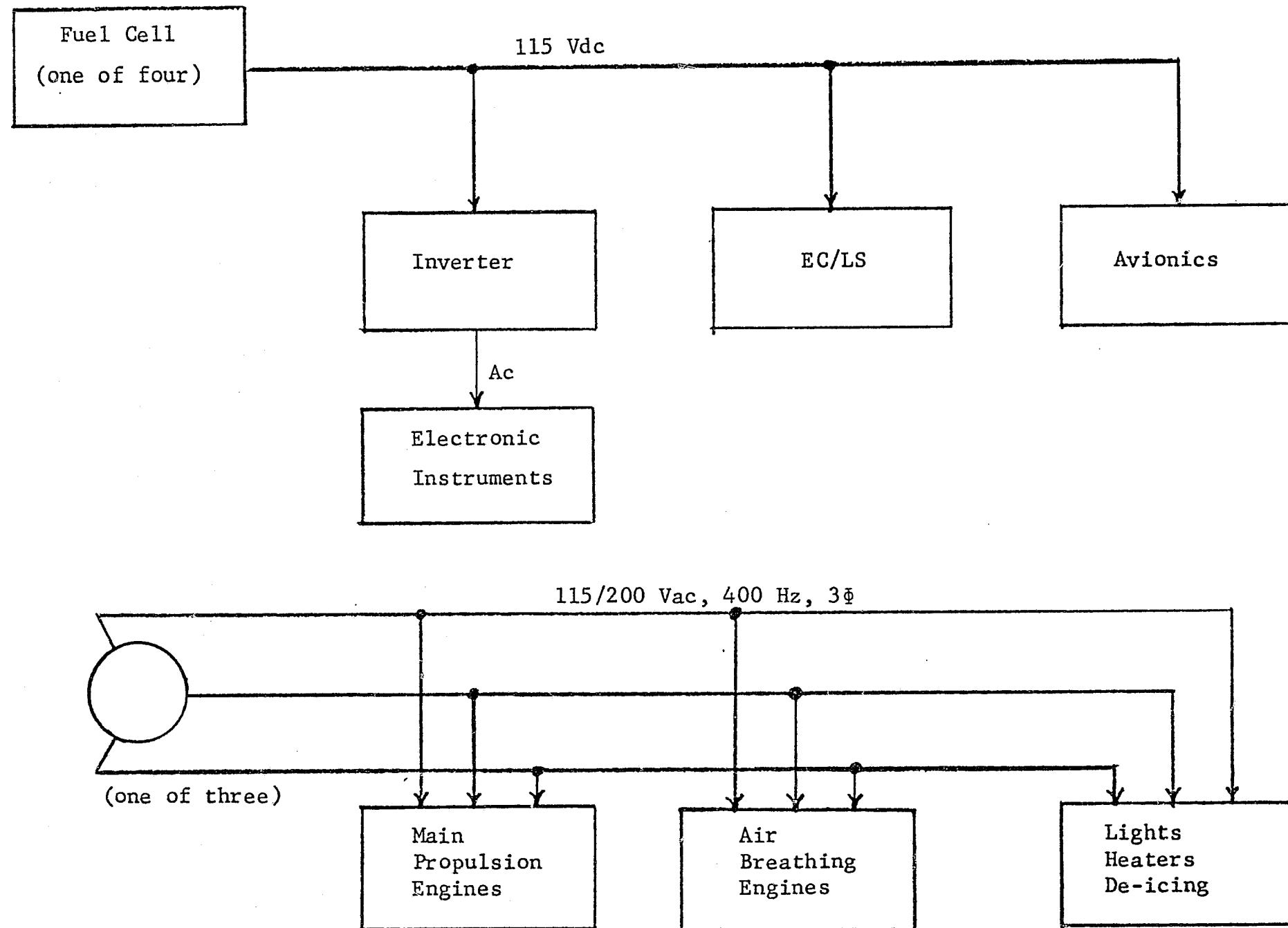


Figure 24. Orbiter Power Distribution System. From MDAC [12].

