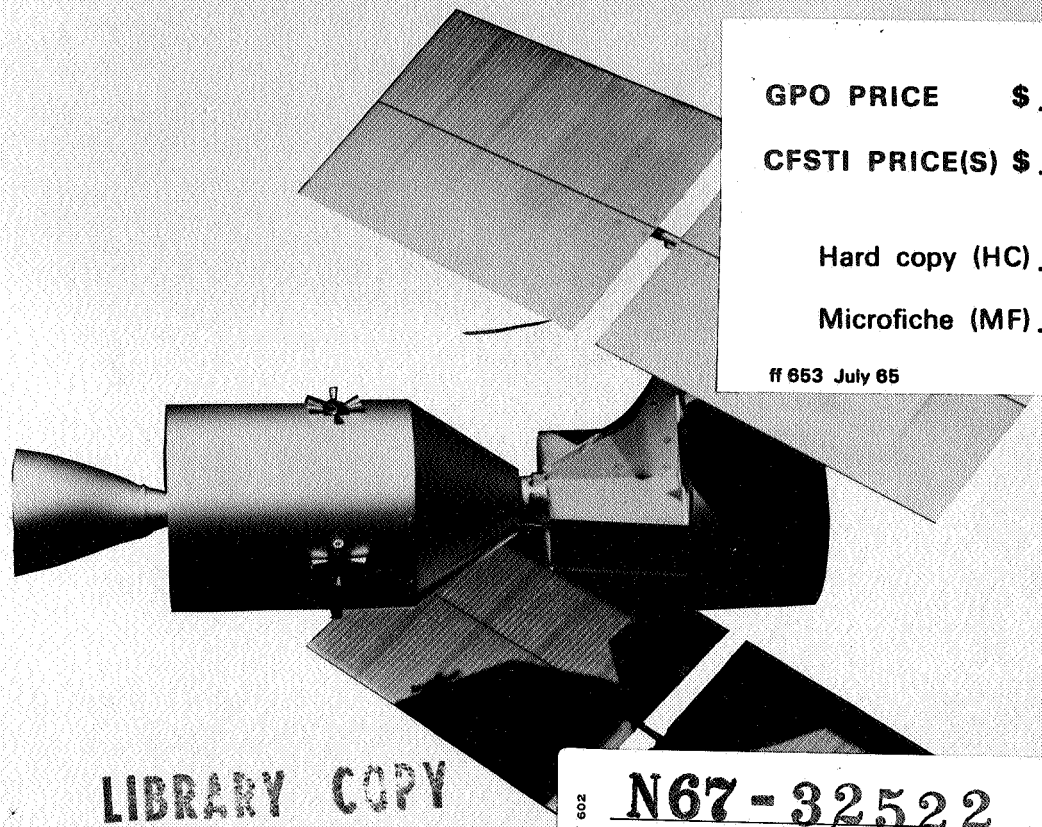
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VOLUME I : PROGRAM SUMMARY



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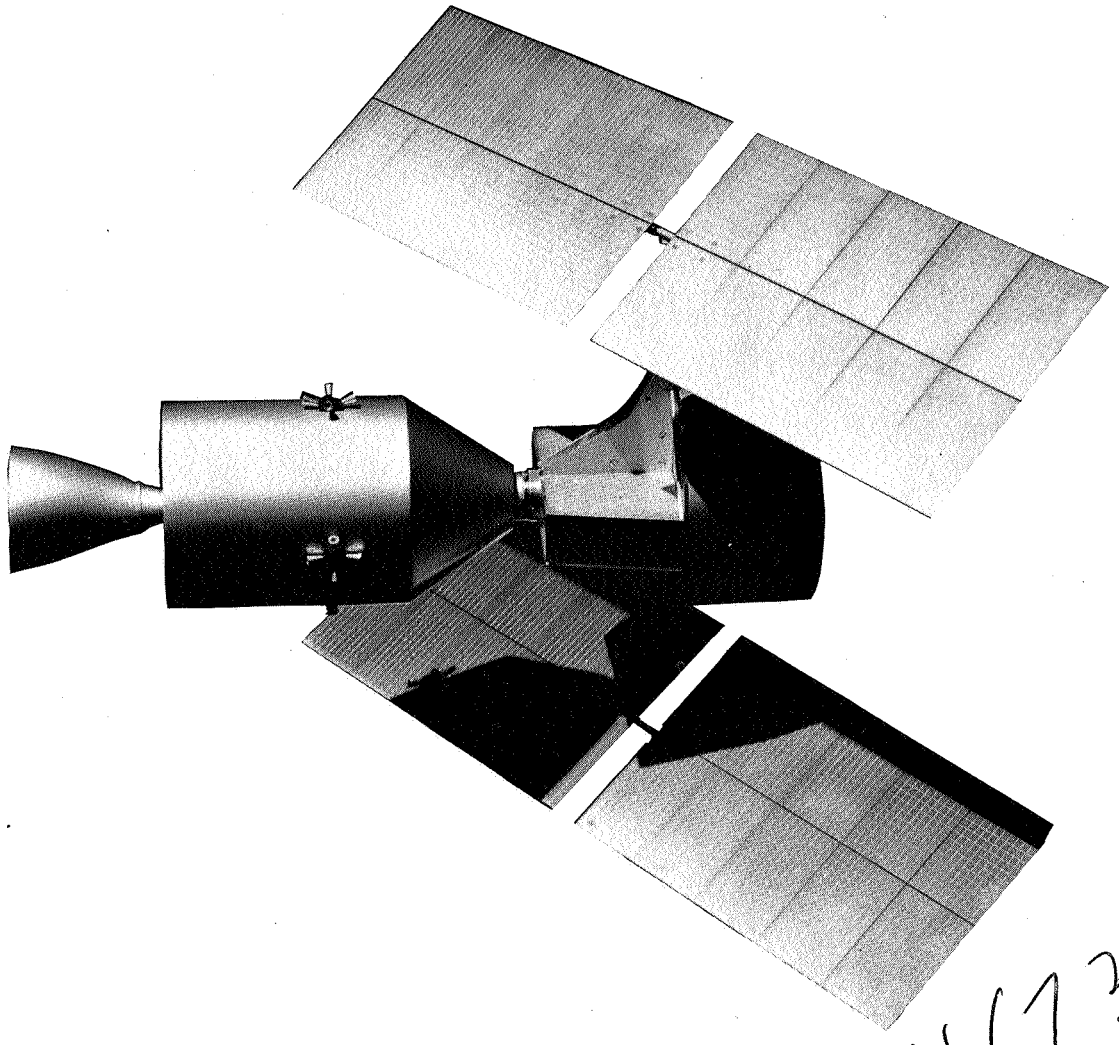
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PRINCETON, NEW JERSEY

-R-3155

June 9, 1967

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FOREWORD

This is Volume I of the final engineering report on the "Manned Mission Photovoltaic Power System Study". This report was prepared in three volumes by the Astro-Electronics Division of the Radio Corporation of America, Princeton, New Jersey, under Contract NAS 9-5266 for the Propulsion and Power Division, Manned Spacecraft Center, National Aeronautics and Space Administration, Houston, Texas. This volume contains a summary of the study program. Volume II includes a detailed technical discussion, and Volume III includes documentation that supports the technical discussion.

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SECTION I

INTRODUCTION

SECTION I

INTRODUCTION

A. REVIEW AND SUMMARY

This report describes a study to develop a conceptual design of a photovoltaic power system for a manned orbiting laboratory module. The effort described herein was performed between April 20, 1966 and January 20, 1967, and concluded with the delivery of a computer program for sizing photovoltaic power systems as a function of mission parameters and submittal of this detailed report of the technical tasks performed.

This final report comprises three volumes: Volume I, "Program Summary", consists of Sections I through V, which summarize the total study effort, conclusions, and recommendations: Volume II, "Technical Discussion", consists of Sections VI through X, which contain all of the detailed technical analyses performed during the course of the study and the justification for the conclusions and recommendations summarized in Volume I and Volume III, "Supporting Documentation."

Volume I includes the Introduction (Section I), Summary of Results (Section II), Conclusions (Section III), Recommendations (Section IV), and Development Plan (Section V). This volume constitutes a summary final report in that it presents, in summary form, the objectives, constraints, conclusions and recommendations of the study which led to the preparation of a development plan for the design and fabrication of a manned orbiting photovoltaic power system.

Volume II contains the detailed technical studies and analyses performed during the study. Section VI is the overall Power System Analysis. Section VII is a detailed Mechanical System Analysis of the integrated power system (IPS) approach, while Section VIII is a Mechanical System Analysis of the power system module (PSM) approach. An Array Orientation Analysis is presented in Section IX, and Section X is the overall Electrical System Analysis.

Volume III includes a collection of detailed calculations and studies performed in support of the major study analyses.

B. PROGRAM GOALS

Although the "ground rules" and mission definition for this study were modified several times during the course of the contract (see Table I-1), the primary purpose has remained unaltered, viz., "to define the major problems associated with large-area solar-array power systems for manned orbiting missions and provide recommendations for solution of or methods of resolution of these problems".

A secondary but nonetheless important goal was the performance of adequate power-system studies and tradeoff analyses to enable preparation of a development plan that could be

implemented in logical steps consisting of: (1) detailed in-depth study tasks, (2) component design, (3) prototype fabrication and test, and (4) flight-unit fabrication and test.

C. PROGRAM CHRONOLOGY

Contract NAS 9-5266 was initiated on April 20, 1966, as a six-month "Design and Integration Study of a Photovoltaic Power System". Deliverable items were an energy-balance type of computer program and a final technical study report. A summary of the major milestones during performance of this contract is presented in Table I-1.

TABLE I-1. SIGNIFICANT PROGRAM MILESTONES

	Study Program		Mission Characteristics					
	Type	Period of Effort	Launch Vehicle	Orbit	Orbiting Vehicle	Mission Duration	Power System Approach	Power Level[6]
Initial Contract (4/66)	Design & Integration	6 mo.[4]	Saturn *C*-5	one	AAP[7] Lab module	Less than 90 days	IPS	5 kW nom
Revised Work Statement (7/66)	Conceptual	9 mo.[4]	SIB & SV(3)	three	MMSS[1] & Apollo C&SM[8]	1 yr	IPS[2]	5 to 10 kW
Technical Direction No. 1 (10/66) .(Special Task)	Conceptual	(2 wks)[5]	SIVB	two	MMSS	1 yr	IPS & PSM	max.
Technical Direction No. 2 (11/66) (Program Redirection)	Conceptual	(6 wks)[5]	SIB	three	MMSS & Apollo C&SM	1 yr	PSM[3]	5 kW

NOTES: 1 - Modular Multipurpose Space Station (See Reference 1).
2 - Integrated Power System.
3 - Power System Module.
4 - Total Contract Period.
5 - Time available for performance of task.
6 - Nighttime load power requirement.
7 - Apollo Applications Program.
8 - Command and Service Module.

The initial level of effort was concerned with the development of a detailed study-task and manpower-implementation plan for performing the tasks outlined in the contractual work statement. This plan was submitted to the NASA Technical Officer on April 26, 1966. At this time, the designated orbiting vehicle was changed from the Apollo Applications Program Laboratory Module to the Modular Multipurpose Space Station (MMSS, Reference 1) and Apollo Command and Service Module, with six basic mission/vehicle combinations as outlined in Table 1-2. Based on this vehicle redefinition, three primary missions were established for the study (see Table 1-3). The mission goals and three orbiting-vehicle configurations (one room, two compartments, and six rooms) were defined and the Reference 1 report was provided by NASA as a baseline description.

TABLE 1-2. GENERAL MISSION REQUIREMENTS

Vehicle Configuration / Mission Parameters	One Room (Eng. Room)	Two-Compartment			6-Room	Undefined
		Low Incl.	Polar	Synch		
Altitude, nmi	200	200	200	19340	200	200
Inclination, degrees	28.5	28.5	90	30	28.5	28.5, 90
Crew Size	3	6	3 to 6	3 to 6	6 to 9	6 to 12
Launch Vehicle	SIB (Note 4)	SIB	SV(3) (Note 5)	SV(3)	SIB	SV(2) (Note 6)
	135 days	1 yr.	1 yr.	1 yr.	1 to 5 yr.	1 to 5 yr.
Power Level (See Note 1)	5-10	5-10	5-10	5-10	5-10	5-10
Resupply Interval	45	45	45	45	See Note 2	See Note 3
Launch Year	1969	1972	1972	1972	1972	1970

NOTES:

1 - "5-10" means 5 kW with capability of upgrading to 10 kW. System design should include both levels.
2 - 45-day resupply interval during first year, 90-day resupply interval during second year. 180-day resupply interval during subsequent years.
3 - Same as Note 2 except that a no-resupply case must also be considered for the first year.
4 - Saturn IB.
5 - Saturn V, 3-stage.
6 - Saturn V, 2-stage.

Parameter	Characteristics		
Altitude, nmi	200	200	19,340
Inclination, degrees	28.5	90	30
Mission Duration	1 yr.	1 yr.	1 yr.
Resupply Interval	45 days	45 days	45 days
Launch Site	AMR	AMR	AMR
Launch Date	1970	1972	1972
Launch Vehicle	SIB	SV(3)	SV(3)
Vehicle Attitude	Longitudinal axis always tangent to orbital velocity vector.		

As a result of detail-definition conferences with the NASA Technical Officer on May 25 and 26, 1966 and with the Technical Officer and the Division Chief of the NASA Advanced Spacecraft Technology Division on June 3, 1966, the program was changed from a design study to a conceptual study of a modular, integrated power system for a manned laboratory with the MMSS as the orbiting vehicle and the missions as shown in Table 1-3. A revised work statement (see Appendix G) was prepared and RCA developed a revised program plan to place greater emphasis on parametric and tradeoff studies involving mission- and vehicle-dependent parameters and including a conceptual design optimized for the vehicle and the three missions.

Approval to proceed was received from the Technical Officer on July 6, 1966, following a review of the revised program-plan details. In addition, because of the extensive replanning required of RCA, the program period of performance was extended by three months. (See Table I-1.)

As work progressed, it became apparent that the Integrated Power System (IPS)* approach required a more detailed definition of the missions and of the spacecraft (MMSS) and its operational modes than was possible within the time span of the contract. Furthermore, launch-vehicle shroud modification was necessary to provide stowage for the large solar array required to support the maximum load-power profile.

These and other pertinent points were emphasized during the Mid-Contract Review at the Manned Spacecraft Center on October 9 and 10, 1966. As a result of this review, the Technical Officer directed RCA to conduct a brief analytical investigation to provide preliminary

*An Integrated Power System is defined as a power system whose components become an integral part of the mission vehicle. A Power System Module (PSM), on the other hand, is a separate power-system unit that can be mated to a variety of mission vehicles.

definition of the electrical-power-system sizes that could be physically incorporated into four specific vehicle launch configurations. (The details of this investigation are presented in Appendix A.)

On November 1, 1966, the results of the investigation were reviewed with the Technical Officer, who directed that effort on the IPS approach be curtailed and that the PSM concept be pursued in as much depth and detail as possible within the remaining time and funds. It should be pointed out that only a Mechanical Systems effort was involved in the PSM study and analyses, since the Electrical Systems analyses previously performed were still generally applicable. Section VII of this report presents the work relating to the IPS approach while Section VIII presents the PSM concept analyses. Sections IX and X are generally applicable to both IPS and PSM concepts.

The Final Program Review was held on February 2, 1967 at the Manned Spacecraft Center.

D. GUIDELINES AND REQUIREMENTS

Two separate concepts have been studied for a manned orbital mission power system: an IPS and a PSM. The basic "ground rules" were similar for both, although there were some significant differences. These differences are almost wholly within the mechanical constraints. Therefore, the ground rules stated in Paragraphs I.D. 1 and I.D. 2 apply to both concepts. Paragraph I.D. 3 pertains to the IPS alone, while Paragraph I.D.4 applies only to the PSM concept.

1. General Ground Rules

The launch period for flight equipment is defined as 1970 to 1972. It is thus apparent that development of the power system must begin in 1967 to 1968 (assuming that three to four years will be required to bring the power system to the flight-qualified equipment stage). This conceptual study must, therefore, use present state-of-the-art* methods, materials, and techniques wherever possible.

Because detailed mission requirements have not been firmly established, a modular power system must be provided, i.e., a system whose capability may be varied by adding or deleting modular sections of major functional elements.

Three specific circular orbits have been selected and the launch vehicle defined as shown in Table 1-4 and Figure I-1.

TABLE 1-4. ORBIT PARAMETERS

Altitude (nmi)	Inclination (degrees)	Launch Vehicle
200	28.5	SIB
200	90	SV(3)
19,340	30	SV(3)

*State of the art is defined to mean that no technological breakthrough is required to achieve a demonstrated capability.

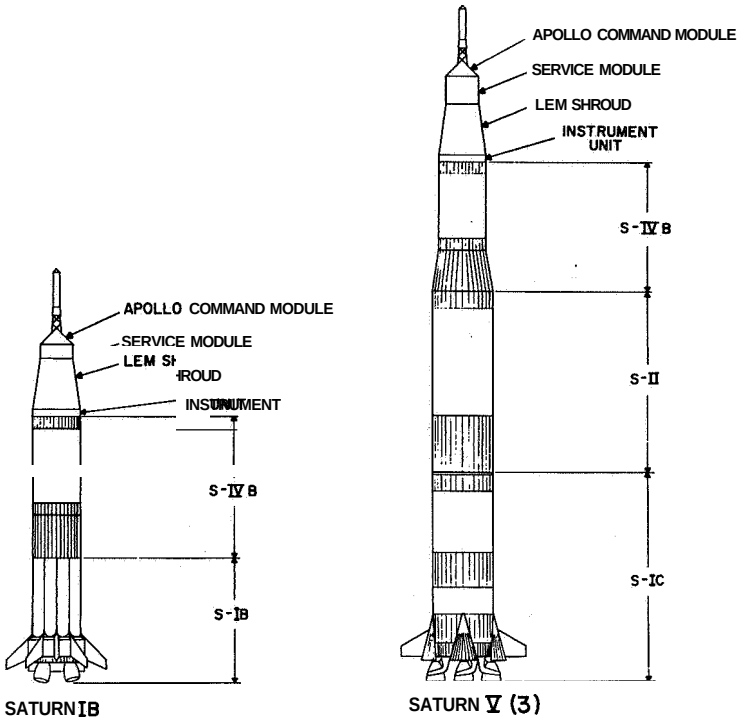


Figure I-1. Launch Vehicle Configurations

All missions are to be one year in duration with no resupply considered for the PSM concept. However, resupply may be utilized in 45-day increments for the IPS concept.

2. Electrical Ground Rules

Typical vehicle subsystems that will require electrical power either continuously or periodically have been defined as follows:

(1) EC/LSS: Environmental Control and Life Support System consisting of oxygen regeneration, waste processing, MMSS pressurization, etc.

(2) Comm: Communications System consisting of voice and data transmission facilities, both primary and secondary.

(3) N/G: Navigation and Guidance System consisting of orbit parameter computers, star trackers, etc. .

(4) S & C: Stabilization and Control System consisting of vehicle orientation and attitude sensors, etc.

(5) Prop: Auxiliary Propulsion System; not defined at present.

(6) HKPG: Vehicle "housekeeping" requirements consisting of displays, data management and signal processing, lighting, TV, maintenance, etc.

(7) EXP: Experiments, consisting of the scientific programmable experiments, i.e., the primary purpose of the mission.

A detailed breakdown for each of these subsystems is contained in Paragraph X.B.2.b. The load-power profile used for this study is a logical combination of the specific subsystem requirements as shown in Figure 1-2.

For the PSM study, a single load-power profile was utilized with P_{LD}* equal to 8.22 kW and P_{LN} equal to 5 kW.

Since this is a manned mission, the astronauts may be required to perform routine monitoring of performance and periodic preventive maintenance, provided the average daily time requirements are relatively small. Emergency maintenance may be performed at any time without restriction.

3. Mechanical Ground Rules (IPS)

For the Integrated Power System concept of this study, the mission vehicle is the MMSS (Reference I-1) and the Apollo Command and Service Module. The launch configuration is shown in Figure 1-3, where the power-system batteries and electronic-control circuitry will be housed within one of the MMSS compartments and the solar array will be stowed in the voidwithin the shroud. Modification of the shroud and truss are permitted, if necessary to provide adequate stowage volume for the solar array.