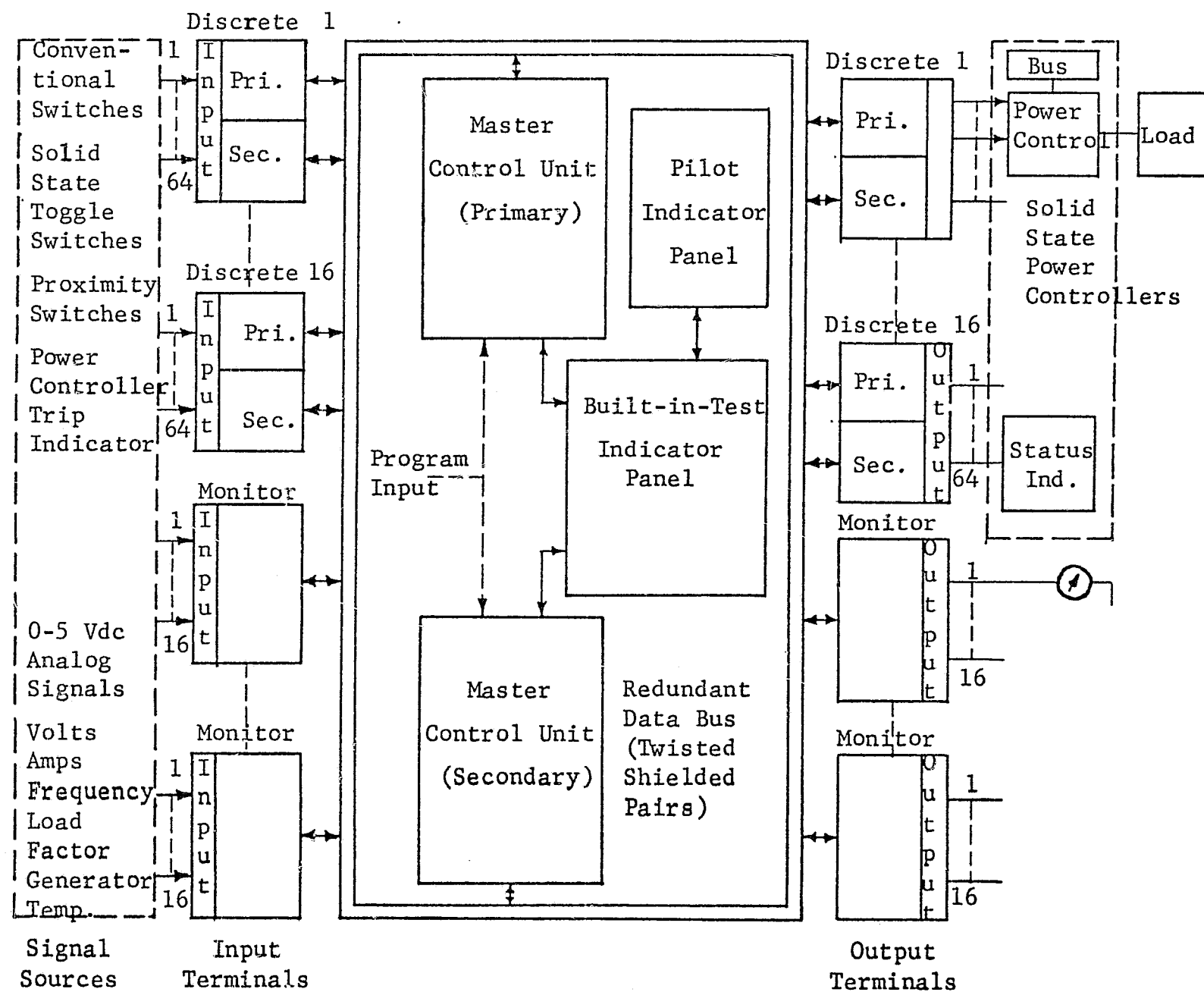


Figure 25. Multiplexed Power Systems.
From King [18].

Solid state devices are proposed for the power controllers.

A comparison of hard wired and multiplex systems is shown in Table XXIV where it is seen that a 45 percent weight savings, plus savings in volume, improved reliability, and better maintainability are realized with the solid state system [19]. In addition to the advantages that a solid state multiplex system has for aircraft, there are several important advantages for spacecraft:

1. Load shedding.
2. Soft turn-on and turn-off, which produce decreased transient amplitudes.
3. Preflight checkout.

Although multiplex systems have not yet been designed for spacecraft, there is merit in transferring the technology of multiplex power distribution which has been designed for aircraft to the booster and orbiter, which perform as aircraft for part of their mission. For this reason several organizations are developing system concepts for a Shuttle multiplex power distribution system. Some of the basic studies required for the multiplex system are:

1. Dedicated power system control versus shared control through one central computer.
2. Configuration of remote control units.
3. Power controller design.

TABLE XXIV
ADVANTAGES OF SOLID STATE DISTRIBUTION SYSTEMS
FOR AIRCRAFT [19]

Parameter	System		Improvement (percent)
	Conventional	Solid State	
Weight	576 lb	315 lb	45
Volume	2756 cu in	1500 cu in	45
Reliability	498 fail/10 ⁶ hr	116 fail/10 ⁶ hr	77
Maintainability	0.014 mmh/flt hr	0.0028 mmh/flt hr	80

The trade off study of a shared central computer control system versus a dedicated power control system (with a separate computer) has many aspects. The only advantage of using part of the central computer for controlling power distribution would be that one computer would perform all mission functions. (Since a central computer would be high-rate, 2×10^5 to 2×10^6 bits per second, the power distribution control information could be easily accommodated.) One of the advantages of a dedicated data bus system would be that control would be maintained within the power system itself, thus avoiding accidental power shutdowns from override modes that might occur in a central computer. Another advantage would be better noise immunity for power control messages because the relatively slower speed multiplex system could use longer bit pulses. A third advantage would be that the dedicated data bus could be tailored to the power system, with better utilization of the master control units. Redundancy could be readily obtained with two (perhaps even three) master control units, multiple data bus links, and multiple remote control units. From a preliminary study it would appear a dedicated data bus system has strong advantages over a shared central computer system.

The configuration of the remote control units will be mainly influenced by weight savings. Studies made on the Boeing 707 show that 300 lb can be saved by using a number of dispersed remote control units instead of a central control panel [20]. The LTV multiplex system for the A-7 aircraft uses 32 remote control units. Since the orbiter and booster are considerably larger than the A-7 aircraft, there would be perhaps 64 or 128 remote control units. Since it is expected that two or even three master control units [18] would be used, the same degree of redundancy should be built into the remote control units.

The design of the power controllers is certainly critical in the development of a solid state multiplex power distribution system. Both low-level and high-level ac and dc loads must be switched, and some loads such as motors and heaters will draw large starting currents. Since the peak load on the booster will be approximately 48 kW, and both 115 Vdc and 115 Vac have been recommended, maximum ratings of solid

state devices are important. It has been reported that available diodes, transistors, silicon controlled rectifiers, and triacs (bidirectional triode thyristors) have sufficiently high ratings [16]. Because the power controller must perform many functions, its configuration is quite complicated. Some of the functions are:

1. Shape the switching characteristic for load and wire protection.
2. Timing.
3. Current limit.
4. Self protection.
5. Load protection.

A block diagram of a dc controller [21] is shown in Figure 26. The power transistor has two inputs. One is the drive circuit signal derived from the data bus; the other is a current limit signal derived from local current and voltage sensors. The current limit circuit also generates a time delay initiation pulse which starts the controller trip cycle. A typical trip cycle for a dc power controller [21] is shown in Figure 27, where it is seen that a current of from 130 to 300 percent of normal can be drawn for a short time without tripping the circuit. The short-circuit limit is determined by transistor and heat sink thermal characteristics and by wire protection needs. There are two problems with transistor power controllers: the present state-of-the-art is only about 100 Vdc, and the saturation voltage drop is about 0.5 Vdc.