The orbital vehicle configuration is shown in Figure 1-4. The vehicle attitude is controlled by the Apollo Service Module Reaction Control System (RCS) so that the vehicle roll axis is coincident with the velocity vector and the yaw axis is coincident with the local vertical at all times. The peripheral docking ports must be unobstructed during orbital operation to permit resupply vehicles access to the MMSS. The Apollo Service Module SPS* engine will provide propulsion for maneuvering and orbital changes.

Within the MMSS, heat sinks (whose temperature range will be 21°C to 32°C) will be available for battery and control electronics heating and cooling. These heat sinks will have adequate capacity to handle the power-system heat requirements.

*

P_{LD}: Daytime power load
P_{LN}: Nighttime power load
SPS: Service Propulsion System

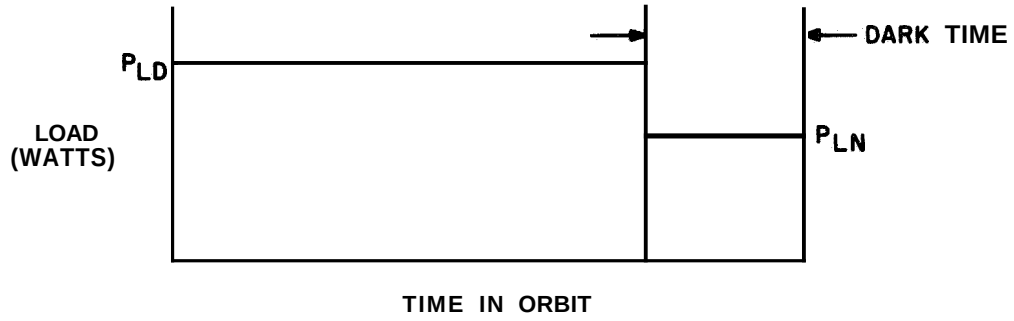


Figure I-2. Load Requirements

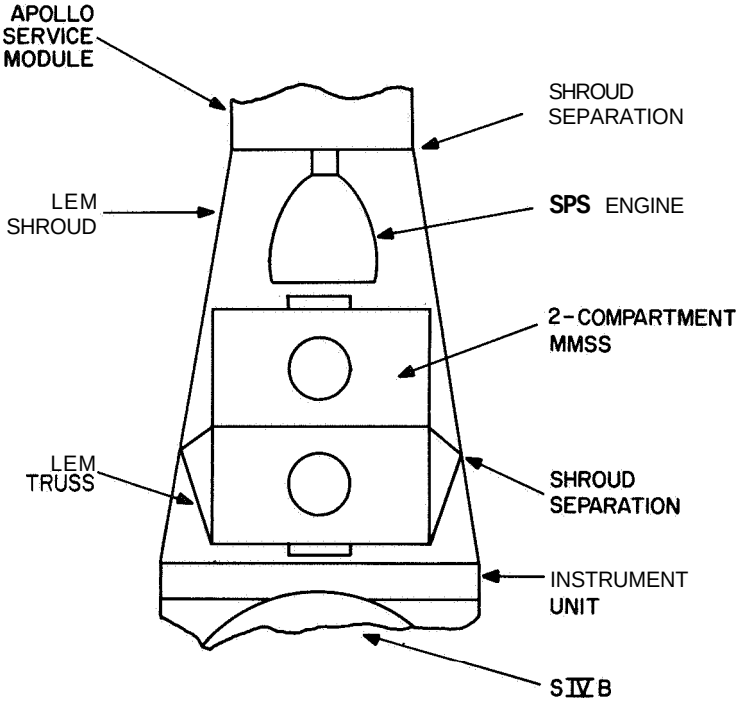
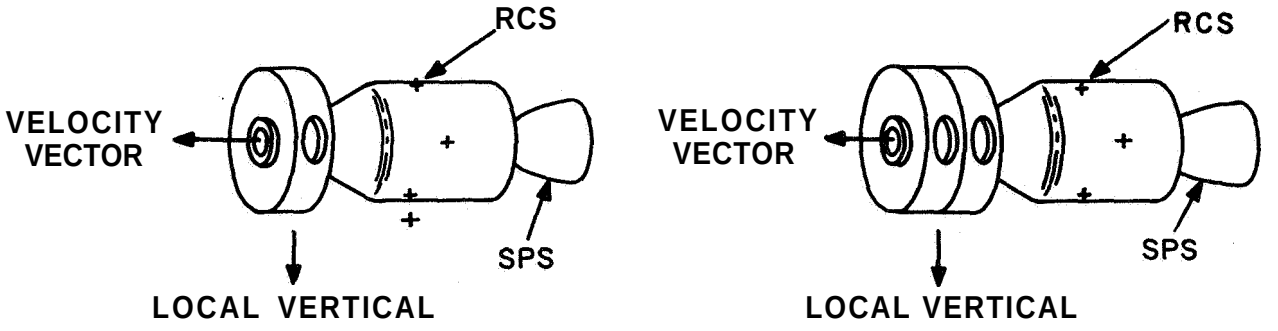


Figure I-3. IPS Launch Configuration



ONE-COMPARTMENT CONFIGURATION TWO-COMPARTMENT CONFIGURATION

Figure I-4. Orbital Configurations

4. Mechanical Ground Rules (PSM)

A different set of ground rules has been established for the Power System Module concept. The basic PSM structure will be as shown in Figure I-5. This structure will house the solar-array (during launch), the batteries, and the control electronics of the power system. The PSM must provide a thermal control system for maintaining the batteries and control electronics within permissible operating temperature limits.

The docking tunnel (or airlock) is not defined in any detail but it may have cryogenic tankage and other equipment associated with it. Therefore, space must be left available for this requirement.

For launch, the PSM structure will be mounted in the LEM shroud with a single-compartment MMSS, as shown in Figure 1-6. The PSM launch support legs will mate with the LEM truss at the shroud separation points. No shroud or truss modifications are permitted.

In orbit, the vehicle configuration will be as shown in Figure 1-7. The vehicle attitude is maintained in the same manner as for the IPS, although vehicle roll may be considered if necessary. No peripheral docking ports are to be considered for this concept.

Retraction of the solar array is to be considered in order to minimize the effects from the SPS engine during maneuvering or orbit changes.

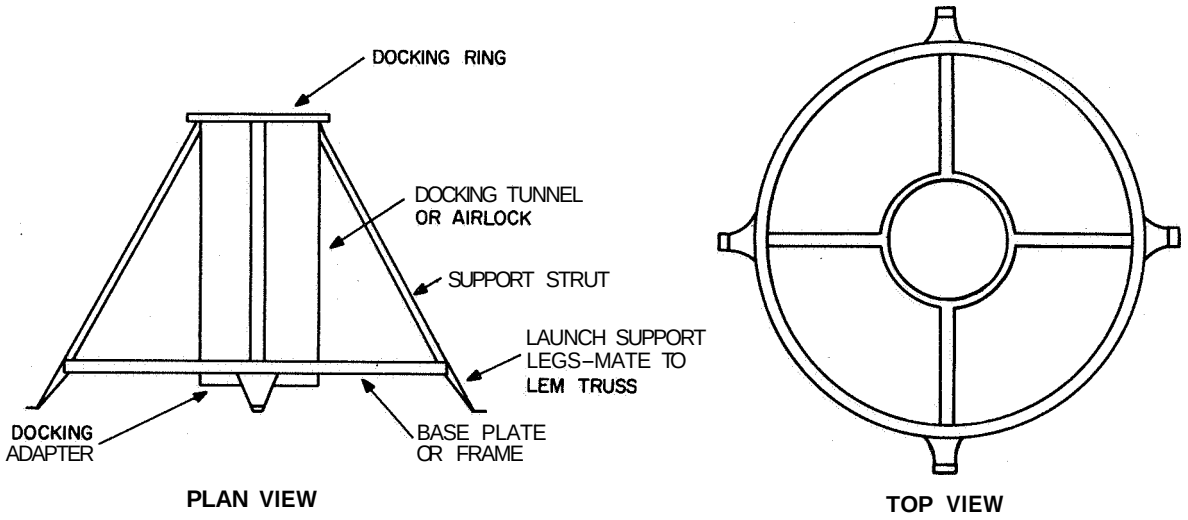


Figure I-5. PSM Base Structure

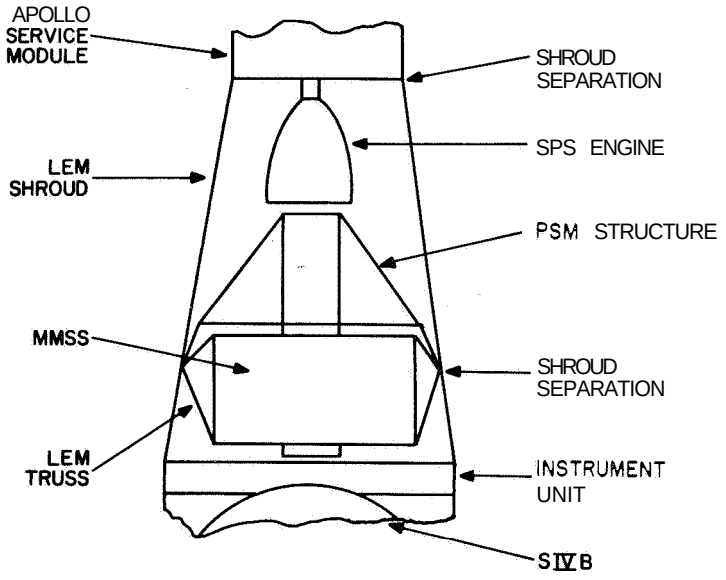


Figure I-6. PSM Launch Configuration

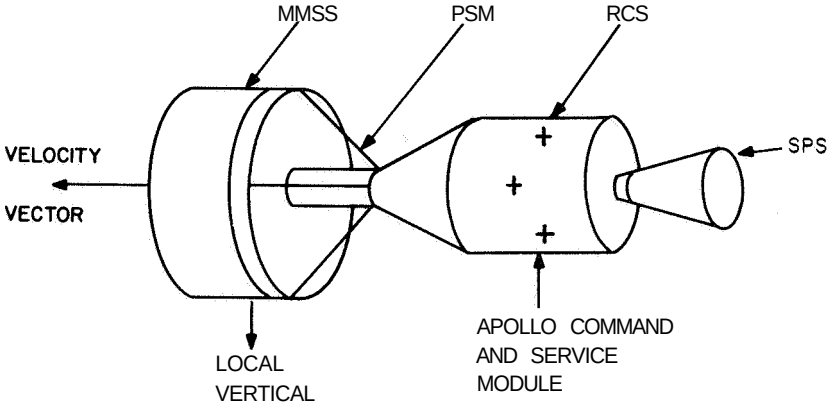


Figure I-7. PSM Orbital Configuration

LIST OF REFERENCES FOR SECTION I

I-1. Lockheed-California Company, Modular Multipurpose Space Station Study, Report LR18906, July 30, 1965.

SECTION II

SUMMARY OF RESULTS

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A. GENERAL POWER SYSTEM ANALYSIS

An evaluation of the approach to a power-system analysis has resulted in a definition of the primary tasks to be performed and their interrelationships, definitions of an Integrated Power System **(IPS)** and a Power System Module (PSM), and the establishment of a modular approach.

The technical tasks to be performed during a power-system conceptual study may be separated into three interrelated sections, as shown in Figure **11-l:** Electrical System Analysis, Orientation System Analysis, and Mechanical System Analysis. The ground rules are utilized along with technical component, materials and methods data to perform detailed iterated analyses.

An Integrated Power System (IPS) is designed as an integral part of the vehicle that it services. The batteries and electronics units are housed within (and affect the design of) the vehicle. Temperature control for these units is provided by the vehicle environmental control system liquid loop. During launch, the solar array is stowed in the void between the vehicle and the shroud envelope; multiple-point attachment of the array to the vehicle is provided for withstanding the launch environment. In orbit, the array, attached to the vehicle by a boom, is automatically deployed and sun-oriented. As a result, there are eighteen interfaces between the orbiting vehicle and the power system (See Figures **11-2** and **11-3).**

A Power System Module **(PSM),** on the other hand, is a separate module which houses the batteries and electronics in a component housing assembly; this assembly provides structural support as well as the thermal control system. During launch, the solar array is stowed adjacent to the component housing assembly.

In orbit, the array, attached to the PSM structure by a boom, is automatically deployed and sun-oriented. Only nine vehicle-to-power system interfaces exist, thus permitting vehicle design to be relatively independent of the power-system design. (See Figure **11-4** and II-5).

Although the PSM has fewer interfaces, the penalties involved (in terms of power-system complexity and overall mission cost) have not been evaluated.

A "functional-element paralleling'' modular concept, as shown in Figure II-6, has been selected for both the IPS and PSM because it provides the flexibility required and minimizes the weight of the fundamental elements.

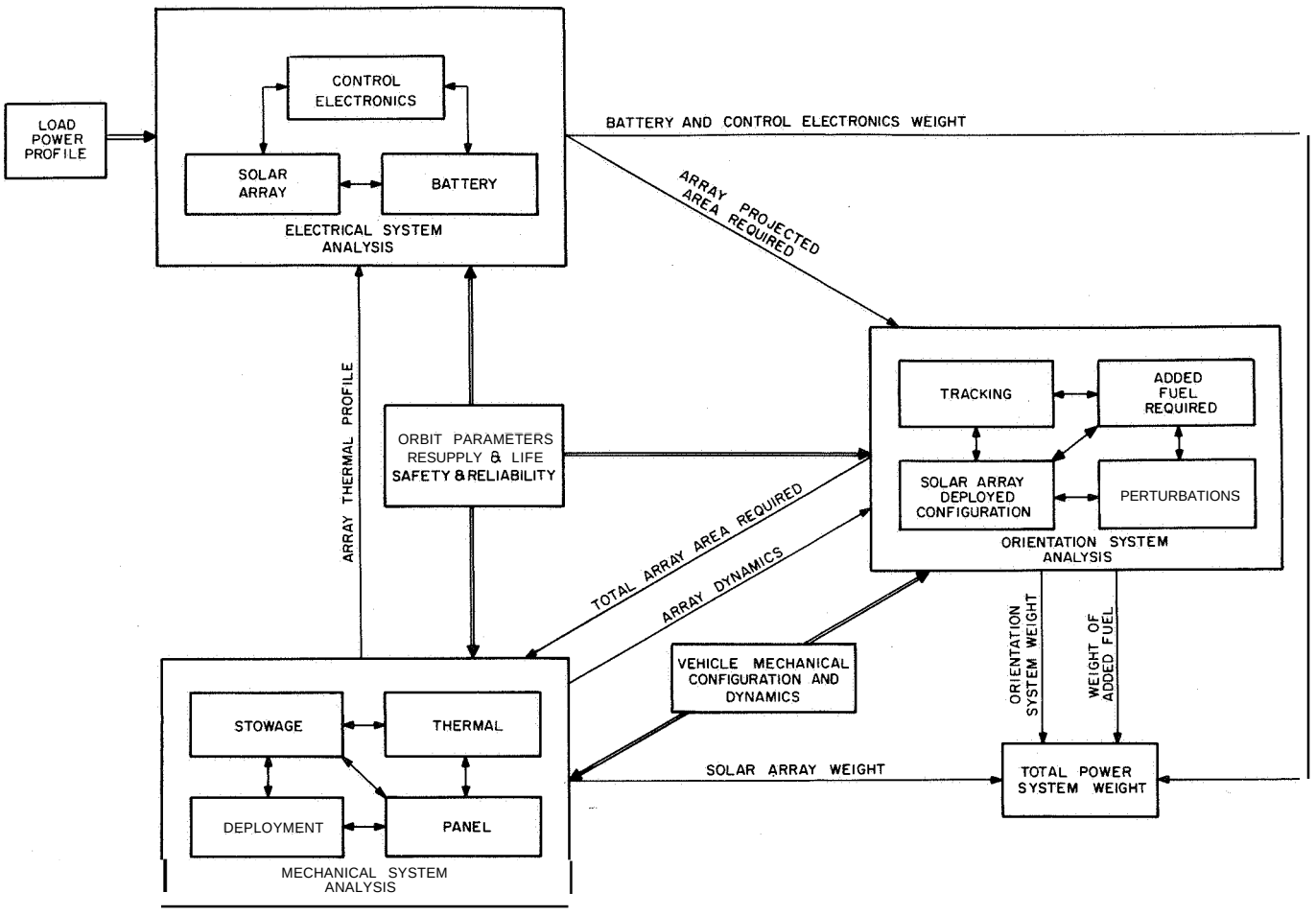


Figure 11-1. Power System Technical Study Tasks

B. IPS ANALYSIS

1. General

The Integrated Power System requires 4900 square feet of solar array in a "Z" configuration (Figure 11-2) to support the largest electrical load requirement (5 to 15 kW). The electrical-system configuration is a parallel-tracker (as shown in Figure 11-7) consisting of a solar array, a maximum power tracker/battery charger, storage batteries, PWM* type DC regulator, and a 3-phase, 400-cycle inverter. During the sunlight portion of the orbit, the load power is supplied by the solar array. The array power output in excess of the load demand is used to charge the batteries. When the batteries are fully charged, the charge electronics reduces the charge rate to a trickle charge and the maximum power tracker changes the solar-array operating point reducing the power drawn from the array to a level required for satisfying the load demand and battery trickle charging. During this sunlight period, the solar array is aligned normal to the sun by a two-axis orientation system. One array axis is driven at one revolution per orbit, while the other operates at a rate of up to 8 degrees per day. A sun-sensor servo system controls the orientation drive.

*Pulse-width modulator

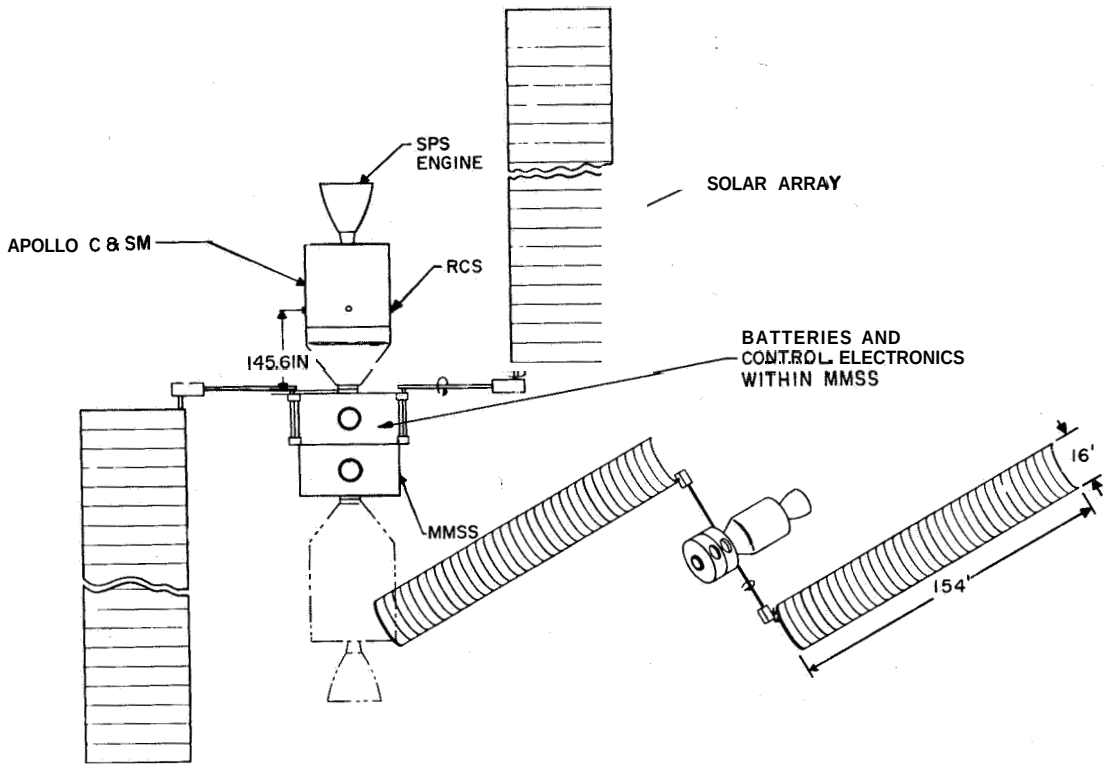


Figure 11-2. IPS Deployed Configuration

INTEGRATED POWER SYSTEM — A POWER SYSTEM WHOSE COMPONENTS ARE AN INTEGRAL PART OF THE VEHICLE

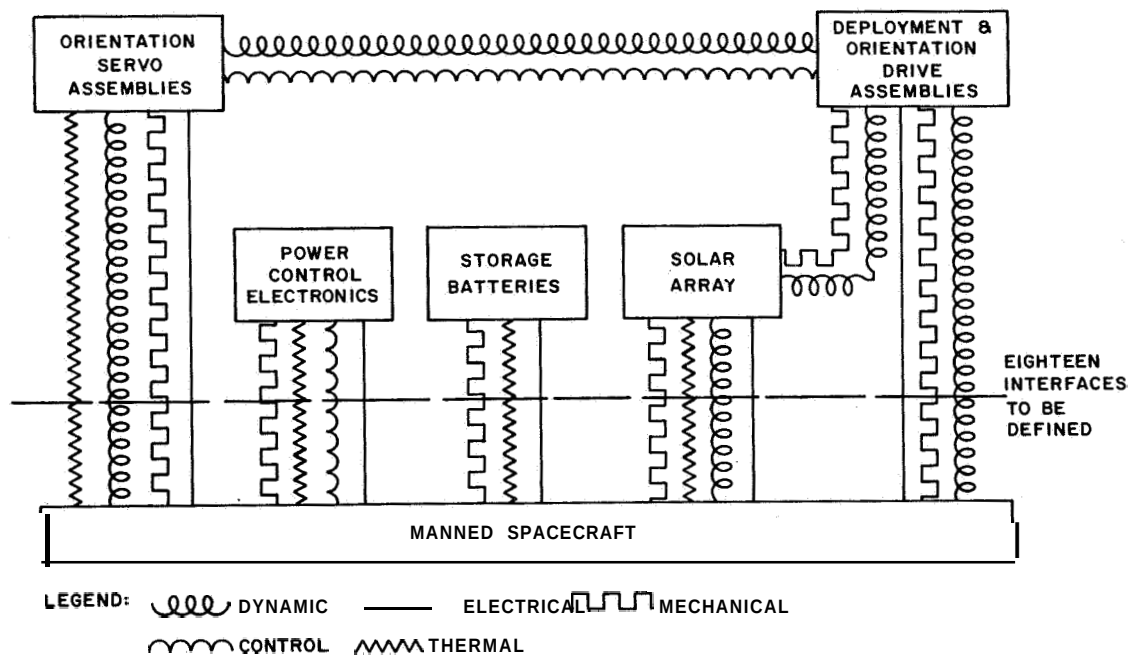


Figure 11-3. IPS Interfaces

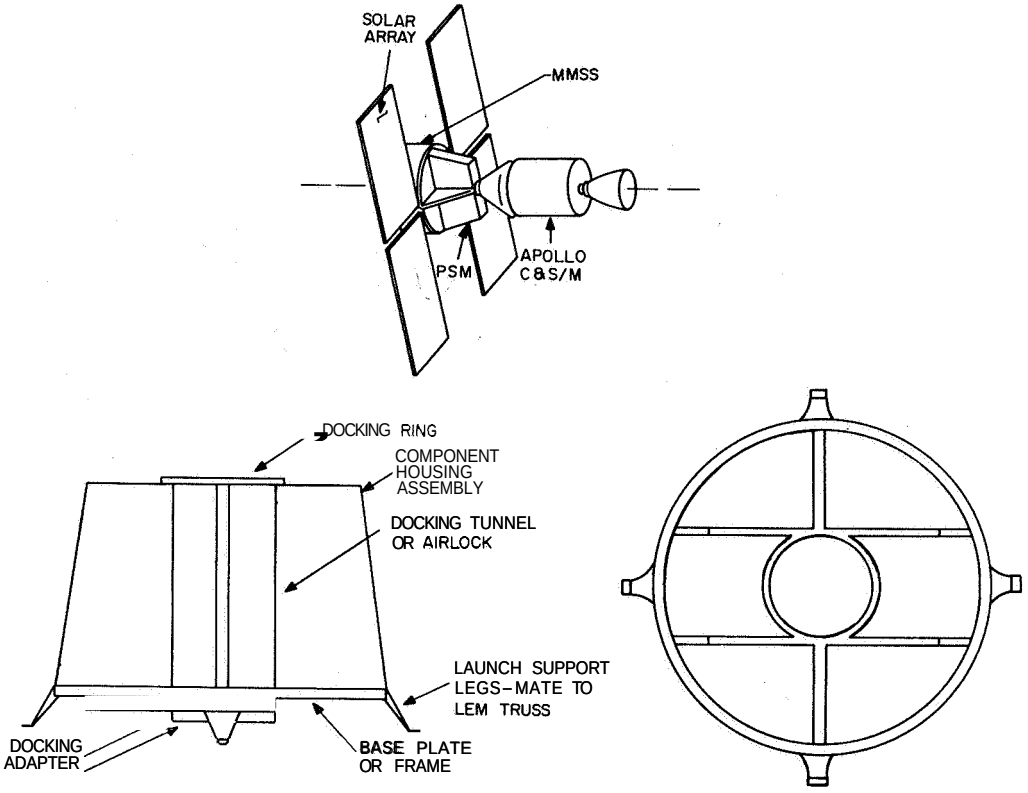


Figure II-4. Power System Module (PSM) Configuration

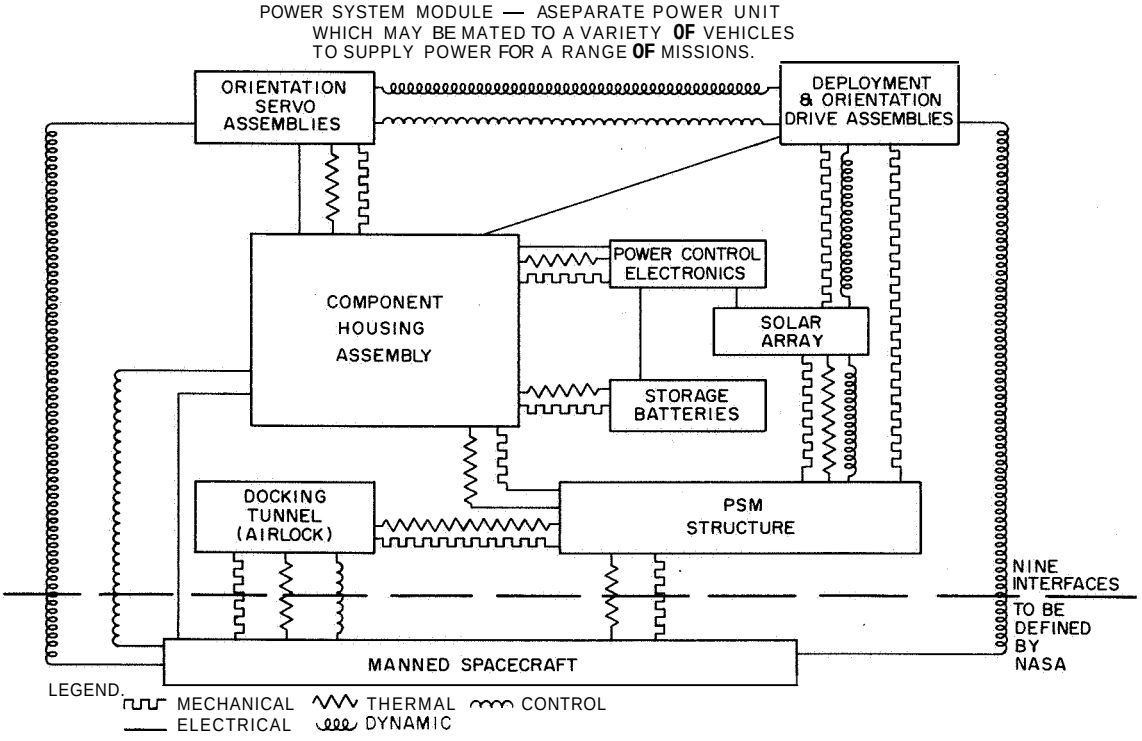


Figure II-5. PSM Interfaces

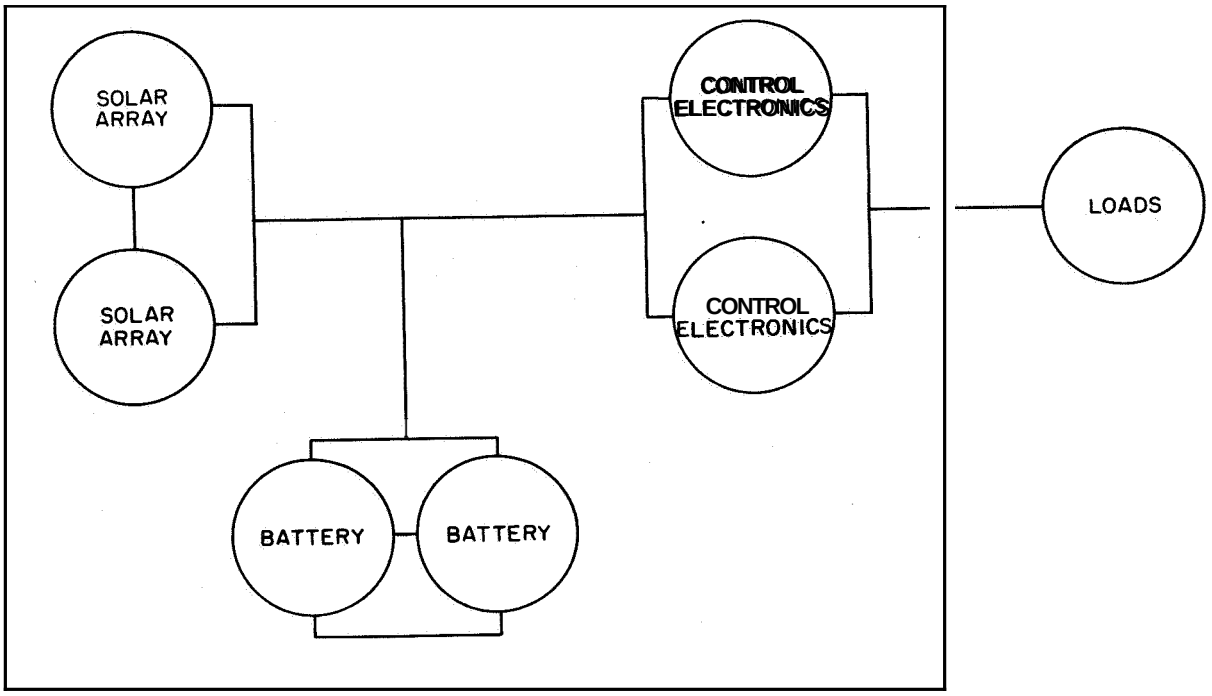


Figure II-6. Functional Element Paralleling

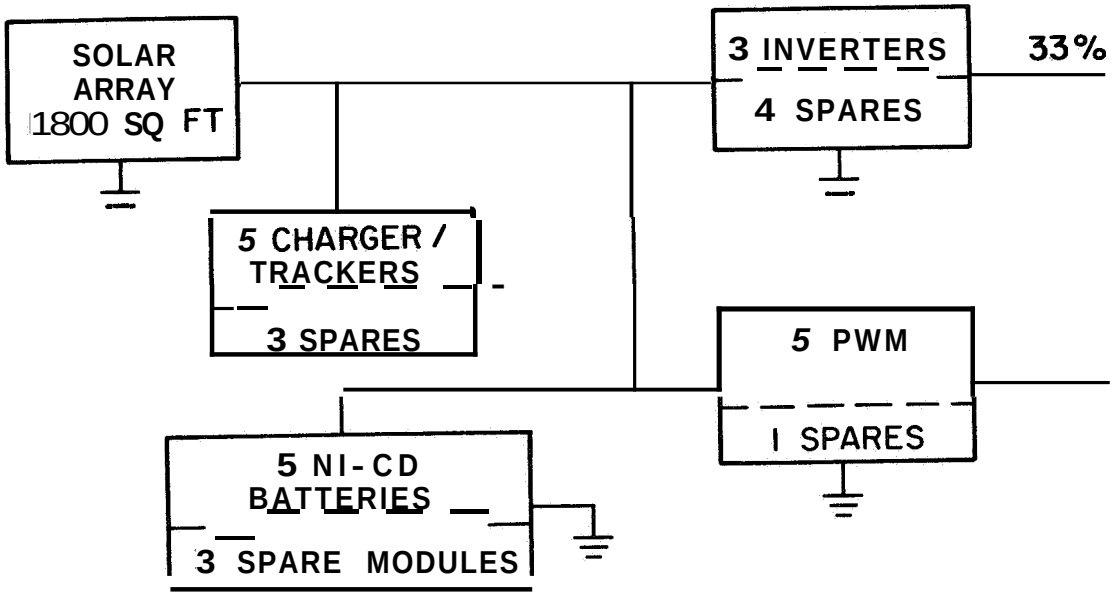


Figure II-7. IPS Electrical Configuration

During occultation, when the array is not illuminated, load power is supplied by the batteries and the solar array is driven at a constant rate so that at the transition from occultation to sunlight, there will be a minimum error in array normality to the sun.

For launch, the solar array is stowed in accordion fashion around the periphery of the MMSS compartments. Upon injection into orbit, array restraining cables are separated by pyrotechnic devices, and the solar array is deployed. In the deployed condition, the array panels (aluminum honeycomb with an aluminum frame) assume a curved shape, as shown in Figure 11-2.

Docking of the Apollo Command and Service Module with the MMSS can be accommodated with the array oriented in the weak-axis bending mode as shown in Figure II-8. SPS* engine firing requires that the array be placed in the tensile mode but plume effects may be critical.

The array "Z" configuration requires significant additional RCS and SPS fuel to overcome the effects of perturbations on the 4900 square feet of area.

2. Electrical System Studies

From a comparative analysis of four basic electrical systems, a parallel-tracker configuration has been selected on the basis of the criteria presented in Table II-1.

* Service Propulsion System

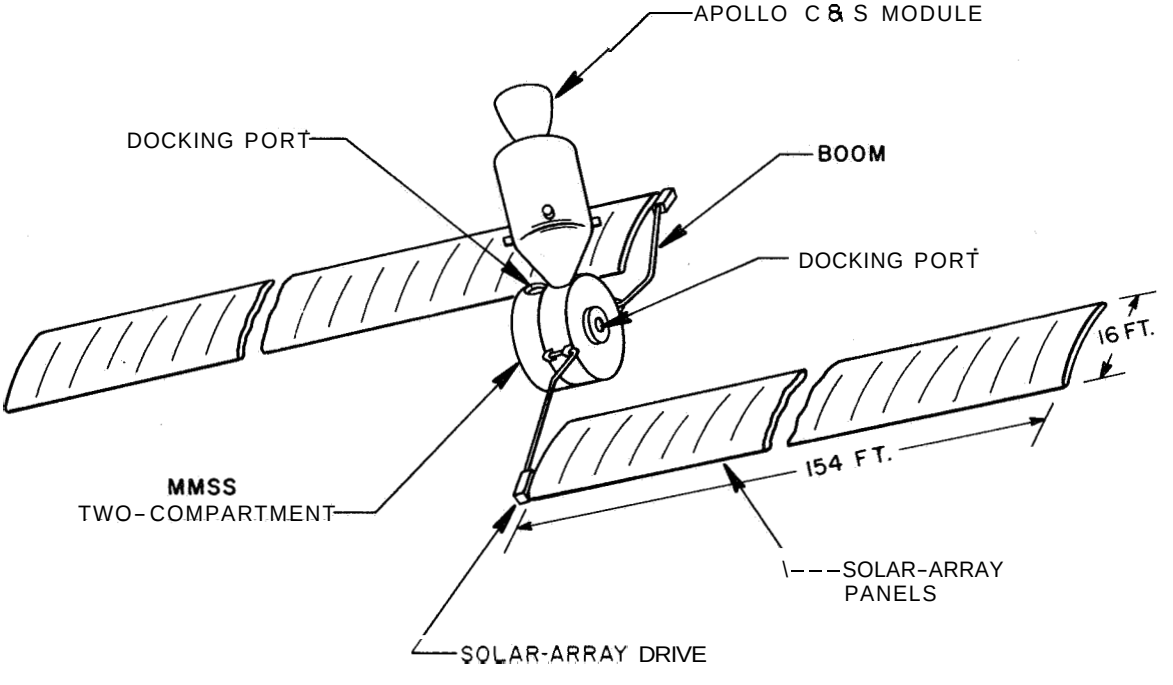


Figure II-8. Preferred Docking Configuration

TABLE 11-1. SYSTEM CONFIGURATION COMPARISON

System	System Relative Efficiency		Baseline Reliability	Number of Power Modules for. 0.995 Reliability	Dissipation	Array Bus Voltage Excursions	Array Elec. Redesign for New Mission
	200 nmi	19380 nmi					
SNT	0.953	0.932	0.43	17	High	LOW	Yes
ST	0.941	0.851	0.53	16	LOW	High	No
PNT	0.986	1.000	0.38	46	Very High	Very Low	Yes
PT	1.000	0.932	0.42	17	LOW	High	No

SNT - Series non-tracker PNT - Parallel non-tracker

ST - Series tracker PT - Parallel tracker

The solar array will consist of a series-parallel arrangement (145 cells per series string) of silicon N-on-P solar cells with a 1 ohm-cm base resistivity and an A M P (bare) efficiency of 10.5 percent at 27°C. The cells will employ conventional contacts and be 2 by 3 CM and 10 mils thick, with a cover glass of 6-mil fused silica. A preliminary layout of the array has been prepared, and is shown in Figure II-9. Array power generated will be transmitted to the spacecraft by means of tapered aluminum bus bars.

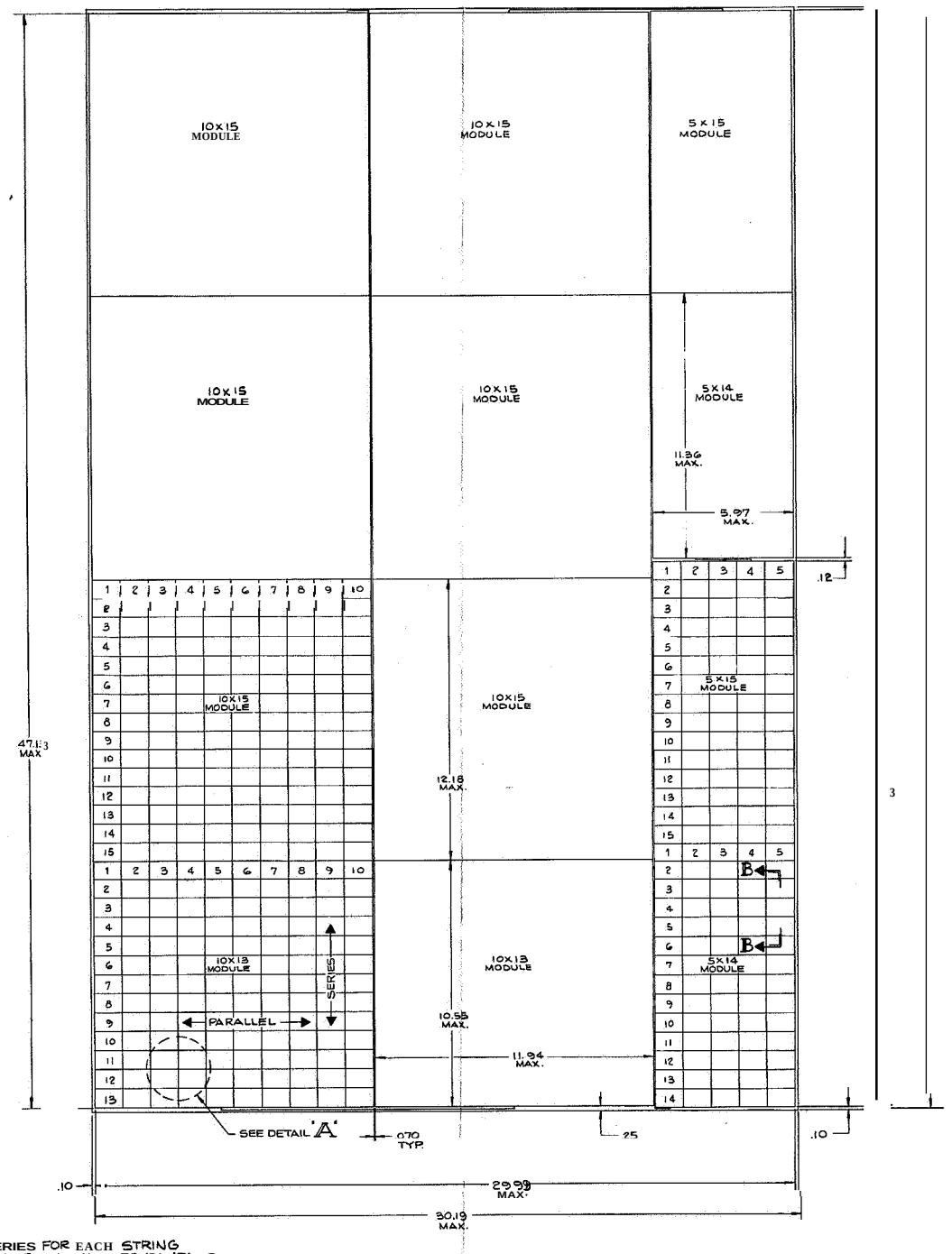
Solar-cell minimum temperature, array backside emissivity and array mechanical design are all factors affecting the watts-per-pound ratio of the solar array. Based on a trade-off analysis of these parameters, an array minimum temperature of -150°C has been chosen. Although a solar array has not been qualified to this temperature, a reasonable projection from the Lunar-Orbiter solar array, which was qualified to -120°C, can be made.