Many of the requirements for dc power controllers also apply to ac power controllers. The switching device, however, would be a silicon controlled rectifier. Typically, SCR's have higher voltage and current ratings than transistors, but the saturation voltage drop across the device is about 1.5 Vdc. An SCR does not have the self protection feature of a transistor, which becomes unsaturated on a shorted load. For this reason, trip characteristics must be incorporated in the SCR control circuitry to protect the wiring. An additional factor with SCR's is that they can be switched near the zero crossing of the ac current wave. This greatly increases the average load current that can be switched. A zero crossing sensor would be built into the

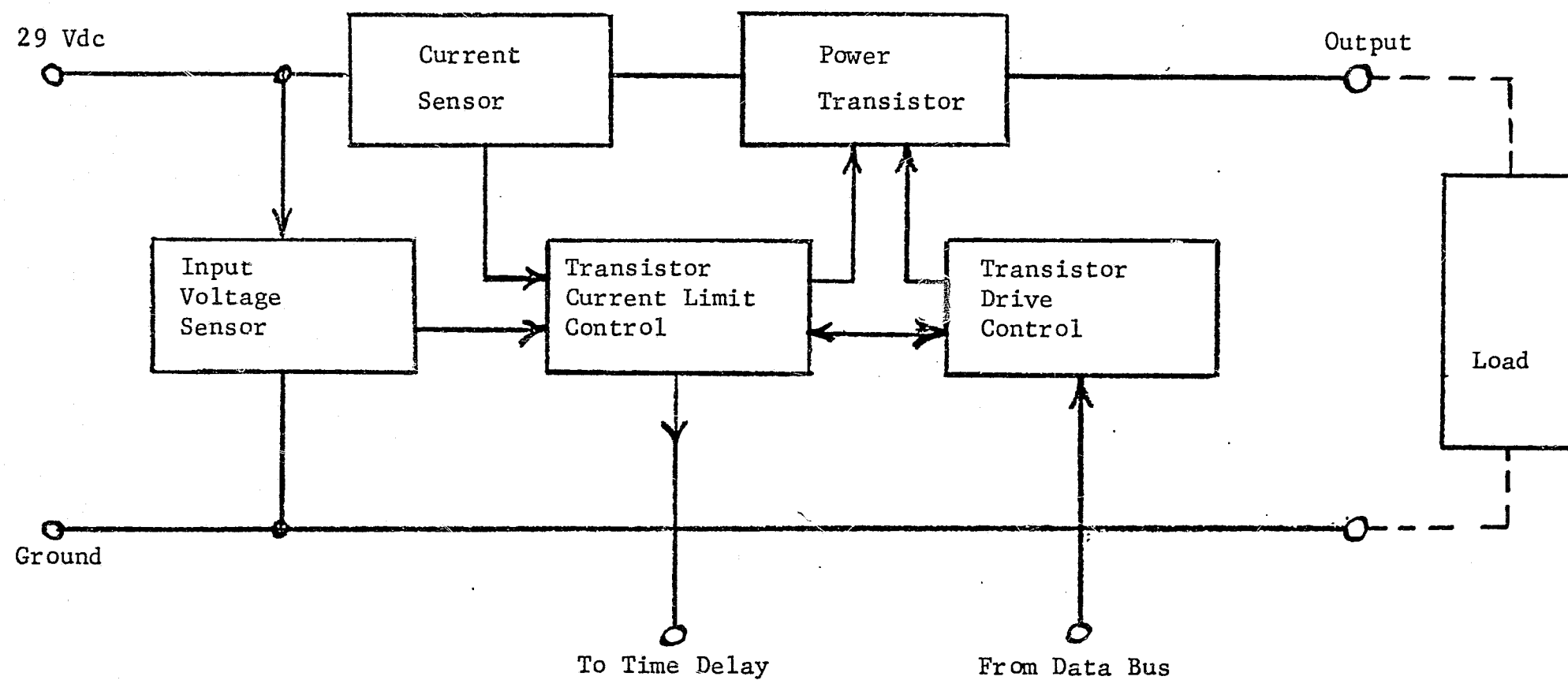


Figure 26. Dc Power Controller. From Westinghouse [21].