The selection of solar-cell type was based on radiation resistance, temperature characteristics, packing factor, cell cost, assembly cost, and weight.

The storage batteries will use either silver-cadmium or nickel-cadmium sealed cells packaged in a series-parallel arrangement as shown in Figures 11-10 and 11-11. General characteristics will be as shown in Table 11-2. A weight trade-off analysis has shown that different battery types may be applicable to a specific range of altitudes and mission durations (see Figure 11-12).

Battery-charge control will be accomplished by an ampere-hour counter, a multi-level, constant-current controller, and a voltage limiter. The system will contain provisions for

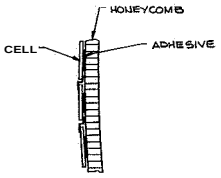
* Air-Mass-Zero



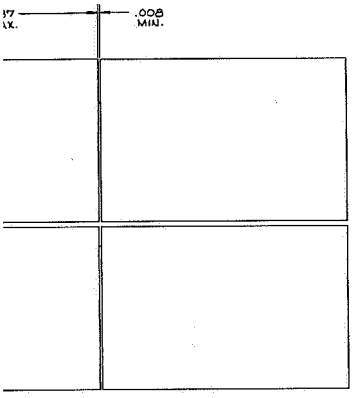
NOTES'

1 145 CELLS IN SERIES FOR EACH STRING
2 TOTAL NO. OF 10X15 CELL MODULES/PANEL 6
TOTAL NO OF 10X13 CELL MODULES/PANEL 2
TOTAL NO OF 5X15 CELL MODULES/PANEL 2
TOTAL NO OF 5X14 CELL MODULES/PANEL 2
TOTAL NO OF CELLS 1450
3 PACKING FACTOR-ACTIVE SILICON AREA/TOTAL PANEL AREA = 895

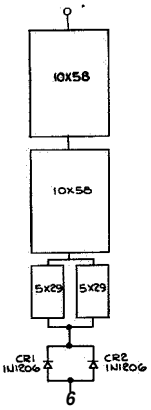
Figure 11-9. Preliminary Solar Array Section Layout



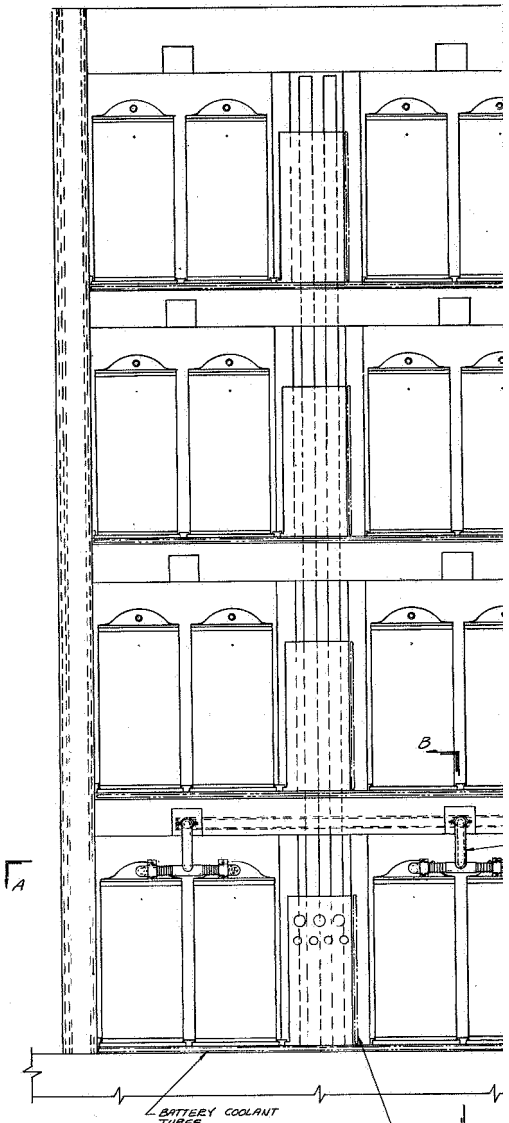
SECTION B-B
(TYP. CELL INSTALLATION)
SCALE: NONE



DETAIL A
(TYP-ALL CELLS)
SCALE: 4/1



TYP PANEL SCHEMATIC



FRONT VIEW

INSTRUMENTATION DOORS REMOVED

NOTES:
1. REFERENCE DRAWINGS:
BATTERY MODULE SK.1769403
BATTERY MODULE WIRING SK1365275
BATTERY STACK & CONTROL UNIT WIRING SK 1965276
BATTEEY STACK & CONTROL MODULE WIRING SK 1769402
BATTERY CHARGE MODULE SK 1966551

TI-9-1

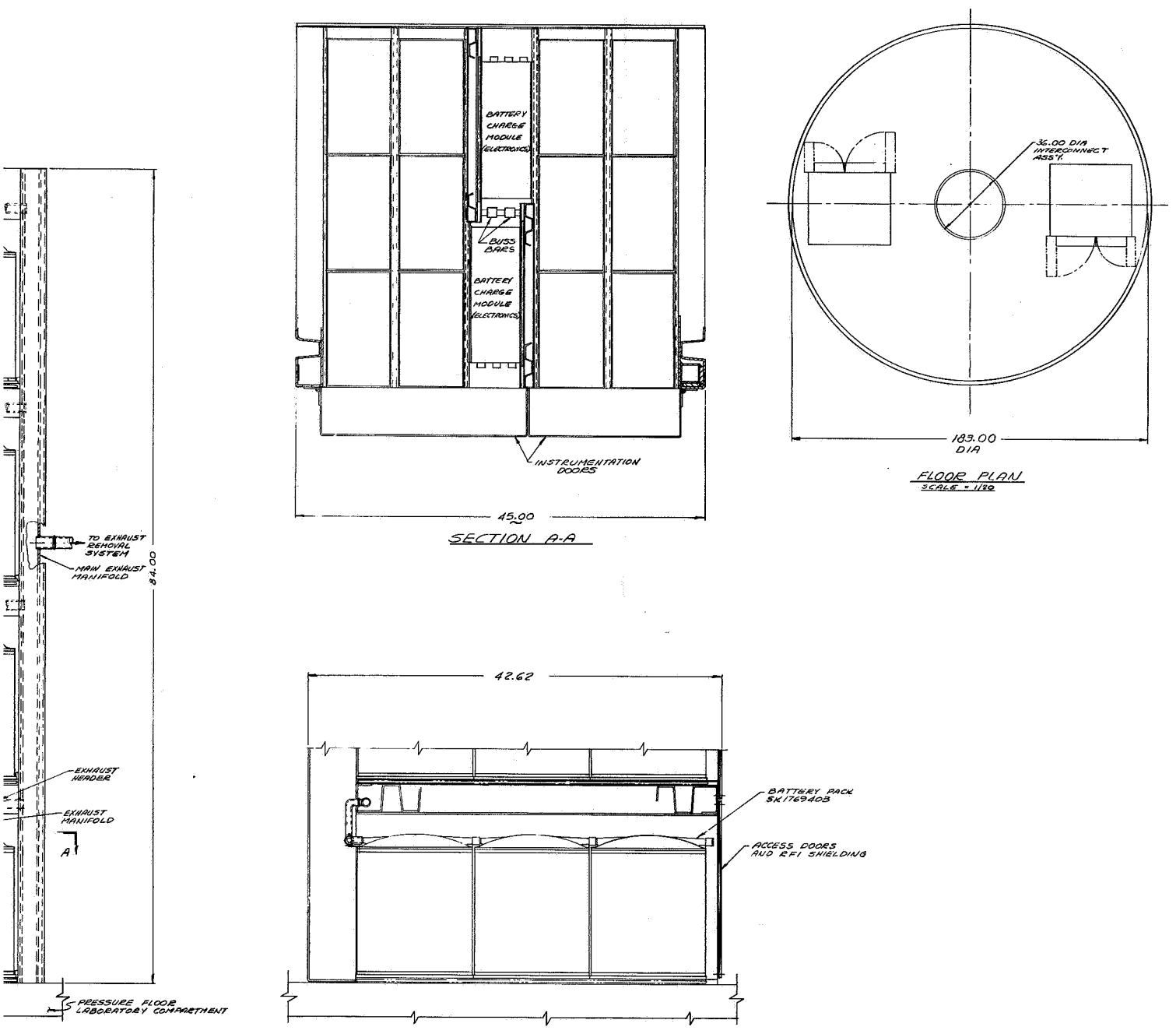
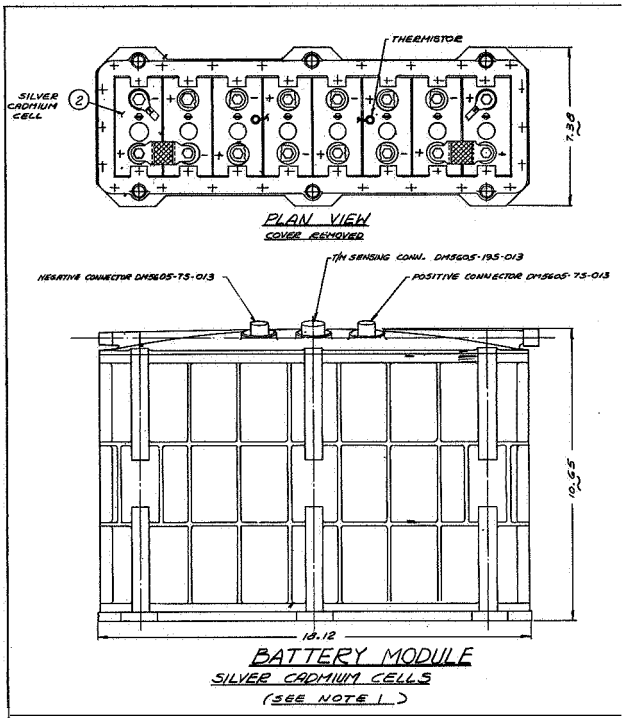
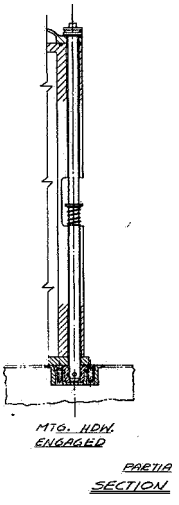


Figure II-10. Battery Arrangement, Ni-Cd





NOTES:

1. ALL STRUCTURAL DETAILS FOR SILVER CADMIUM CELL BATTERY MODULE ARE SIMILAR TO THOSE SHOWN FOR NICKEL CADMIUM CELL BATTERY MODULE.
2. REFERENCE DRAWINGS:
   BATTERY MODULE WIRING - SK 1965275
   BATTERY STACK & CONTROL UNIT WIRING - SK 1965276
   BATTERY STACK & CONTROL MODULE WIRING - SK 1769402
   BATTERY ARRANGEMENT - SK 1769799
3. NICKEL CADMIUM BATTERY MODULE CAN BE CONNECTED TO USE FOUR CELLS OR FIVE CELLS DEPENDING UPON WIRING OF MATING NEGATIVE PLUG.

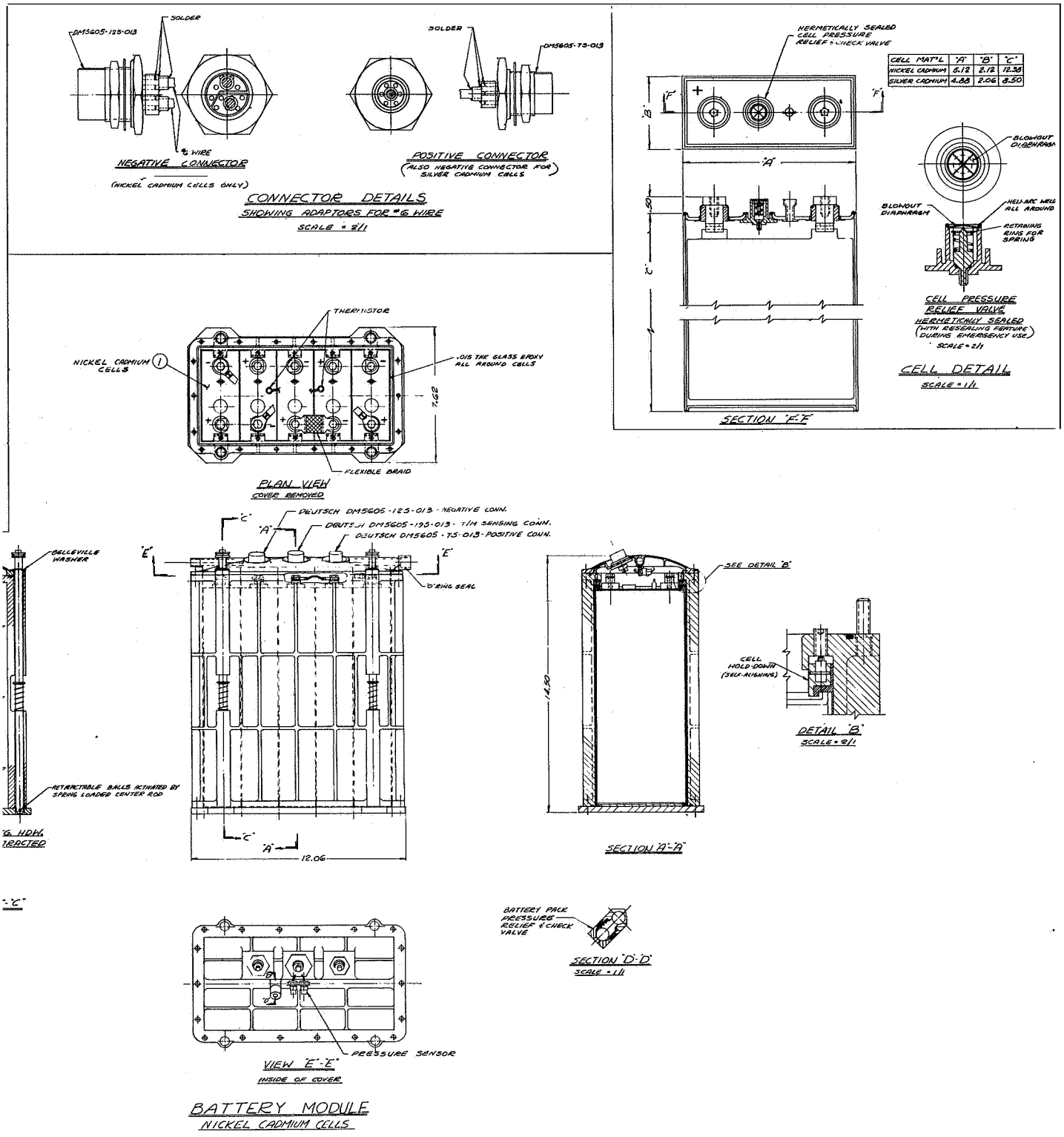


Figure II-11. Battery Module

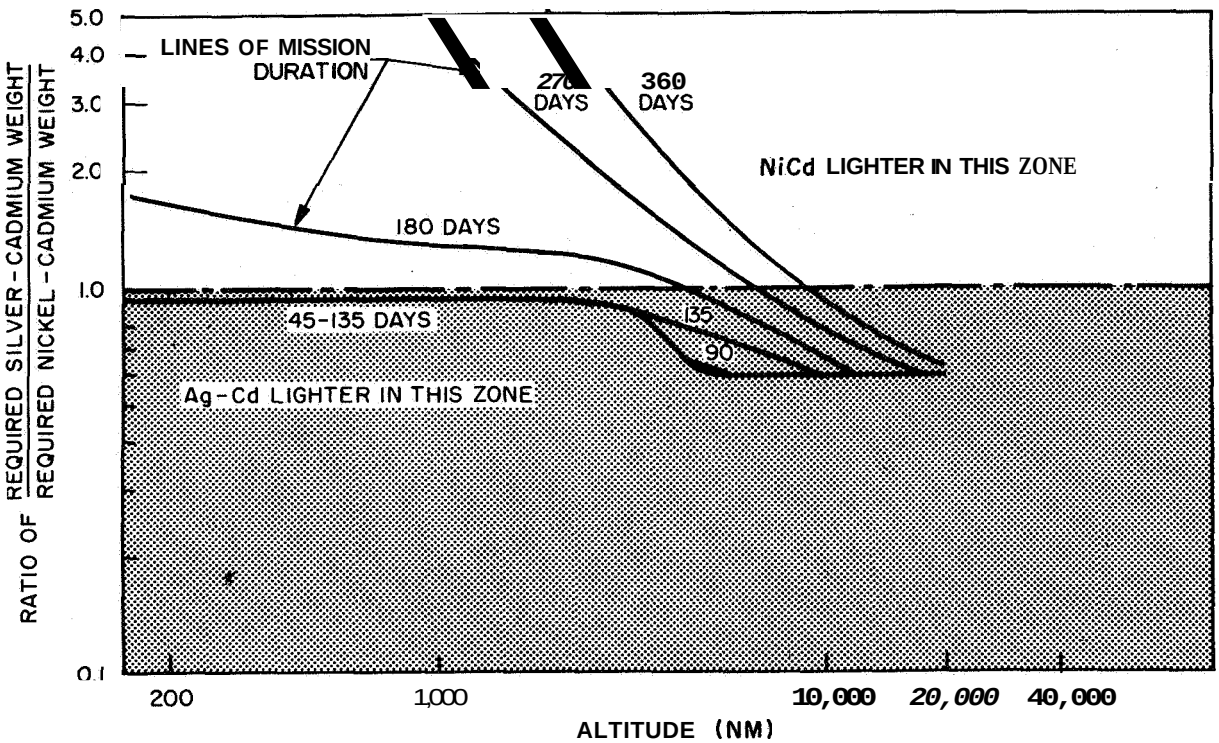


Figure 11-12. Battery Weight Ratio versus Altitude

high-temperature protection as well as for manual adjustment of control criteria. The charging cycle will be performed as follows:

(1) Initially charge at the highest acceptable charge rate until 50 percent of the capacity (ampere-hours) removed during the previous discharge has been replaced.

(2) Reduce the charge current to a lower rate at which most of the charging will take place.

(3) When all capacity (ampere-hours) removed during the previous discharge cycle has been restored (including additional capacity to compensate for battery inefficiency) reduce the current to a trickle rate for the balance of the charge period.

(4) During each of the three steps above, the current may also be limited by the presence of a charge voltage limit which is temperature controlled.

(5) Reduce the voltage limit at the time of final current reduction at 100 percent state-of-charge (for silver-cadmium only).

TABLE II-2. BATTERY DESIGN FACTORS

Item	Nickel-Cadmium	Silver-Cadmium
Maximum depth of discharge	0.50	0.50
H average (maximum)	5	8
H average (minimum)	20	30
H peak	3	5
No. of series cells	29	32
Watt-hrs per pound*	9.2	16.0
Watt-hrs per ft^3 *	950	1,670
Watt-hrs per pound**	7.7	13.2
Watt-hrs per ft^3 †	353	620
Average cell voltage ††	1.20	1.10

*Includes battery modules only. Numbers furnished for information purposes only.