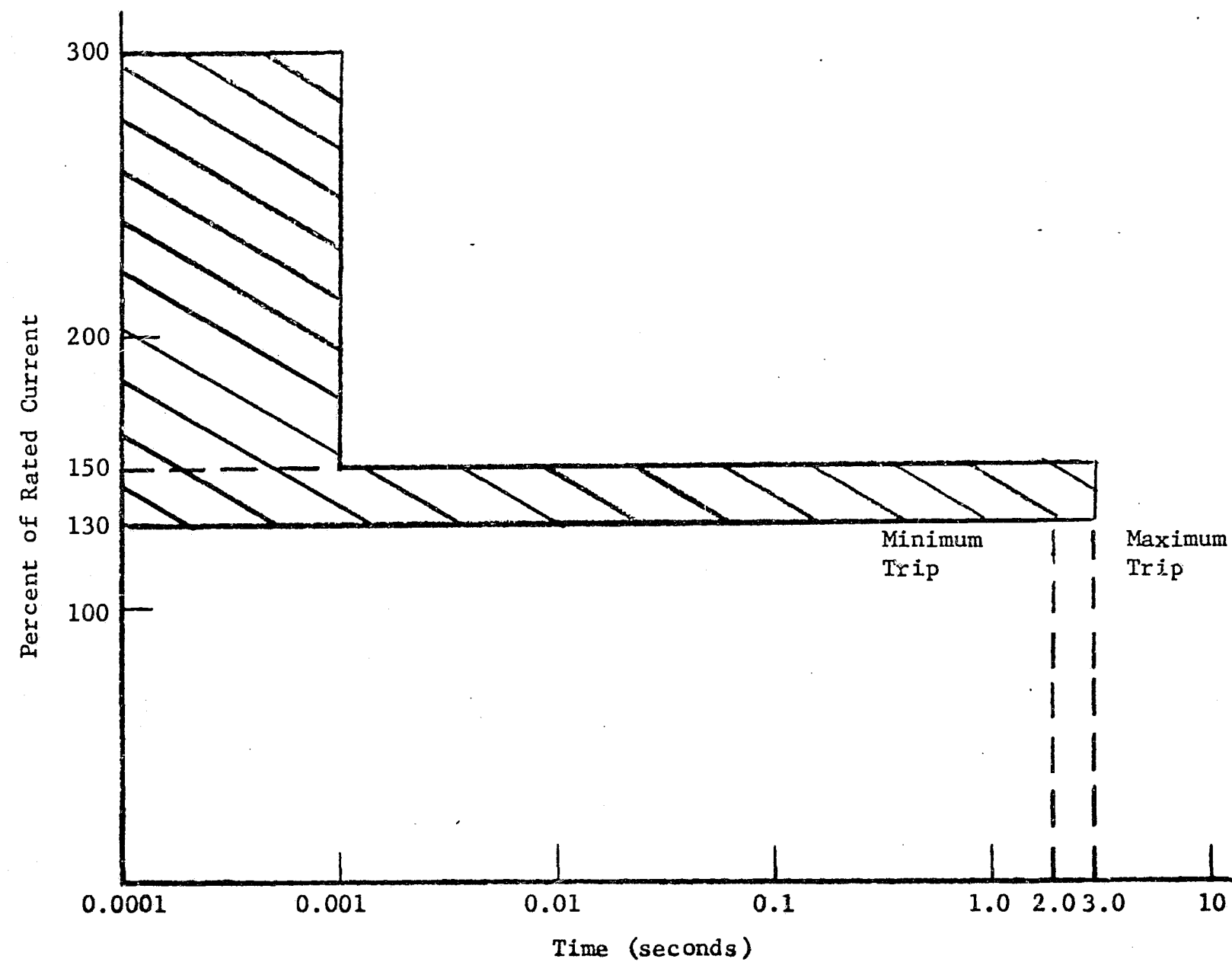


Figure 27. Trip Characteristics for Dc Power Controller.
From Westinghouse [21].

control circuit to provide this feature. A typical current limiting trip characteristic for an ac power controller [21] is shown in Figure 28. Note that the short-time overload capacity (10,000 percent) is thirty times the capacity of the dc controller.

In conclusion, it may be stated that devices are available which would permit the development of a solid state multiplex power distribution system for the Space Station, Space Base, and Space Shuttle.