**Includes weight of structural shelf (with cooling coils and busbars) upon which batteries and electronics will be mounted.

†† Based upon external volume of battery and electronics support structure.

†† To full dishcarge; to be used in converting ampere-hours to watt-hours in the computer program.

There are three major electronic functional assemblies: charger/tracker, DC regulator, and AC inverter. The charger/tracker (C/T) assembly consists of a number of power modules equal to the number of batteries. In addition, the assembly includes two redundant control modules packaged together with, but replaceable independently from, a failure-detection and switchover module. The function of the C/T assembly is to provide for the maximum safe transfer of energy from the solar array to the batteries. This is accomplished taking into account the load requirements (which are satisfied first), the requirements of the batteries, the inherent limitations of the batteries and the instantaneous capability of the source. A functional diagram of the C/T assembly is shown in Figure 11-13.

The DC regulator assembly consists of power modules (whose number is selected on the basis of the DC load and reliability requirements), two redundant control modules and a failure-detection and switchover module. Efficient DC voltage regulation is obtained through pulse-width modulation (PWM) techniques. Current limiting and over-voltage protection are provided. As in the case of the C/T assembly, all circuits are packaged within three types of replaceable modules. A functional diagram of the DC regulator assembly is shown in Figure 11-14.

The AC inverter assembly is assumed to consist of two or more active, self-sufficient power modules placed in parallel. Static-inversion techniques are utilized. (A 2.5-KVA, 3-phase, 400-Hz inverter development is being performed by Westinghouse Corporation for NASA Lewis Research Center. Therefore, no in-depth analysis of this assembly has been performed by RCA.)

A power distribution unit (PDU) is also incorporated to provide interconnection of the functional elements of the power system.

Mechanical packaging for the PWM regulator, charger/tracker and power distribution unit are shown in Figures 11-15 through 11-18. A proposed internal layout of the power system batteries and electronics in the MMSS compartment is shown in Figure 11-19.

The operational philosophy of a power system in a manned-orbiting-vehicle application is intrinsically related to safety and reliability. Conceptual studies have resulted in a preliminary establishment of those power-system functions to be monitored onboard and those to be telemetered to the ground station. These are indicated in Tables 11-3 and 11-4, respectively. Automatic-protection requirements have been established and crew participation in maintenance defined.

Redundancy requirements for the power-system modules (including batteries) have been defined for a power-system probability of mission success of 0.995 and are tabulated in Table 11-5. The probability of crew survival in terms of the availability of emergency power is 0.9999976 (based on one battery remaining active for emergency power).