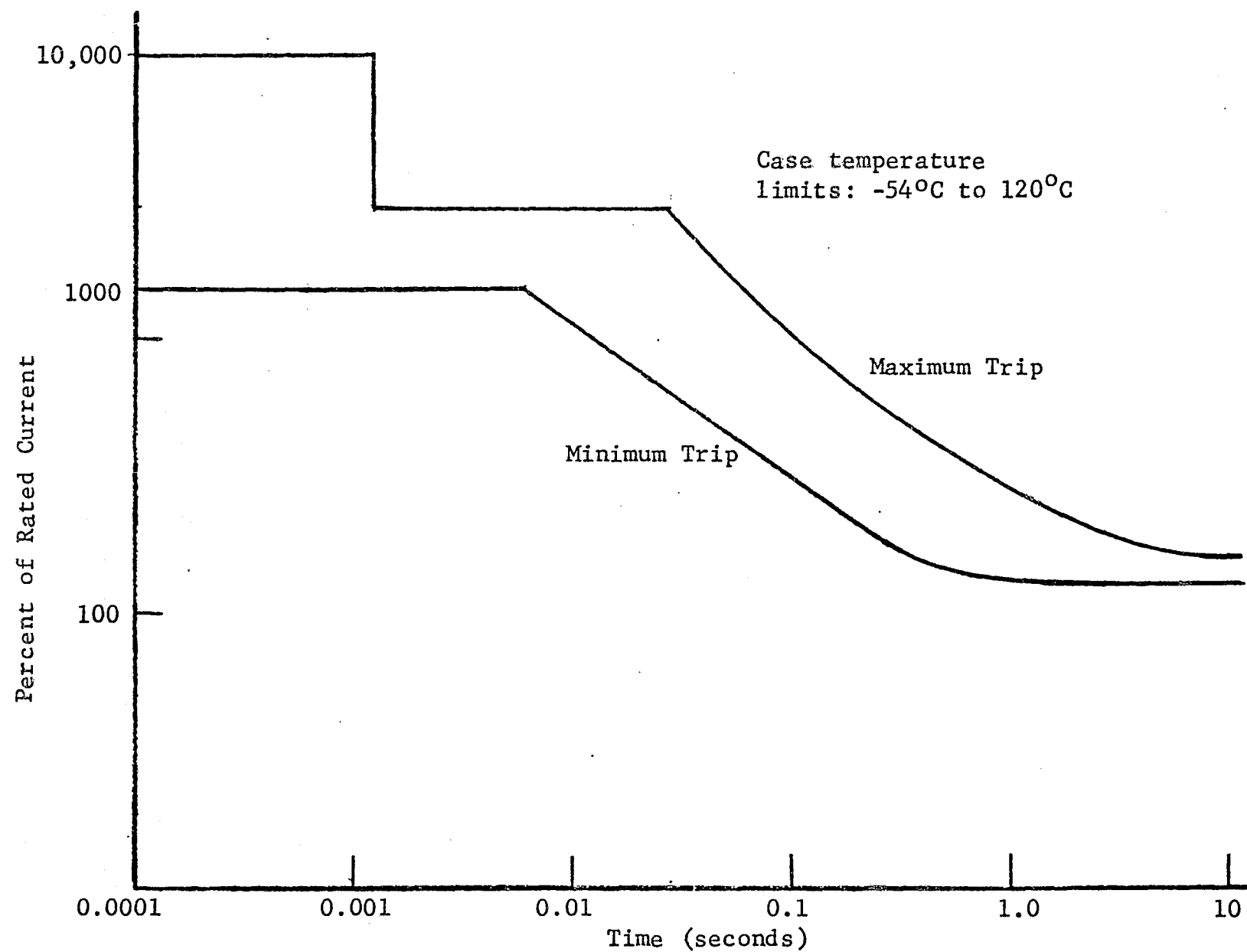


Figure 28. Trip Characteristics for Ac Power Controllers [19,21].

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V. MODELS FOR FUTURE SPACECRAFT POWER DISTRIBUTION SYSTEMS

The load requirements for the following vehicles have been discussed in Chapter II:

Space Station,
Space Base,
Booster, and
Orbiter.

This chapter will present block diagram models of power distribution systems for each of these vehicles based on the load requirements and physical configurations of the expected missions. For all models, multiplexing techniques will be used to obtain the outstanding advantages discussed in Chapter IV. Reference to Table X of Chapter II shows the load breakdown for each of the vehicles.

For the Space Station and Space Base the required amount of dc power is about 6 percent higher than the required amount of ac power while about 20 percent of the loads can be powered by either ac or dc. The booster requires 4 percent more dc than ac power while the orbiter requires 12 percent more dc power. From this tabulation it would appear that either an ac or a dc distribution system could be used in all four systems. However, raw, unconditioned power from an ac distribution system could power motors, lights and heaters. Dc power would have to be converted to ac for motors and fluorescent lights. Electronic loads, which include many values of dc voltages, and which also need filters to reduce interference, would require conversion equipment whether the distributed power is dc or ac.

From a standpoint of minimizing system weight it would appear that the highest possible voltage should be used for the distribution system to reduce the weight of wire and power conversion equipment. However, there are several constraints that modify this assumption. First, corona effects have been reported at fields as low as 100 Vdc/cm [3]. Any distribution system design should aim at avoiding the onset of corona. Second, smaller wire sizes are to be avoided because of low mechanical strength. Third, switching is more difficult at higher voltages, especially solid state switching of dc power. Fourth, most user equipment

is designed for operation with either 28 Vdc or 115 Vac, 400 Hz power, and therefore additional conversion equipment will be required if high voltage transmission is used. The next section will propose model distribution systems which conform to the above mentioned constraints.

A. Model Space Station Power Distribution System

The model power distribution system for the Space Station should consider the use of advanced techniques in the selection of voltage levels and distribution techniques. The system should allow for the use of multiplex control and solid state switching and possess a high degree of flexibility for expansion to the Space Base. Redundancy is necessary to insure that essential loads are maintained when a bus fails and that the correct sequencing of load shedding will preserve essential loads in the case of catastrophic failure. An additional requirement for the Space Station power distribution system is that it must be compatible with Space Base, since it is expected that the Space Station power distribution system will be the backup power for the Space Base power system. Also the orbiter power distribution system must interface with the Space Station in its docked configuration. Since the load power for the Space Station is about equally divided between dc and ac, both a dc and an ac distribution system have been modeled, and they are given as Figures 29 and 30.

The dc distribution system shown in Figure 29 includes a 110 Vdc primary bus where the outputs of a number of sources are cross tied for load sharing. The dc sources shown are solar arrays, and a transmission cable drop of 2 Vdc is anticipated.

Two distribution centers are shown in Figure 29. Power conversion and multiplex switching control would be accomplished at the centers which are cross tied for load sharing. Raw 110 Vdc power, regulated dc power at a number of voltage levels, and ac power would be distributed from the two centers. A charger/battery system would furnish backup and emergency power.

It is felt that the 110 Vdc bus voltage represents a good compromise. At higher voltages corona and solid state switching are problems.

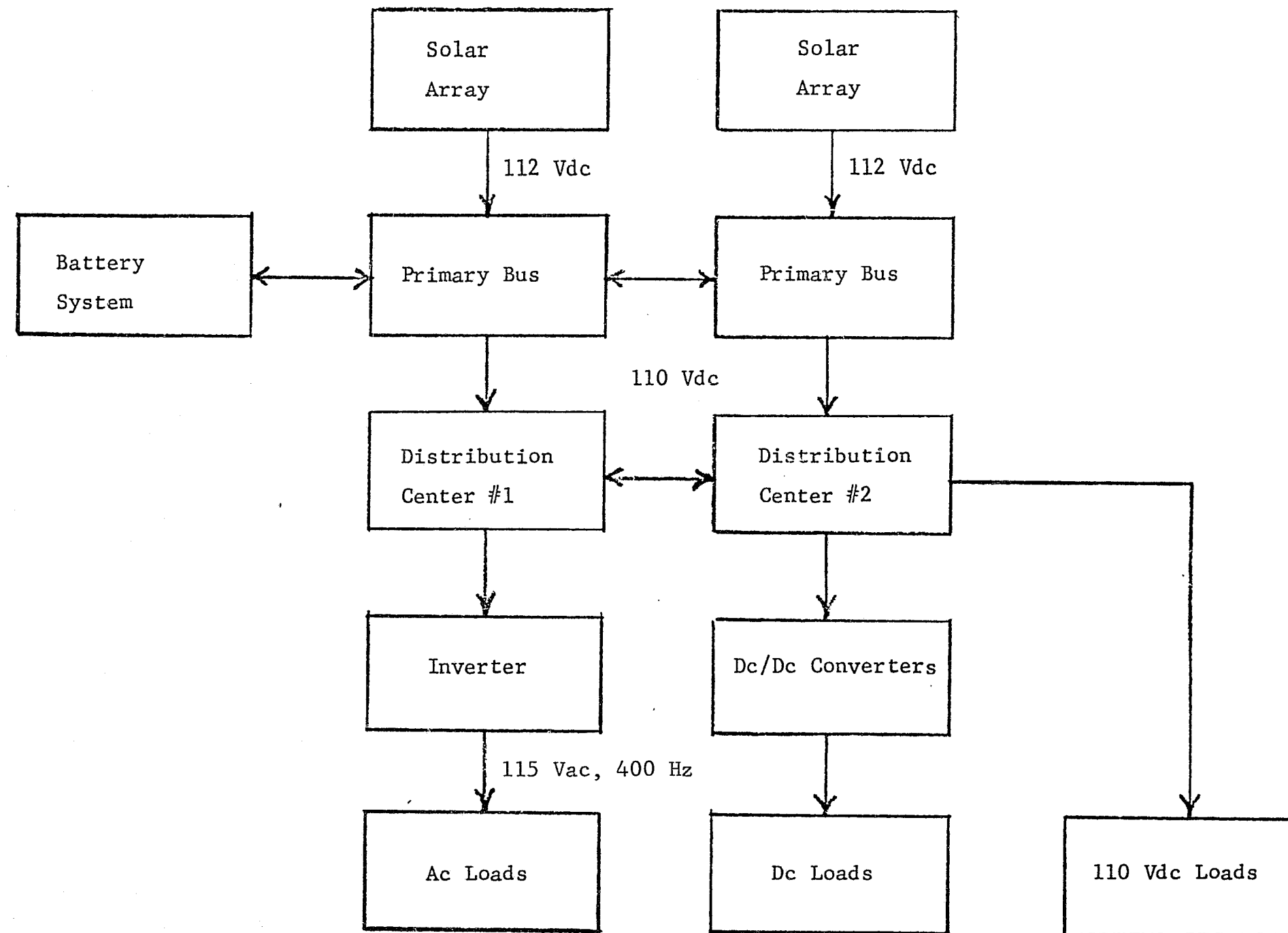


Figure 29. Space Station Power Distribution Model.

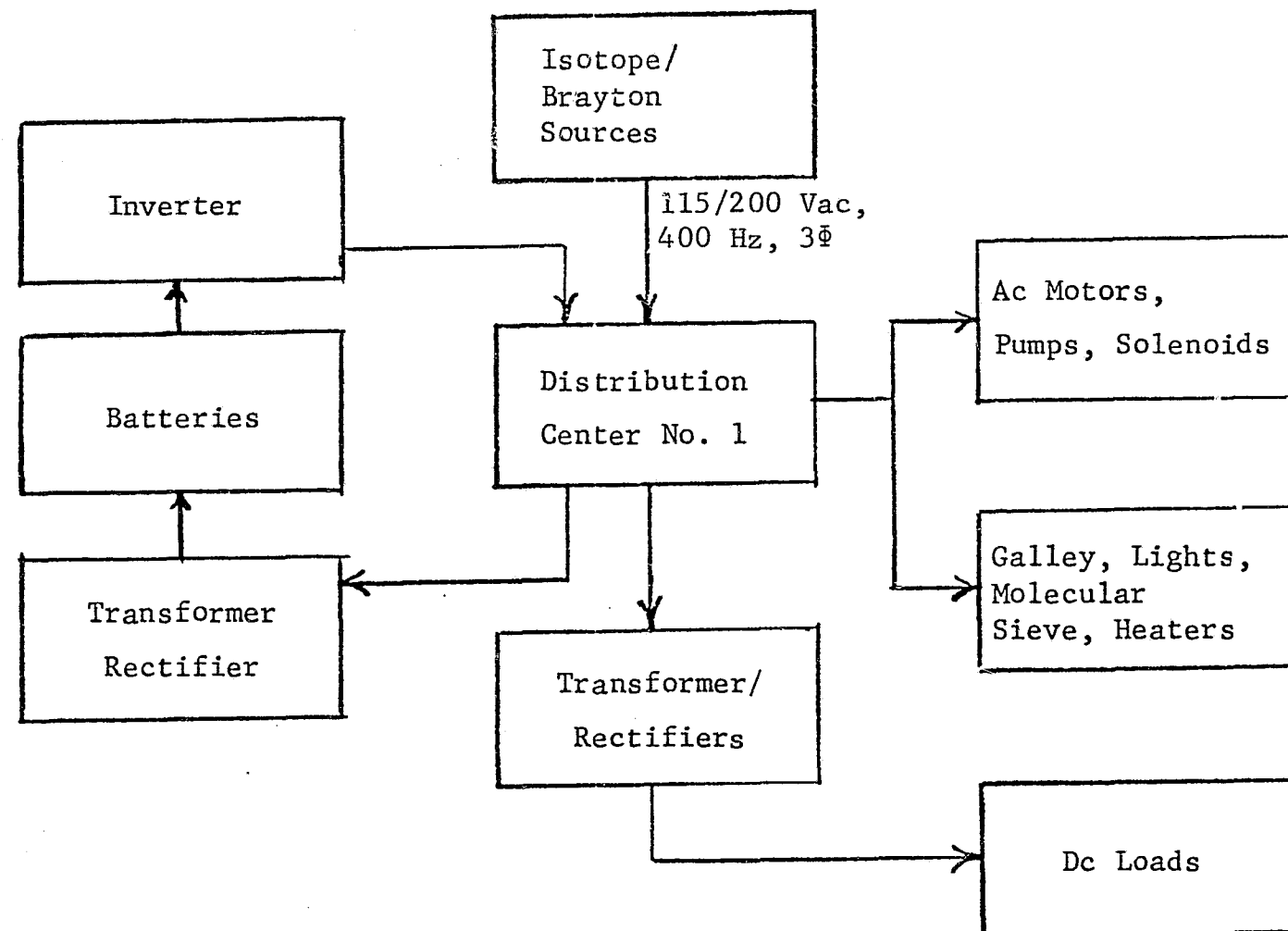


Figure 30. Recommended Space Station Power Distribution Model.