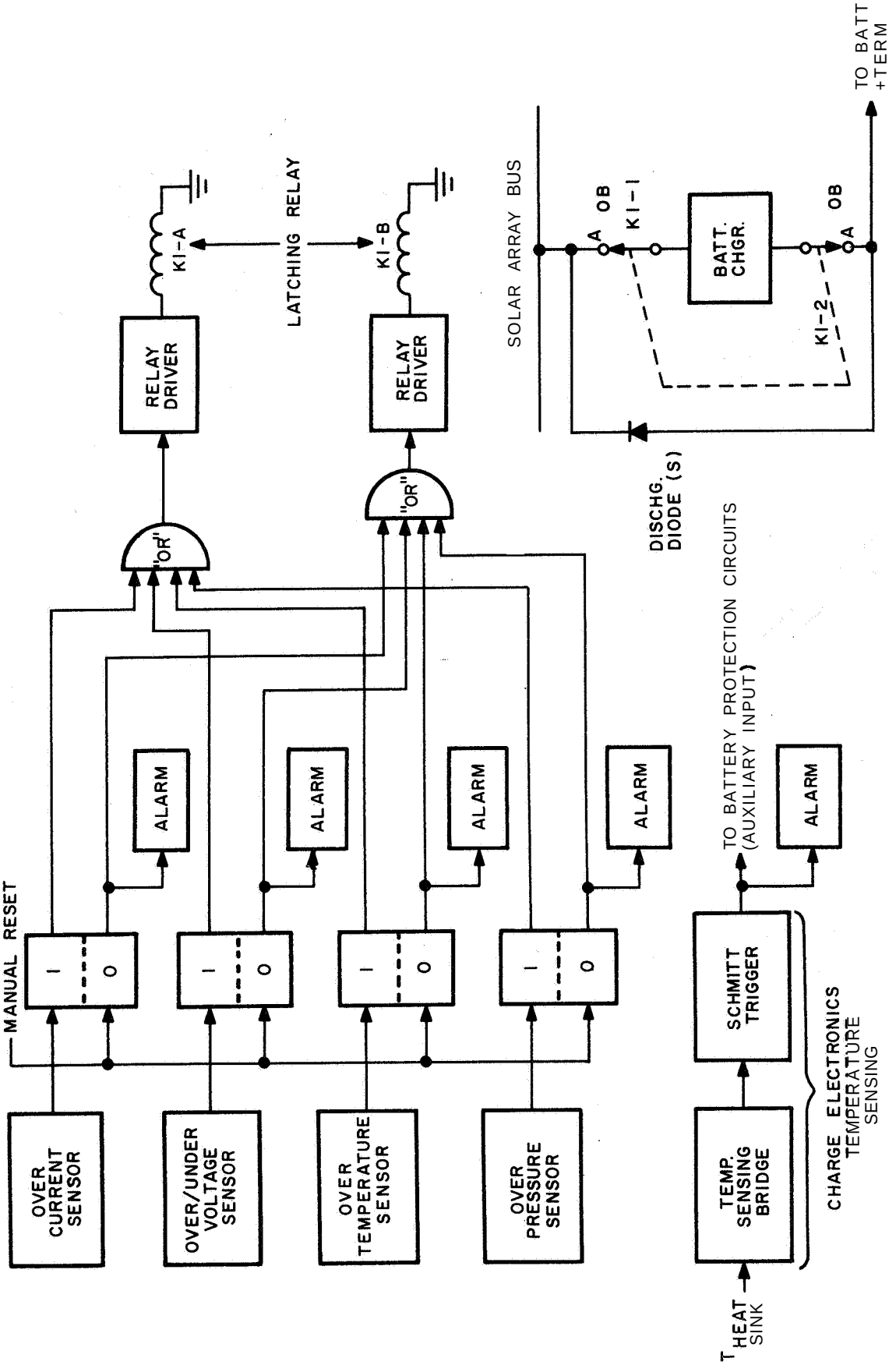


Figure II-13. Functional Diagram of the Charger/Tracker Assembly

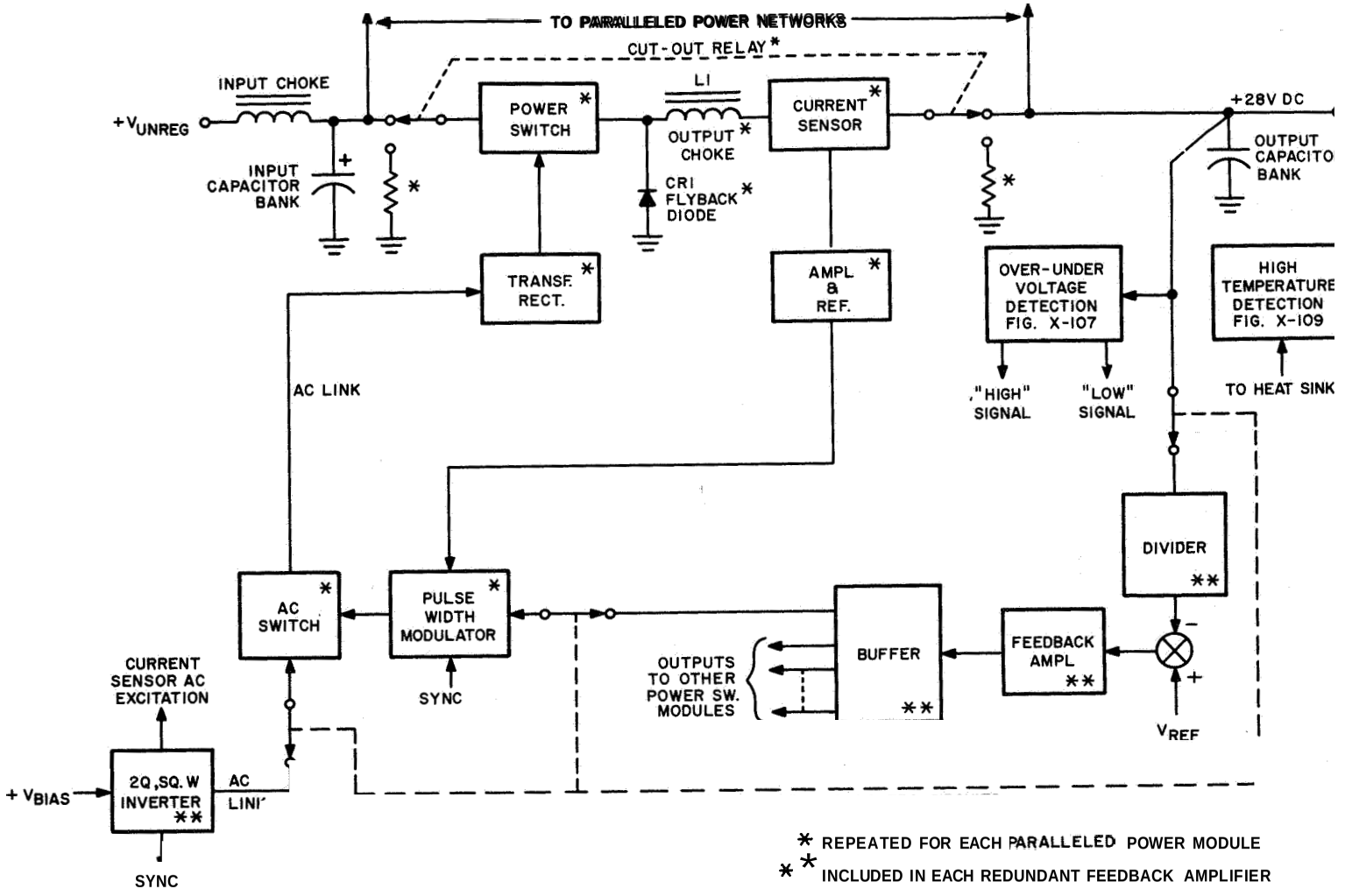


Figure 11-14. Functional Diagram of the DC Regulator Assembly

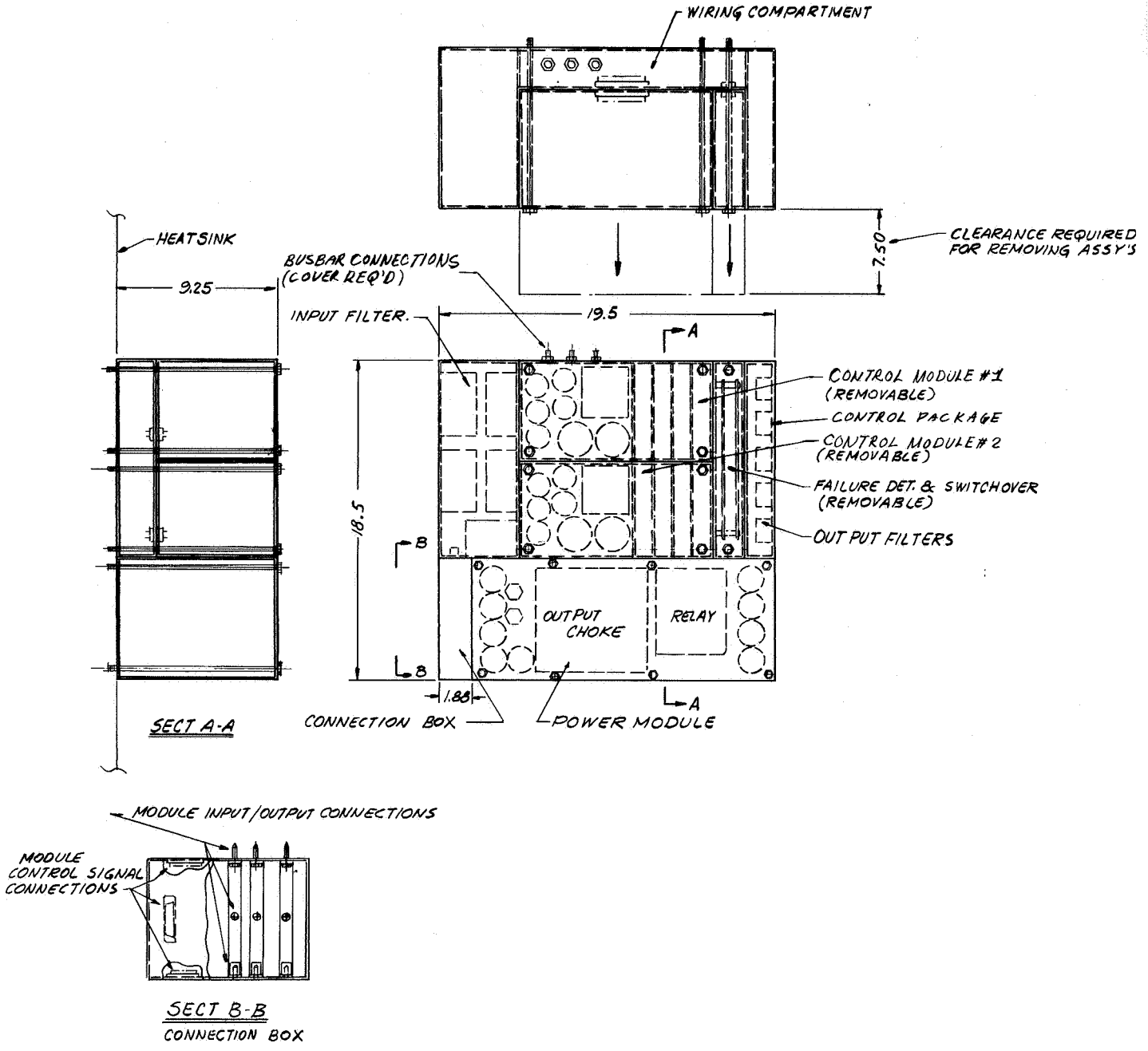


Figure II-15. PWM Regulator Packaging

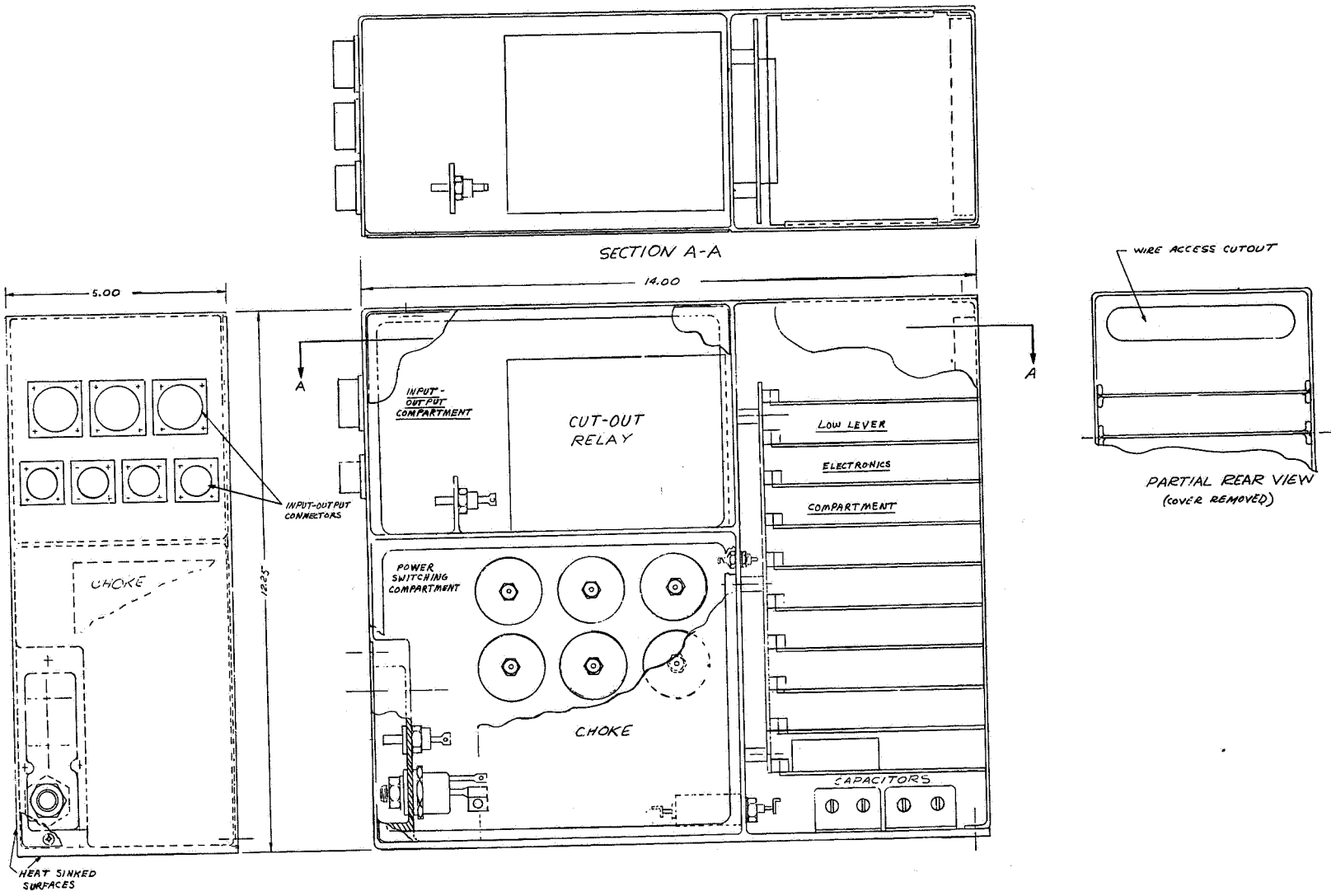


Figure II-16. Charg-A-Tocker Power Module Packaging

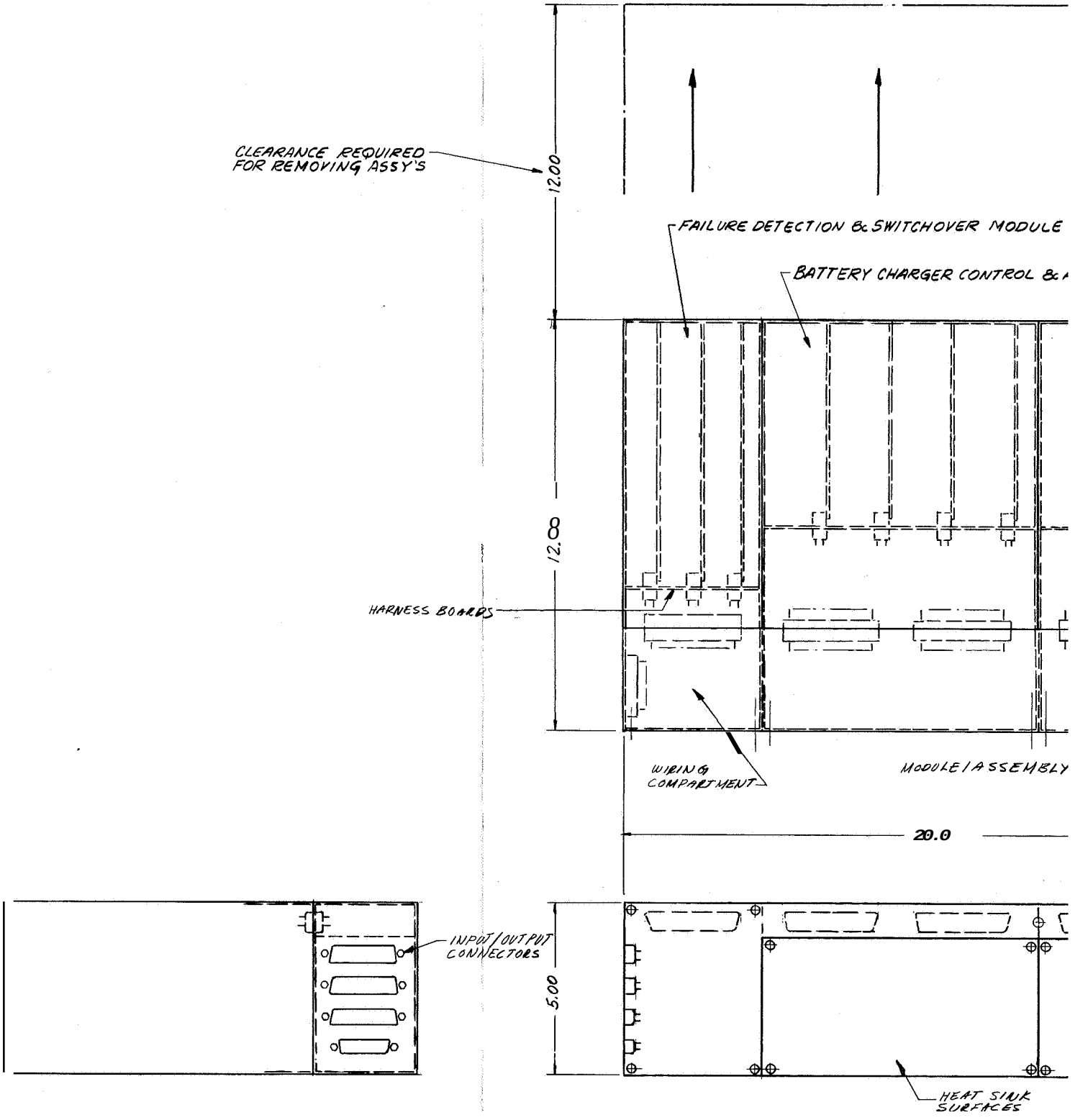
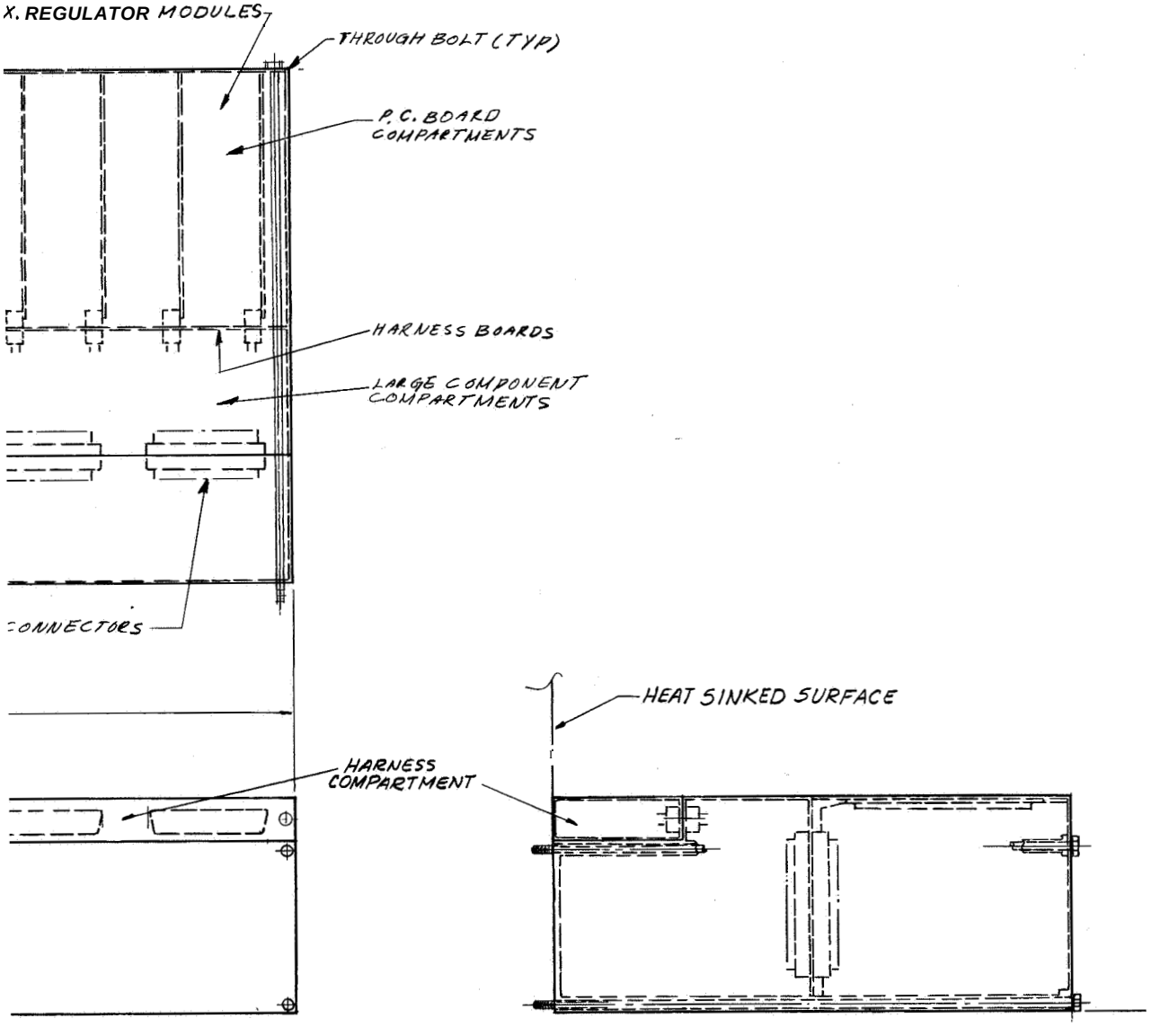


Figure 11-17. Charger/Tracker Control Assembly Packaging

X. REGULATOR MODULES



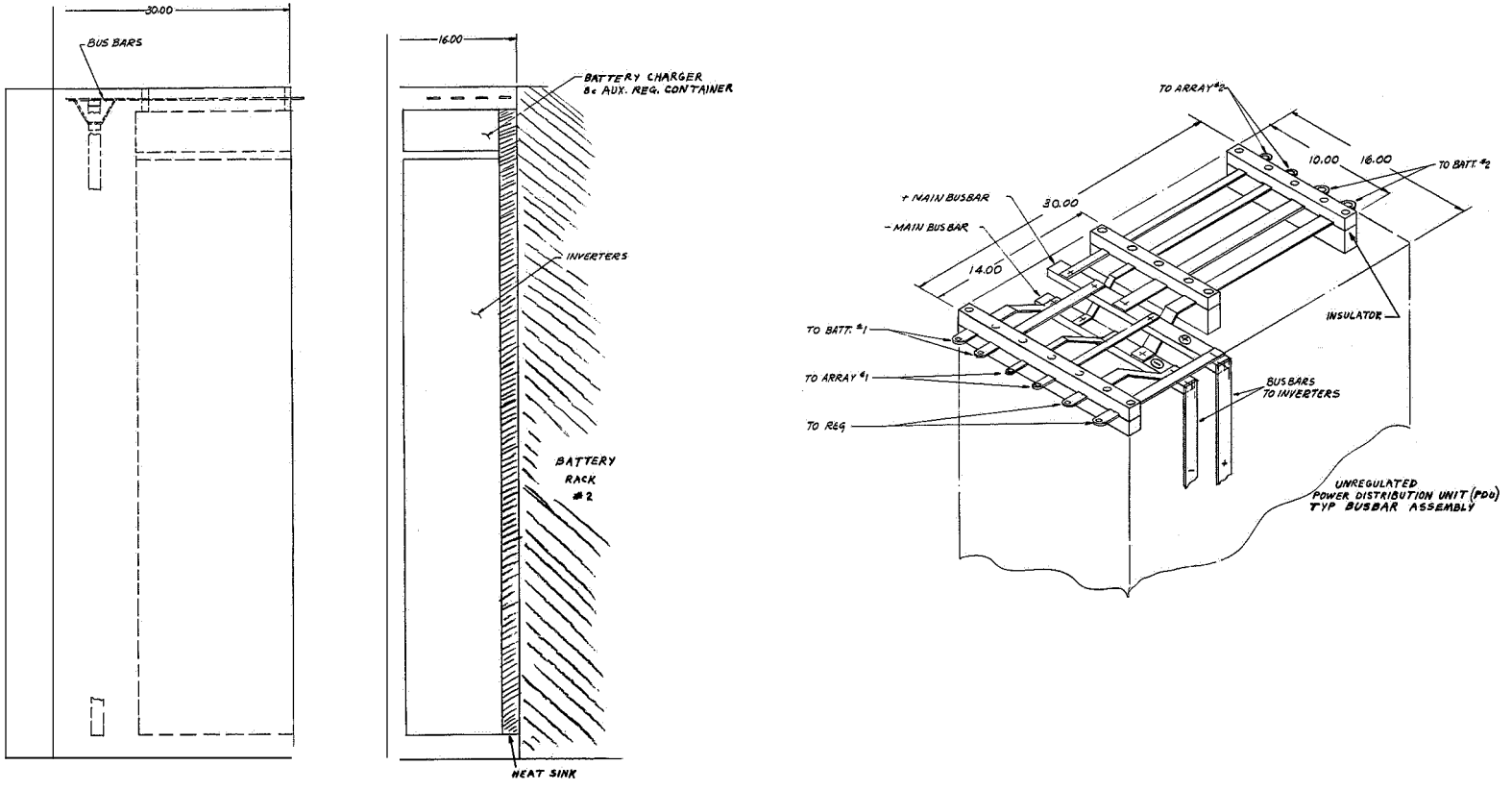


Figure II-18. Power Distribution Unit Packaging

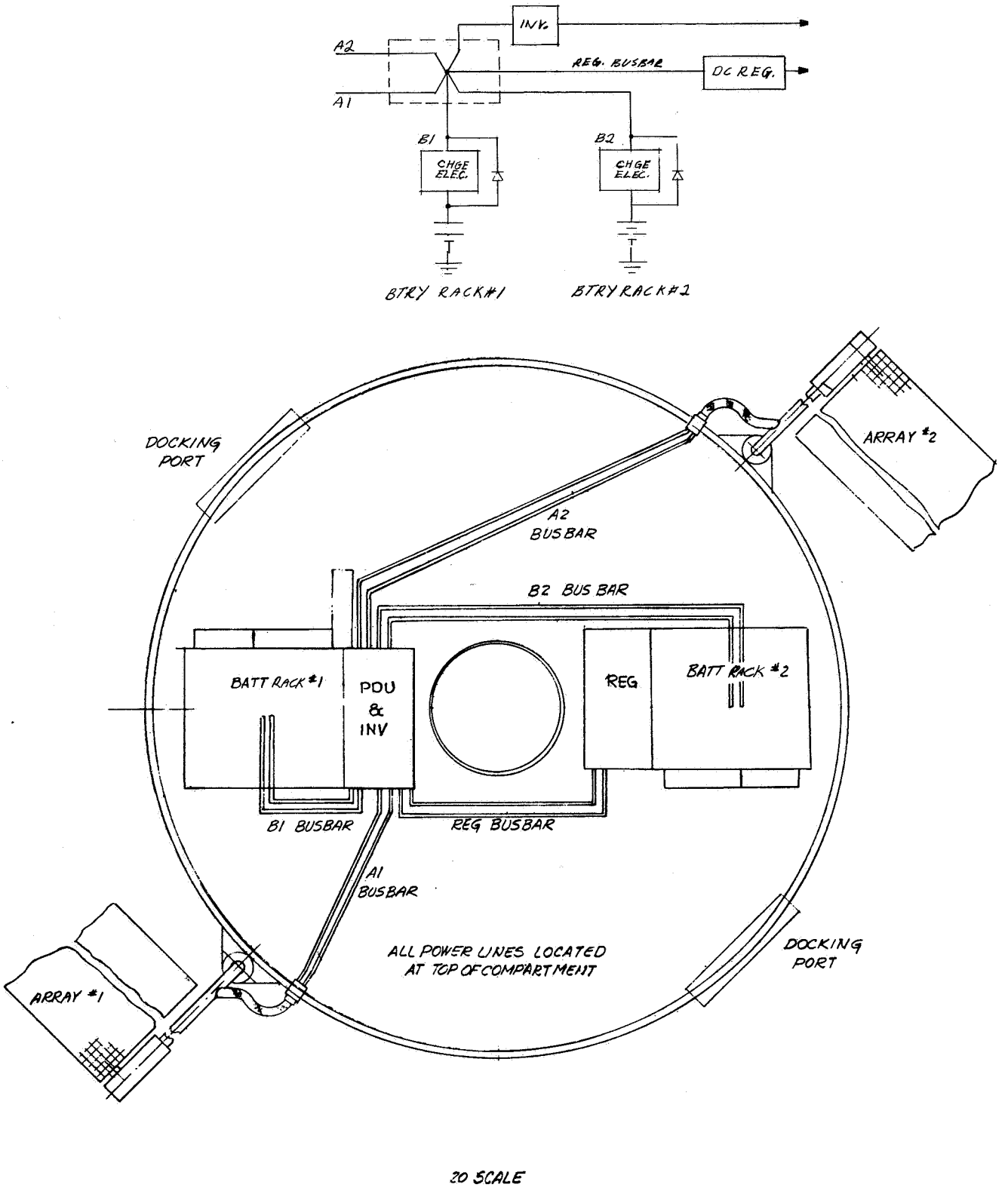


Figure 11-19. Battery and Battery Electronics Layout

TABLE II-3. POWER SYSTEM PARAMETERS TO BE MONITORED ON-BOARD

Equipment Group	Metered Performance Parameters
1. D-C Regulators	Load voltage Load current Operational status of power modules and FBA (i.e., on or off) Temperature
2. Inverters	Load voltage Load current Frequency Load power factor Module status (on or off) Temperature
3. Batteries	Terminal voltage Current Temperature State-of-charge
4. Battery Chargers	Temperature Operational status of redundant elements (i.e., on or off)
5. Solar Array	Voltage Current Temperature Sun sensors
6. Redundant Auxiliary Regulator	Voltage Current Operational status (on or off) Temperature
7. Ancillary Equipments includes: Telemetry and status monitoring Malfunction detection and protection Maintenance checking	None
P parameter allowing individual measurements as desired.	

TABLE 11-4. POWER SUBSYSTEM TELEMETRY POINTS

Equipment Group	Telemetry Points
DC Regulators	Regulated DC voltage Regulated DC load current (total) D-C Regulator Control Module status (on or off) D-C Regulator Power Module status (on or **off)** D-C Regulator heat sink temperature D-C Regulator over-under voltage indication D-C Regulator current limiting indications D-C Regulator input voltage (unregulated bus)
Batteries Batteries	Battery voltage (total) Battery charge current (per battery) Battery discharge current (per battery) Battery temperature (per battery) Battery charge/discharge acoumulations (per battery) Battery over-discharge indication (per battery) Battery insufficient charge indication (per battery) Battery charger over/under voltage indication (per battery) Battery over pressure indication (per battery) Battery over current indication (per battery) Battery over temperature indication (per battery)
Battery Charger	Charge Electronics Control Module status (on or **off)** Battery Charger Power Module status (on or **off)** Battery Charger electronics heat sink temperature
Inverters	Static Inverter AC output voltage (per phase) Static Inverter AC load current.(per phase) Static Inverter Power Module status (on or **off)** Static Inverter load power factor (per phase) Static Inverter frequency Static Inverter heat sink temperature Static Inverter over-under voltage indication Static Inverter current limiting indication Static Inverter high temperature indication Static Inverter output voltage (line to line)
Auxiliary Regulator Auxiliary Regulator	Auxiliary Regulator voltage (per regulator) Auxiliary Regulator current (per regulator) Auxiliary Regulator heat sink temperature (per regulator) Auxiliary Regulator over-under voltage indication (per regulator) Auxiliary Regulator current limiting indication (per regulator)

TABLE IC-5. REQUIRED SPARE MODULES

Baseline*	Resupply Interval (days)								Battery Modules	
	45			90			180		360				
	C	R	I	C	R	I	C	R	I	C	R	I	Ni-Cd	Ag-Cd
1	1	1	1	1	1	2	1	1	2	2	1	3	2	1
2	1	1	2	2	1	2	2	1	3	2	1	4	2	2
3	1	1	2	2	1	2	2	1	3	3	1	4	2	2
4	1	1	2	2	1	3	2	1	3	3	2	5	3	2
5	2	1	2	2	1	3	2	2	4	3	2	5	3	3
6	2	1	2	2	2	3	3	2	4	4	2	6	3	3
7	2	1	2	2	2	3	3	2	4	4	2		3	3
8	2	1	2	2	2	3	3	2	5	4	2		4	3
9	2	1	3	3	2	4	3	2	6	5	2		4	3
10	2	1	3	3	2	4	4	2	6	5	3		4	3

*Either electronics modules or batteries
C - Charger/Tracker Assembly
R - DC Regulator Assembly
I - AC Inverter Assembly

Note: Inverter spares based on 60,000 hours MTBF (See Section X for others)

3. Orientation System Studies

A "Z" solar-array configuration has been selected in order to meet the ground-rule requirements (See Section I). Gravity gradient, aerodynamics drag, magnetic dipole, and solar-pressure effects have been determined for the 4900-square-foot array area. The RCS* and SPS fuel requirements to overcome these effects were calculated and are presented in Table 11-6.

*Reaction Control System

TABLE II-6. REQUIREMENTS TO CANCEL PERTURBATIONS FOR A WORST-CASE 45-DAY PERIOD

PROPELLANT REQUIREMENTS	200 NMI, i = 28.5" SINGLE COMPARTMENT		200 NMI, i = 28.5" DOUBLE COMPARTMENT		200 NMI, i = 90° DOUBLE COMPARTMENT	
	WITH MSD	WITHOUT MSD	WITH MSD	WITHOUT MSD	WITH MSD	WITHOUT MSD
AERODYNAMIC						
*DRAG	510	510	510	510	510	510
LIFT	0	235	0	342	0	313
GRAVITY GRADIENT						
PITCH	0	3160	0	3160	0	3160
ROLL	2050	2050	2050	2050	2050	2050
SOLAR PRESSURE						
PITCH	4	4.5	5.8	6.5	5.3	6.5
YAW	0	1.7	0	2.4	0	2.3
ROLL	0.9		1.2		1.1	
MAGNETIC DIPOLE	9	12.4	9	12.4	5.1	17.6

Two types of momentum storage device (MSD) were considered to reduce the propellant penalty; a reaction-wheel type, and a control moment gyro. The propellant savings realized compared to the two MSD weights are tabulated in Table II-7. No final selection of an MSD has been made since many other vehicle-dependent factors must still be considered in the trade-off analysis.

The dynamic interaction of the solar array and the vehicle has been considered only in a preliminary manner. This is a highly complex relationship since it includes the mechanical-response characteristics of the deployed array, the response characteristics of the two-degree-of-freedom, solar-array orientation servo system, the sun-sensor type and location, the vehicle transmissibility and attenuation characteristics, the vehicle three-axis attitude control system performance, and the vehicle attitude stability requirements (see Figure 11-20). This interaction, however, must be thoroughly analyzed before an optimum power system can be designed.

A servo-drive system concept for array orientation has been developed, as shown in Figure 11-21. The system provides control functions for initial sun acquisition, normal tracking (during sunlight and occultation), and reacquisition subsequent to redeployment. The primary control occurs at a constant rate of one revolution per orbit around the boom (μ) axis, while the array-axis (e) rotation only requires movement at a maximum rate of 8 degrees per day.

TABLE II-7. NET PROPELLANT SAVED (3) COMPARED WITH WEIGHT OF MOMENTUM STORAGE DEVICE FOR A WORST-CASE 45-DAY PERIOD (PITCH AXIS ONLY)

MISSION NO.	PROPELLANT SAVED (LBS)	REACTION WHEEL (LBS)	C. M. G. (LBS)
MISSION 1 200 NM, I = 28.5' ORBIT (1 COMPARTMENT)	3240	284(1)	203(2)
MISSION 2 200 NM, I = 28.5' ORBIT (2 COMPARTMENT)	3274	284(1)	203(2)
MISSION 3 200 NM - POLAR ORBIT (2 COMPARTMENT)	3265	284(1)	203(2)

4. Mechanical System Analysis

Evaluation of several possible array-stowage locations revealed that the 4900 square feet of array, could be efficiently and practically stowed only if a shroud and adapter modification were performed. Figure II-22 shows the launch configuration proposed and the modification required for the shroud, shroud/MMSS separation points, and the MMSS-SNB adapter.