The voltages shown are high enough to obtain considerable weight savings in wire and some reduction in losses over 28 Vdc or 56 Vdc. Parallel operation is easily achieved, since it is only necessary to match voltage levels in the dc system. The 110 Vdc at the two distribution centers would be converted to 28 Vdc and to 115 Vac, 400 Hz for use by some of the system loads. Many loads would be supplied directly from the 110 Vdc bus without the use of conversion equipment. These loads include heating, lighting, molecular sieve and galley. When Figure 29 is compared with Figure 5, Chapter II, it is seen that the boom mounted inverter of Figure 5 has been eliminated. The system of Figure 29 more closely resembles Figure 21, but distribution is at 110 Vdc instead of 28 Vdc.

The ac distribution system block diagram is shown in Figure 30. The primary distribution voltage for this system is 115/200 Vac, 400 Hz, 3 ϕ . The voltage would have to be below the corona onset voltage, however; further investigation is recommended before the voltage is decided. Switching of ac power with SCR's is possible at 200 Vac, therefore this system readily adapts to multiplex, solid state, data bus control. It is not recommended that the sources be paralleled for the ac distribution system because of the tight tolerance on phasing generators for parallel operation. The power obtained from isotope/Brayton sources is transmitted to two distribution centers where it is conditioned and controlled. Many loads would be driven directly with raw power at the distribution voltage level, while loads requiring standard voltages would be driven from the outputs of regulated transformer rectifiers. The system could be adapted to the 1200 Hz alternator which is presently a leading candidate for Space Station, but the choice of frequency should be determined by a thorough trade off study. A 400 Hz ac system offers the advantage of good interface with the Space Base and with off-the-shelf loads, while the dc system affords good interface with the Space Shuttle, where the use of 112 Vdc has been recommended. Because of the inherently lower efficiency of the solar array/battery system, the isotope/Brayton, 115/200 Vac, 400 Hz, 3 ϕ system is recommended.

B. Model Space Base Power Distribution System

The Space Base is a larger vehicle than the Space Station. High powered nuclear reactors have been recommended by MDAC to supply the primary power. The reactors would be located several hundred feet from the distribution centers because of the radiation hazard to personnel. With these constraints the use of high voltage transmission would be desirable, although the use of pressurization or insulation and high voltage practice would be needed to prevent corona along the cables and at the connectors. The desire to use high voltage transmission is further enhanced by the large load (approximately 50 kW) that each nuclear reactor must supply. The transmission voltage recommended was 115/200 Vac, 400 Hz, 3 ϕ . A block diagram of the MDAC Space Base power distribution system is shown in Figure 22, Chapter IV. If parallel operation is necessary to preserve essential loads and to insure uninterrupted power flow to such critical loads as the computer and the flight control loads, it would best be done at the low voltage buses. Line regulators might be required to maintain the low voltage buses within tolerance to permit paralleling. Solid state switching could be employed on this system on the 115 Vac, 200 Vac, and 28 Vdc buses. Electromechanical relays would be required on higher voltage buses.

Although only reactor systems were described by MDAC, a clear case was not made for the reactor/Brayton source. An immense weight of boom and shielding would be required. Furthermore, the excellent operating characteristics of a Brayton system require an operating temperature of about 1600^oF, whereas only about 1100^oF has been obtained in a test reactor [22]. Integration of Space Base and Space Station systems would be much easier if the power sources for both are isotope/Brayton. It may be that a 50 kWe isotope/Brayton system is realizable. The projected supply of fuel appears to be adequate for isotope/Brayton power sources for Space Base. (See Figure 11, Chapter II.) The 400 feet of transmission cable needed for the nuclear/Brayton system would not be needed for isotope/Brayton sources in Space Base.

C. Model Orbiter Power Distribution System

The orbiter mission profile is shown in Figure 1 and its load profiles are shown in Figures 3 and 4, Chapter II. As seen in its mission profile the orbiter operates both as a maneuvering spacecraft and an aircraft with a mission duration of seven days and therefore it has both spacecraft and aircraft loads. Since the load breakdown on the orbiter is 56 percent dc power and 44 percent ac power, there might be some advantage in selecting a predominantly dc system. Another characteristic of the orbiter load profile is the very high transient load that requires the use of a source with good transient overload capacity and a distribution system with good transient response. Reference to Figure 4 shows that the average dc load is about 4 kW with transient peak loads of about 13 kW when the attitude control and orbital maneuver systems are actuated. The ac load requires about 3 kW average power and 12 kW peak power. As mentioned in Chapter IV, the criteria that should be stressed in the development of the orbiter power system are:

1. Triple redundancy.
2. Weight penalty of \$25,000/lb.
3. Compatibility with Space Station and booster.
4. Use of multiplexing and solid state techniques.

The orbiter is uniquely suited to the above criteria because its load profile and load requirements are relatively simple and its restricted size lends itself to advanced power system development.

Because of its dual mode of operation the recommended power distribution system for the orbiter is the dual system (ac and dc) shown in Figure 24, Chapter IV. As discussed in Chapter IV a dc to dc converter to obtain the 115 Vdc from the fuel cell system is recommended. All of the spacecraft loads are supplied by this source with the triple redundancy supplied by additional sources. This dc system should integrate easily with Space Station since 115 Vdc at the interface, supplied by Space Station, could power the orbiter loads. The aircraft mode ac loads would be supplied by three alternators with one alternator in use and two on standby. The aircraft ac loads include main propulsion engines, lights, heating and de-icing. The ac alternators could be paralleled to obtain better load utilization; however, the loads can

tolerate a temporary loss of power so that it may not be necessary to operate in parallel with the required complicated switching circuitry. The proposed system lends itself to the use of multiplex techniques and solid state switching since transistor switching can be used on the 115 Vdc and SCR's on the ac system.