A number of existing and proposed solar-array structures were reviewed and considered for possible use. An all-aluminum, honeycomb structure with a curved frame has been selected on the basis of the comparison in Table II-8, and the design-limit loads and load factors were developed. The design permits flat stowage in the launch mode and a curved configuration in the deployed mode. The design chosen, when bowed, has better section properties than a flat configuration. The maximum orbital temperature excursion for this design has been calculated and is shown in Figure II-23.

An orientation drive mechanism, which includes slip rings, torque motors, and a gear-reduction unit, has been defined, and is shown in Figure II-24. Sealing will be accomplished with a labyrinth type seal.

A pneumatically operated, double-ended, retracting cylinder with cables and pulleys, has been selected as the deployment actuator on the basis of weight, size, reliability, force levels, and compatibility. A low-pressure (50 to 100 psi), pneumatic disfxibution system will consist of flexible tubing approximately 0.25 inch in diameter. The deployment sequence with this system will be as shown in Figure II-25 and will require a total time of approximately 10 minutes to complete.

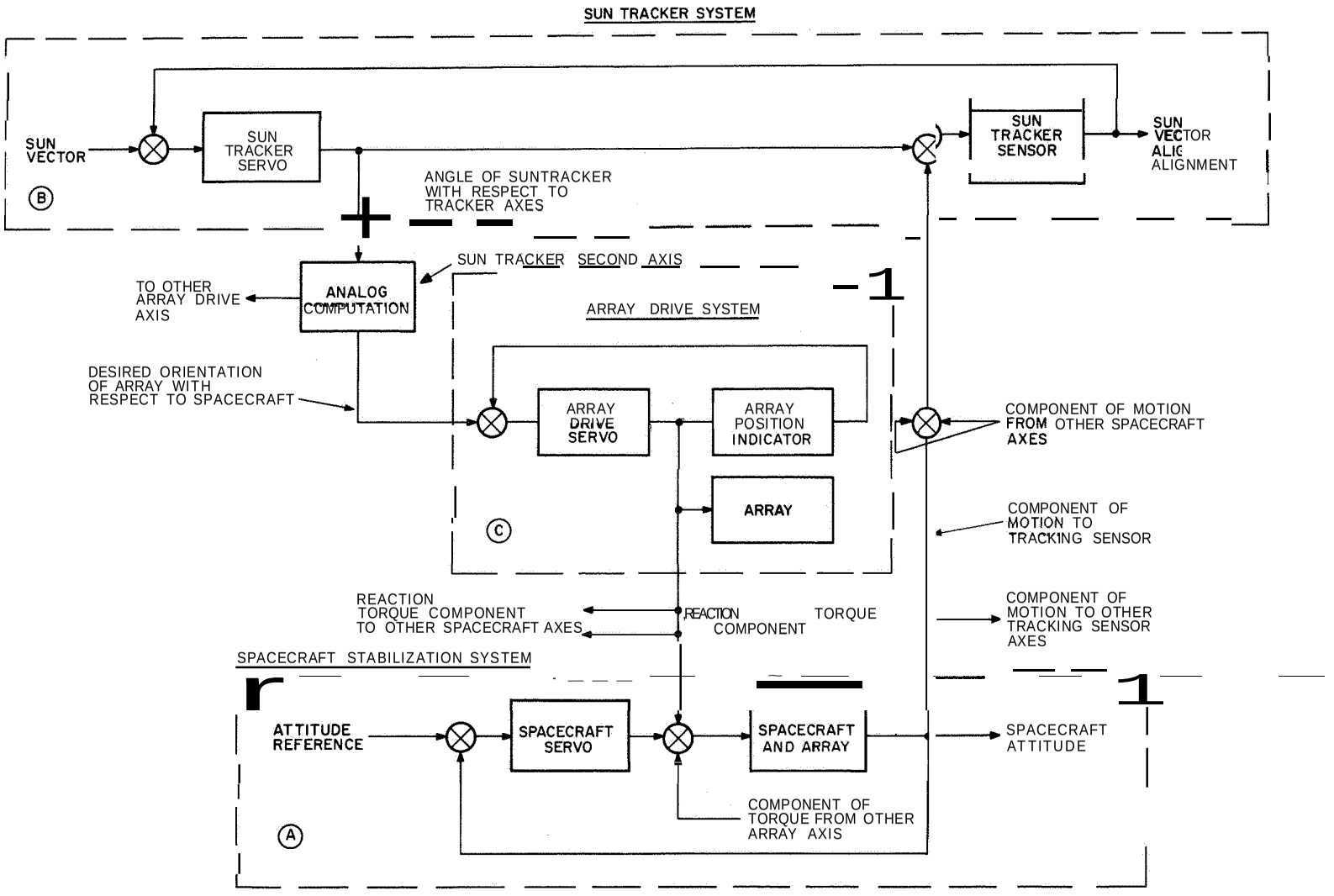


Figure 11-20. Dynamics Control Loop Block Diagram

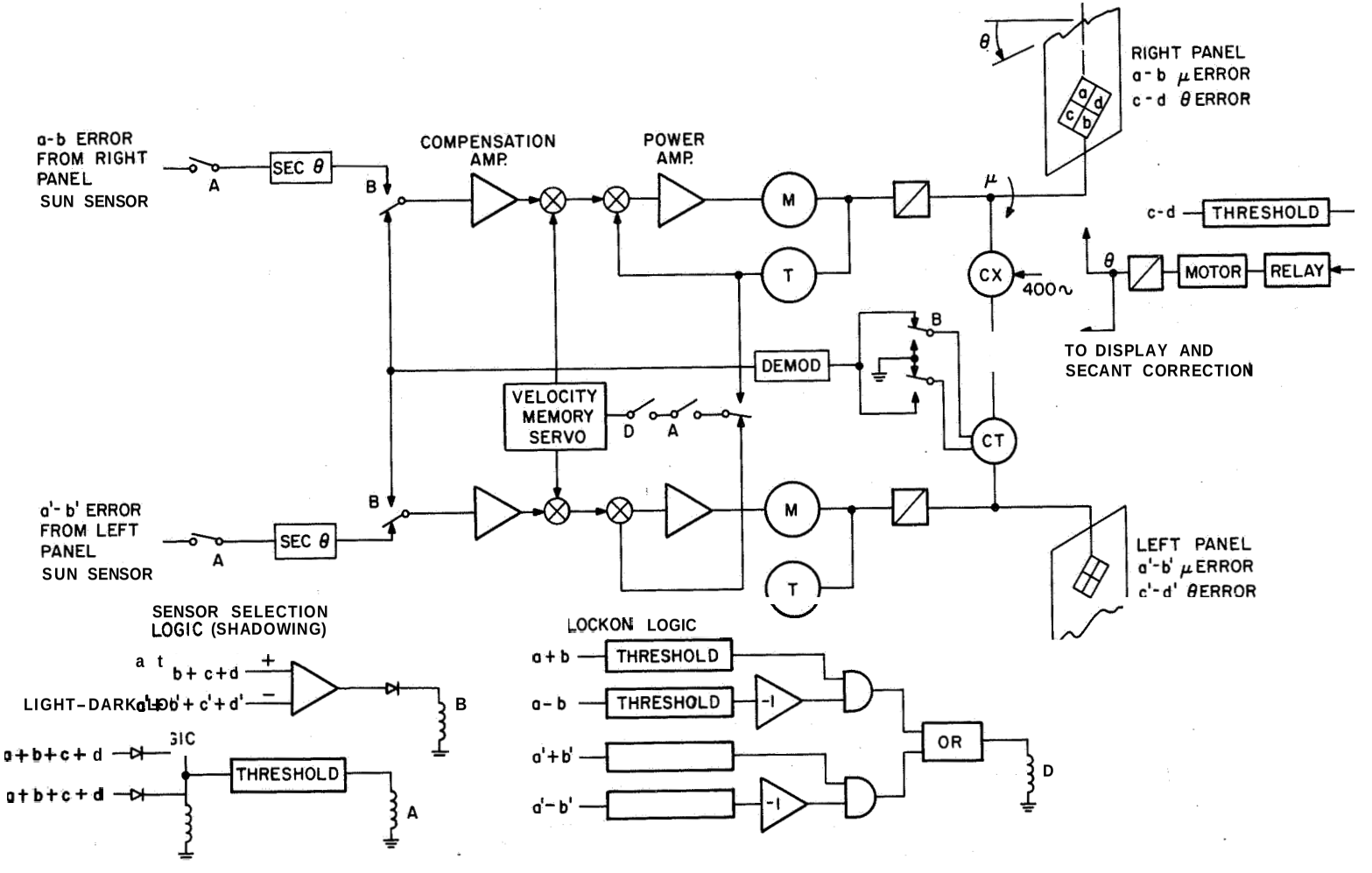


Figure 11-21. Panel Orientation Control System

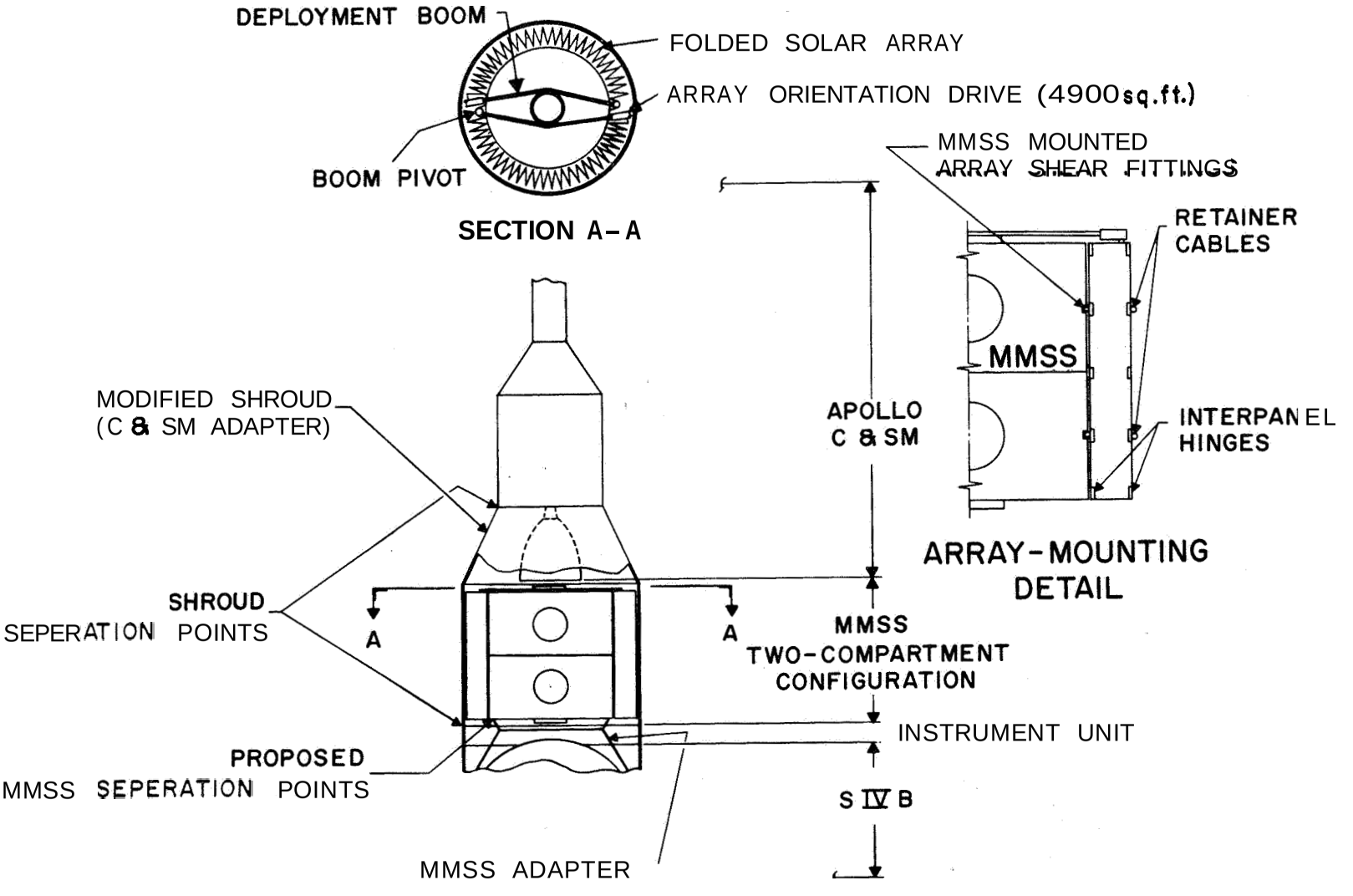
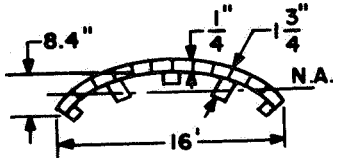
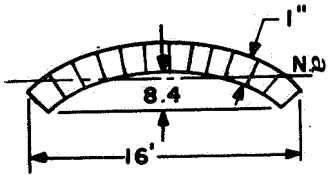
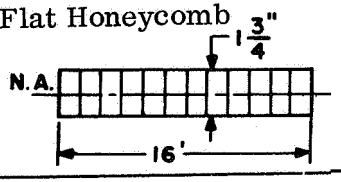


Figure 11-22, MMSS and Solar Array Stowage in Launch Vehicle

TABLE II-8. PANEL DESIGN DATA

Panel Configuration (Material: Aluminum)	Total Array Weight (lbs/ft^2)	Facing Thickness (Inches)	Moment of Inertia ($Inch^4$)	Docking Tip Deflection (Inches)	Margin of Safety (Stress)	
					Stowed	Deployed
Bowed Honeycomb 	0 82	0 0037	21 9	134	(Frame) 0.13 Honeycomb 0.28	(Frame) 1.7 Honeycomb 0.21
B ed Honeycomb 	1 09	0037	17 8	1 0	0 91	0 05
Flat Honeycomb 	3.17	0825	21.92	134	Large	Large
	1.09	0054	1.94	1515	Large	0
	1.71	032	8.8	333	Large	Large

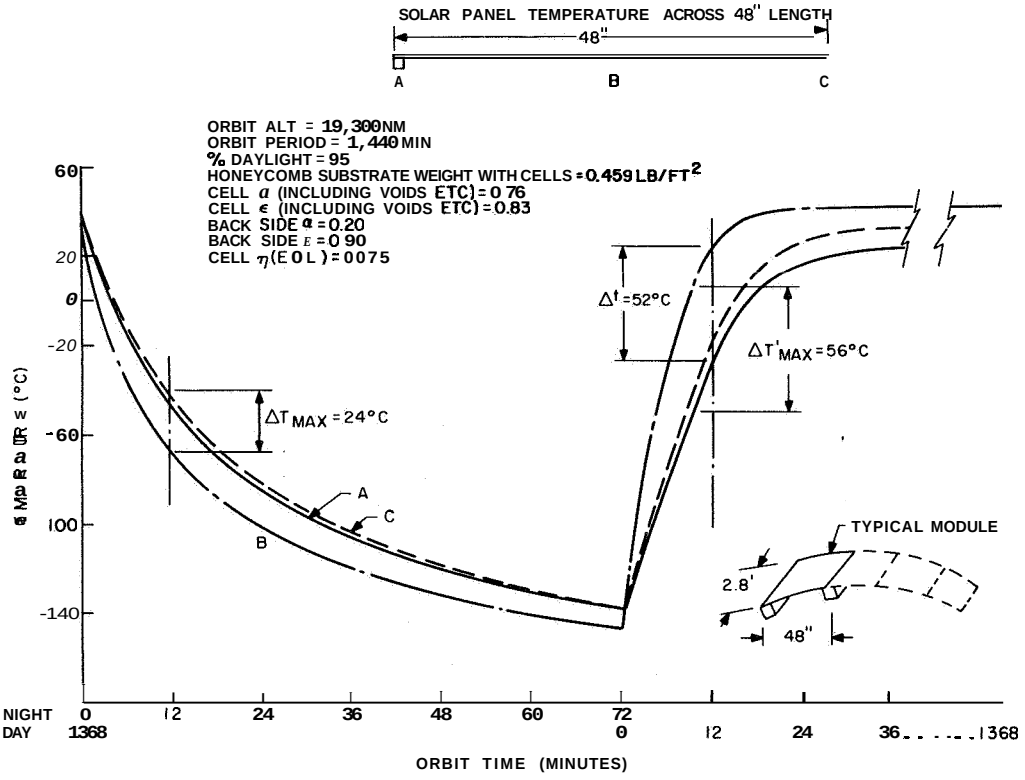


Figure 11-23. Temperature Distribution Across Solar-Panel Length (48-Inch Panel)

With the "Z" array configuration, docking of the Command and Service Module can be accommodated with the array oriented in the weak-axis bending mode (providing maximum vehicle clearance; see Figure II-8). A maximum array-tip deflection of **134** inches results, with the oscillation requiring approximately **4-1/4** minutes to damp out.

The array boom has been sized to withstand SPS-engine firing loads and will be standard 8-inch diameter by 3/8-inch wall aluminum alloy tube. The array is capable of taking the SPS engine firing loads when it is oriented in the tensile mode. In this mode, though, plume effects (both thermal and particle) must be considered further.

C, PSM ANALYSIS

1. General

The Power System Module (PSM) requires **1900** square feet of solar array in an "H" configuration (Figure II-4) to satisfy the specific electrical load requirement. The electrical system for the PSM and its operation are identical to the IPS concept, as is the operation of the array-orientation system.