D. Model Booster Power Distribution System

The booster as part of the Space Shuttle transports the orbiter to an altitude of 186,000 feet and then flies back to earth. The total mission time is about three hours. A load profile for the booster has been given in Figure 2. It is seen that large transient loads must be supplied. The transient loads are the main propulsion engines and the air breathing engines. The recommended power distribution system for the booster is shown in Figure 23, Chapter IV. Only one change would be recommended: 115 Vdc instead of 28 Vdc from the transformer rectifiers. The main propulsion engines and the air breathing engines would be driven directly from the alternators while the dc power for the avionics would be supplied by the 115 Vdc distribution bus. The power for environment control and life support would be supplied at 115 Vdc and 115/200 Vac, 400 Hz. Most of the avionics and EC/LS loads require dc power. The gyros and other sensitive ac loads would be supplied by a sine wave inverter with a 115 Vdc input.

Decentralized bus systems would be used where the user voltages (28 Vdc, 115 Vac, 6 Vdc etc.,) are generated at the local load center. A trade off study has shown that the decentralized bus system produced a large weight savings [12]. It is felt that 115 Vdc solid state switching will be available in time to allow multiplexing of the recommended power distribution system with solid state switching of the 115 Vdc loads. Switching of the 115/200 Vac would be accomplished with silicon controlled rectifiers.

Because eleven main propulsion engines must be started and cut off by electrical power (800 watts/engine), it is highly desirable to study the effect on the system dynamic response when these transients occur, and the study must be from a dynamic stability and a minimization of power point of view.

VI. CONCLUSIONS AND RECOMMENDATIONS

The purpose of this study was to help improve planning for design of future spacecraft power distribution systems. In particular, the power distribution systems of the Space Station, Space Base, and Space Shuttle were studied in an effort to devise methods of incorporating new technology into these systems. It is appropriate to make such studies in the earliest stages of conceptual design before such power systems are implemented in order to be able to take full advantage of promising techniques. From very preliminary investigations, it appears that multiplexing and solid state technology offer the possibility of improvement for future spacecraft. Solid state devices would replace electromechanical relays for switching. Unfortunately, solid state devices are limited in voltage, power, radiation, and transient overload characteristics relative to their electromechanical counterparts. These limitations require development to improve device characteristics. Close attention to the dynamic stability of the power distribution system will be needed to protect these sensitive devices. The development will be costly and must be justified in appropriate trade off studies.

The approach that was used in this power distribution study was first to review the power sources and power distribution systems suggested by others and then to recommend model systems. Study of the power systems recommended by others was necessary to establish the present state-of-the-art. It was seen that prior efforts were directed toward power sources, and the principal criterion in the selection of power distribution systems was weight, with a limited effort on such factors as development cost, safety, and reliability. The prior studies started with the selection of a power source and proceeded to the selection of a power distribution system that would fit the power source. Little attention was placed on fitting the distribution system to the loads except by using conversion equipment, and the very important problems of controlling high power loads with sensitive solid state devices were not discussed.

This report recommends models for power distribution systems based on load requirements and the use of multiplexing and solid state switching

devices. The most obvious load breakdown was the categorizing of loads as requiring dc and ac voltage. It was found that the loads were about equally divided between dc and ac so that either dc or ac distribution could be used. Most of the dc loads, however, are electronic devices which require various voltage levels. A trade off study is recommended to compare dc to dc converters with transformer rectifiers; results of the study would influence the choice of distribution power form. It was also determined that about 20 percent of Space Station and Space Base loads can be powered directly from the distribution bus without power conversion.

The extremely important problems associated with supplying critical and sensitive loads from a power distribution system that also supplies high power, large transient loads are recommended for further study. One proposed distribution system recommends cross tying of dc source buses because, for this case, only the voltages of the sources need to be equal. Another proposed system recommends ac distribution, with the buses from individual sources not tied together since paralleling is more complicated because of the need to match the phases of the various sources. For the ac power distribution system it is, however, recommended that the low voltage buses be tied together where control loads that cannot be interrupted are serviced. A problem in ac distribution systems which needs further study is the isolation of sensitive solid state devices from large transients generated by motors, etc. Another problem is the difficulty of paralleling ac sources.

It is recommended that all loads (and particularly large loads such as heating, lighting, and motors) be studied in more detail to determine isolation techniques to prevent large transients from being impressed on solid state devices. The power distribution system dynamics must be analyzed for workable performance. For Space Shuttle it is recommended that the main propulsion and air breathing engine load requirements be analyzed in an effort to reduce peak load requirements and large transients.

Two alternative distribution systems are modeled for Space Station. The first would be a 110 Vdc system, with conversion of some power to 115 Vac, 400 Hz, 3 ϕ , and to 28 Vdc. Paralleling of sources would be feasible. The second system would be 115/200 Vac with conversion of some power to

115 Vdc and 28 Vdc. The latter distribution system parameters are recommended for Space Station. The recommended power source is isotope/Brayton, with a 400 Hz, 3 ϕ alternator.

A 115/200 Vac, 400 Hz, 3 ϕ distribution system is also recommended for Space Base. A study of the feasibility of an isotope/Brayton source, and a trade off comparison of isotope/Brayton versus nuclear/Brayton are urged.

A dual 115 Vdc and 115/200 Vac, 400 Hz, 3 ϕ distribution system is recommended for the orbiter, and a 115/200 Vac, 400 Hz, 3 ϕ distribution system is recommended for the booster.

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