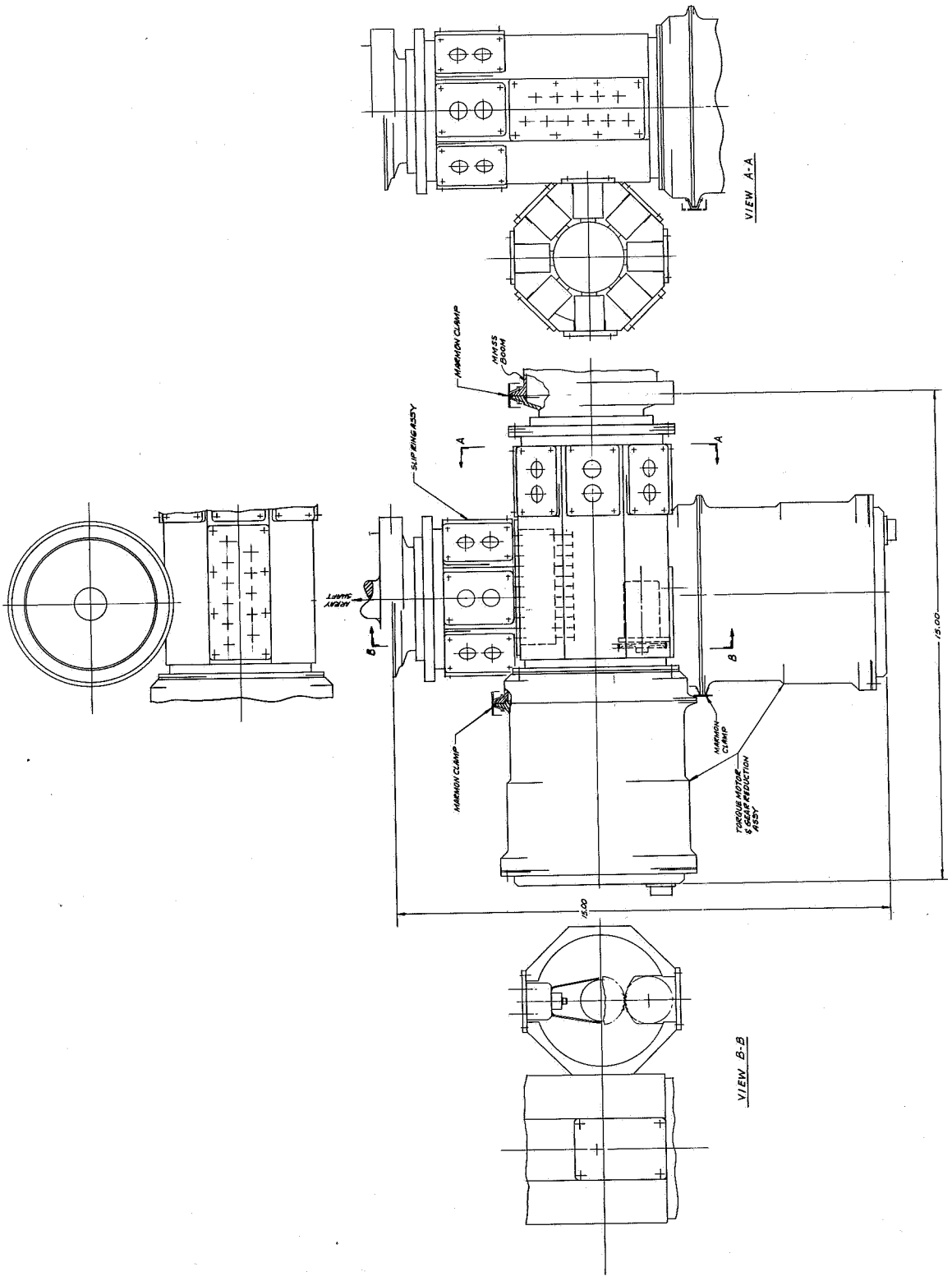


Figure II-24. Orientation Drive Assembly Outline

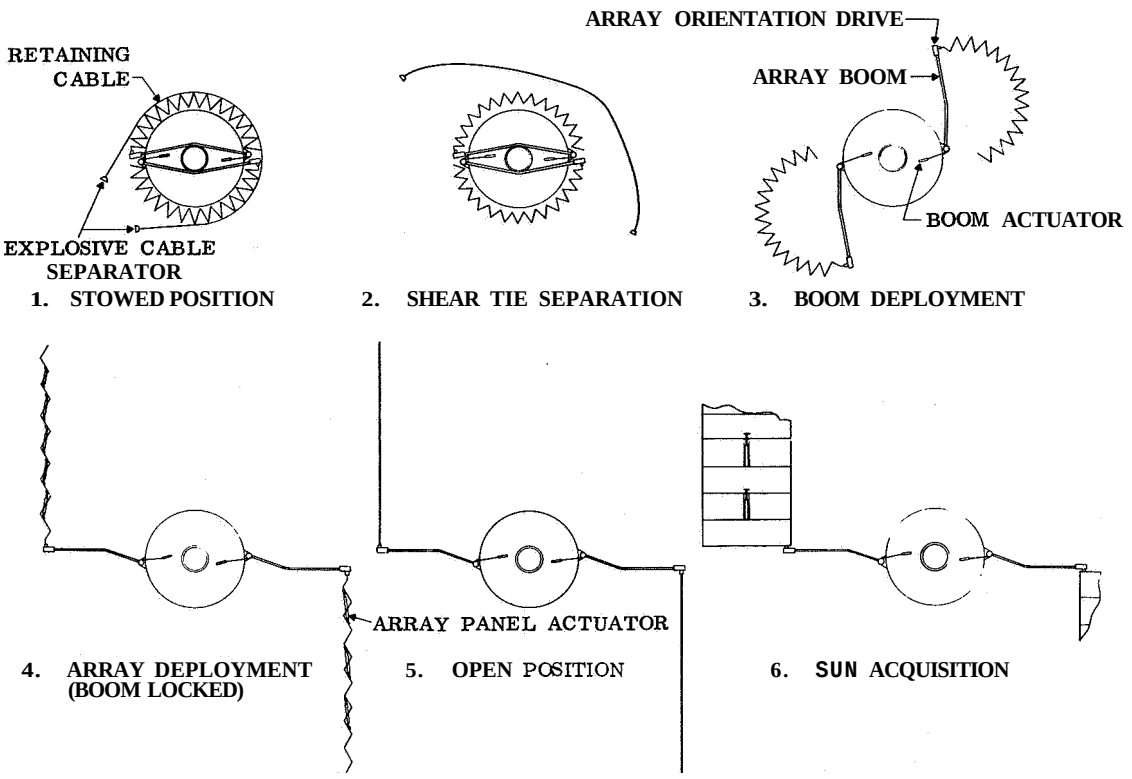


Figure II-25. Solar-Array Deployment Sequence

During launch, the solar array for the PSM is stacked against the component housing assembly (which provides partial support for the docking tunnel) and is secured by tension cables. Upon injection into orbit, the cables are separated and the array boom is deployed with the folded array. The array panels, which are flat honeycomb with an aluminum frame, are then deployed to the "H" configuration as shown in Figure **11-26.**

Although array retraction can be performed, it is not necessary for either Command and Service Module docking or SPS engine firing. For docking, any of the three primary modes is acceptable. For SPS firing, the tensile-compression mode is required. The "H" configuration requires a minimum RCS and SPS fuel penalty to overcome the effects of orbital array perturbations.

2. Electrical System Studies

No additional electrical system study was required **for** the PSM concept since the **IPS** electrical system analysis was performed for a range of load profiles. The specific load profile for the PSM resulted in a system consisting of the components listed in Table **IT-9.**

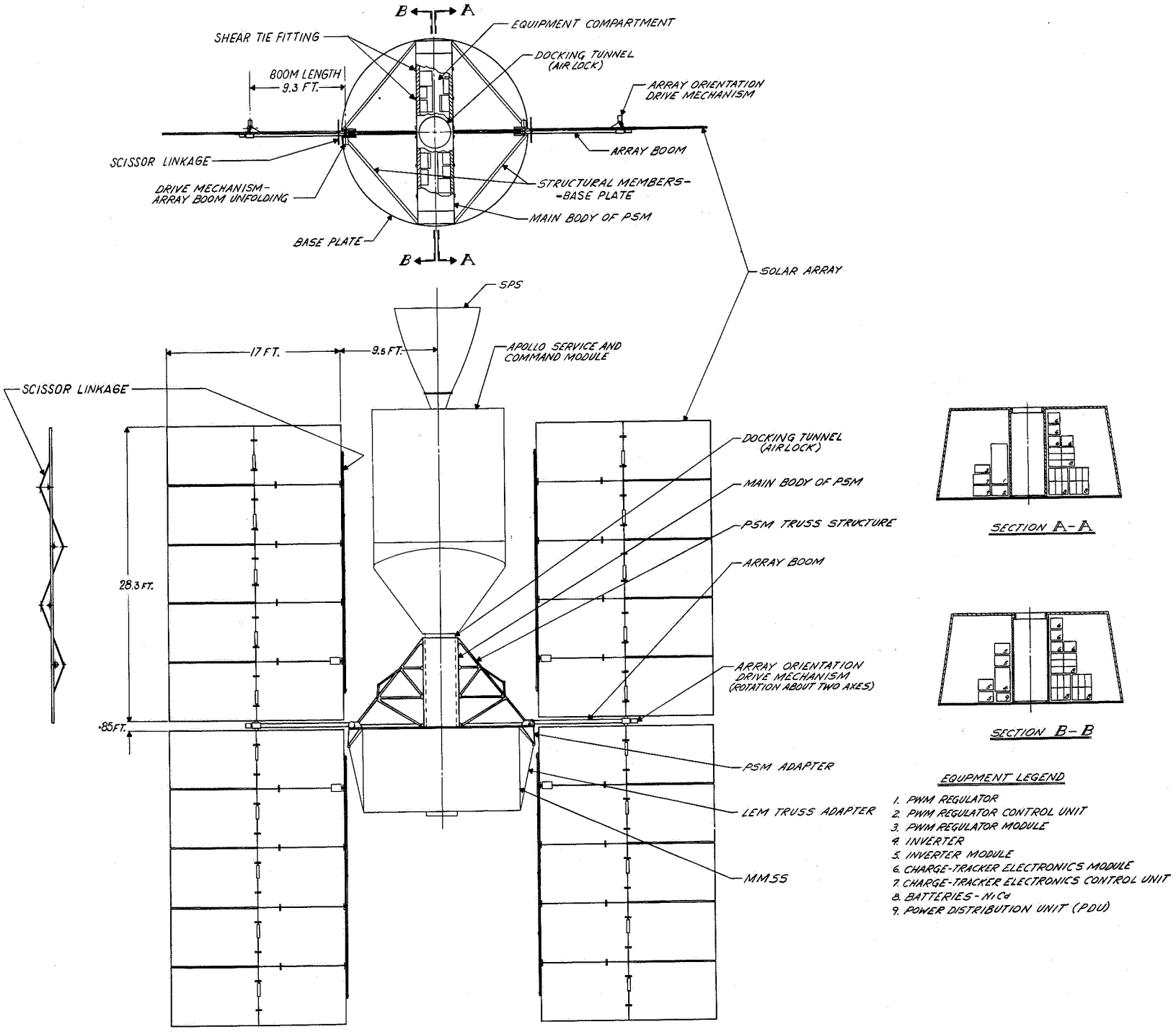


Figure II-26. Orbital "H" Configuration

TABLE II-9. PSM CONFIGURATION

Requirements	Altitude (NM)	
	200	19,340
Array Power Required, watts	15,300	10,400
Array Area, sq. feet	1,800	1,400
Battery Type	Ni-Cd	Ag-Cd
Battery Capacity, ampere-hours	610	410
Active Units		
Battery Units	5*	4†
Charger Control Unit (Figure H-6)	1	1
Charger Modules (Figure H-6)	5	4
Regulator Control Unit (Figure H-4)	1	1
Regulator Modules (Figure H-4)	5	5
Inverter Module (Figure H-5)	3	3
Spare Units		
Battery Modules	5*	4†
Charger Control Unit (Figure H-6)	1	1
Charger Module (Figure H-6)	3	4
Regulator Control Unit (Figure H-4)	1	1
Regulator Module (Figure H-4)	1	1
Inverter Module (Figure H-5)	4	4

* See Figure H-7
† See Figure H-8

3. Orientation System Studies

The perturbation studies were repeated for a 1900-square-foot array in both a "Z" and an "H" configuration. The RCS and SPS fuel requirements for these are shown in Table II-10. Propellant savings and MSD weights are shown in Table II-11.

For the PSM concept, a preliminary shadow analysis was performed. A photographic technique was used to determine shadow effects for an 'H' configuration and the 'Z' configuration with both a long and a short boom. Figures II-27 and II-28 show the orbital average shadow for the two configurations. The 'H' array configuration was selected for the PSM concept on the basis that it provides an acceptable shadow pattern, eliminates the major plume effects from the SPS engine firing, and provides minimum fuel penalty. For the 'H' configuration and a 90-degree inclination, 200-m. mi altitude orbit, the array power at the end of life will be approximately 7 percent less than required for system energy balance for about 5 days twice each year. This condition can be rectified by rolling the vehicle 6 degrees during these periods.

From shadow considerations, there is no great advantage to a boom length of more than 10 feet.

4. Mechanical .System Analysis

For the "H" configuration PSM concept, the 1900-square-foot array stowage and PSM configuration are as shown in Figure II-29. Panel support for each stowed array section will be supplied by a shear tie-cable tension member. Support at the center of the deployment boom will be provided during the launch phase.

The array structure selected (Figure II-30) was based on the planar requirement, the results of the IPS structure analysis, and flight-proven techniques used on the solar arrays for the Lunar Orbiter and Nimbus spacecraft. The design limit loads and load factors developed for the IPS were also utilized to size the PSM array structure.

The panel concept proposed is shown in Figure II-31. In this concept, all panels are connected by hinges to the adjacent panels, thus providing an efficient transfer of loads from the array to the boom.

The array orientation drive assembly is similar to that recommended for the IPS concept except that *two* output shafts are required (because of the 'H' configuration) as shown in Figure II-32. Sealing, lubrication and slip rings are also identical to the IPS concept.

Boom deployment will be accomplished by a spring-powered, fluid-damped actuator. The array deployment sequence is presented in Figure II-33. The phase-I panel-unfolding mechanism has not been selected but may be either a single actuator/scissor linkage, or an individual actuator system. When all primary and secondary hinge axes are aligned, an electrically actuated hinge drive is energized to unfold the array to the flat configuration.

TABLE 11-10. PSM PROPELLANT REQUIREMENTS TO CANCEL ARRAY PERTURBATIONS

PROPELLANT REQUIREMENTS	200 NMI i = 28.50		200 NMI i = 90°	
	WITH MSD	WITHOUT MSD	WITH MSD	WITHOUT MSD
AERODYNAMIC				
DRAG*	1823	1823	1260	1260
LIFT	0	460	0	314
GRAVITY GRADIENT				
PITCH AXIS				
"H" CONFIG. ASPECT = 13.46:1	0	1400	0	1400
"H" CONFIG. ASPECT = 3.36:1	0	313	0	313
"Z" CONFIG. ASPECT = 3.36:1	0	1400	0	1400
ROLL AXIS				
"Z" CONFIG. ASPECT = 3.36:1	910	910	910	910
SOLAR PRESSURE				
PITCH	8.6	9.9	4.9	8.2
ROLL	3.1	65	5.5	16.9
YAW	0		0	
MAGNETIC DIPOLE	16.5	38.4	11.0	55

TABLE 11-11. PSM PROPELLANT SAVINGS ANALYSIS

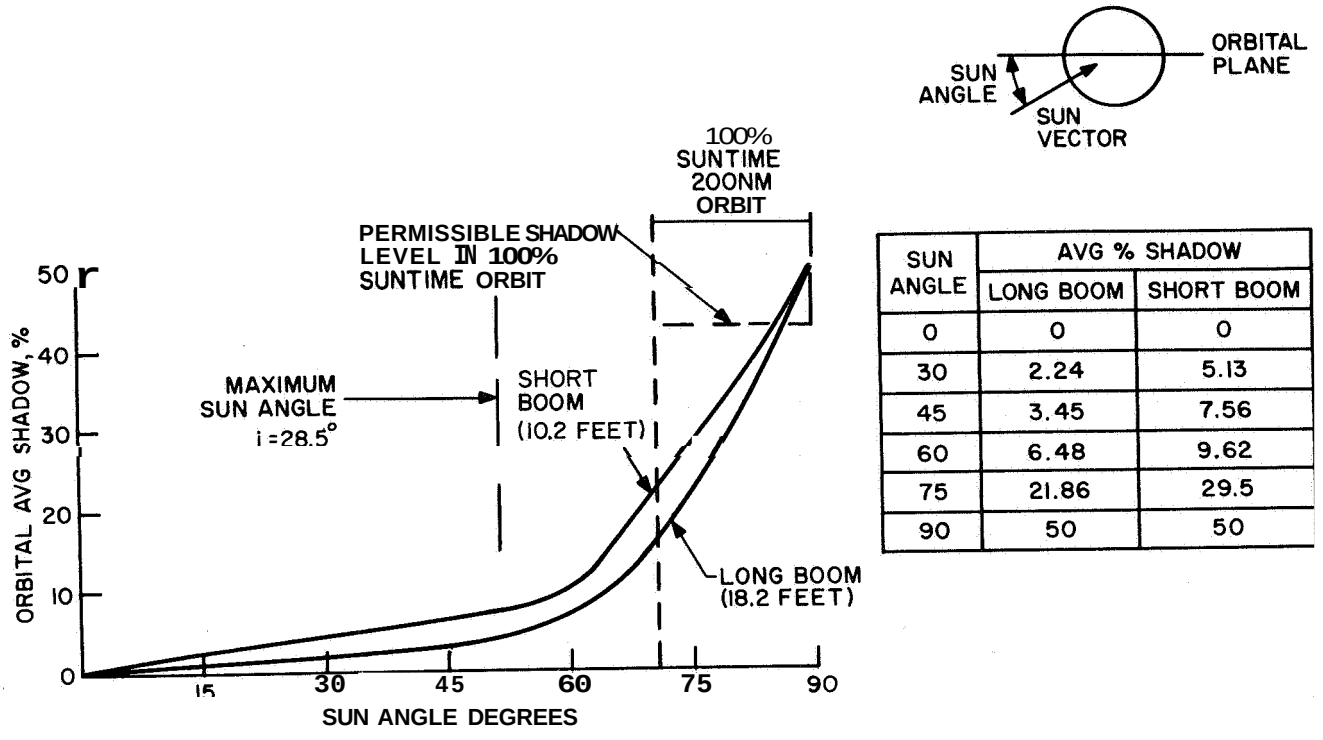
PITCH AXIS ONLY			
200 NM I = 28.5°	PROPELLENT SAVED (LBS.)	REACTION WHEEL (LBS.)	C. M. G. (LBS.)
"Z" CONFIGURATION ASPECT: 3.36:1	1553	35(1)	74(3)
"H" CONFIGURATION ASPECT: 13.46:1	1553	35(1)	74(3)
"H" CONFIGURATION ASPECT: 3.36:1	466	16.5(2)	30(4)
200 NM POLAR ORBIT			
"Z" CONFIGURATION ASPECT: 3.36:1	1505	35(1)	74(3)
"H" CONFIGURATION ASPECT: 13.46:1	1505	35(1)	74(3)
"H" CONFIGURATION ASPECT: 3.36:1	418	16.5(2)	30(4)

(1) 18 LBS CHARGED TO POWER SUPPLY

(2) 10 LBS. CHARGED TO POWER SUPPLY

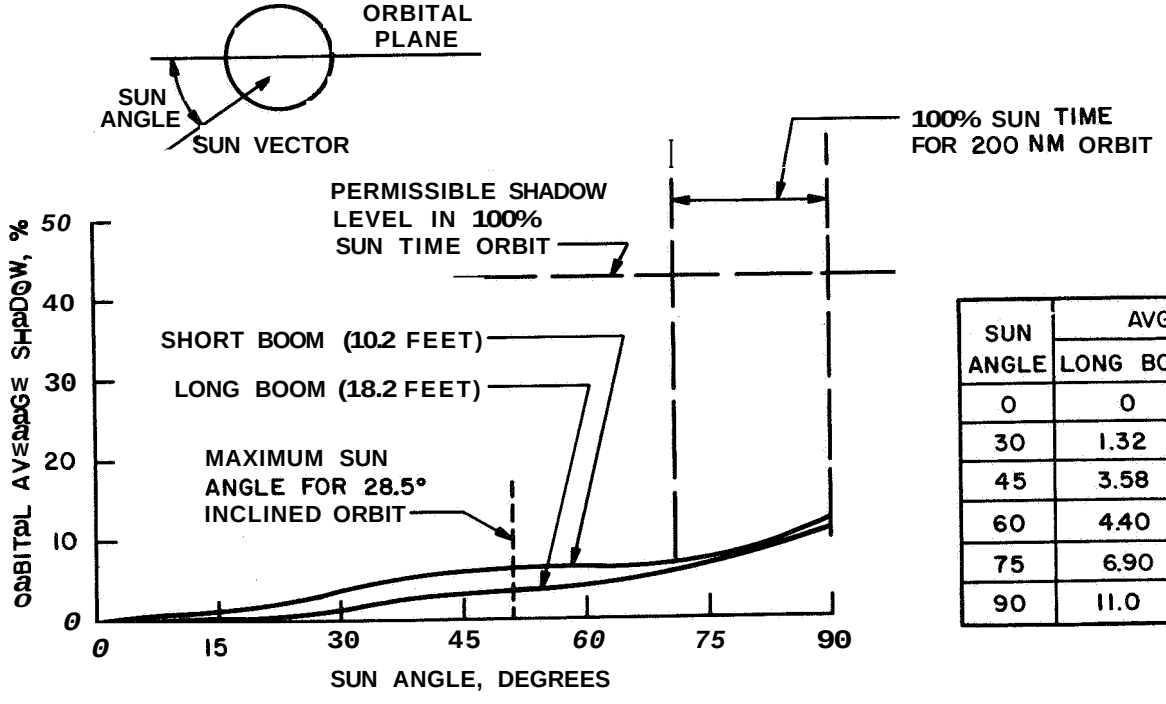
(3) 7 LBS. CHARGED TO POWER SUPPLY

(4) 3 LBS. CHARGED TO POWER SUPPLY



SUN ANGLE	AVG % SHADOW	
	LONG BOOM	SHORT BOOM
0	0	0
30	2.24	5.13
45	3.45	7.56
60	6.48	9.62
75	21.86	29.5
90	50	50

Figure 11-27. PSM Concept H Configuration Shadowing, Aspect Ratio: 3.46:1



SUN ANGLE	AVG. % SHADOW	
	LONG BOOM	SHORT BOOM
0	0	0
30	1.32	3.95
45	3.58	6.20
60	4.40	6.50
75	6.90	7.28
90	11.0	12.0

Figure 11-28. PSM Concept Z Configuration Shadowing, Aspect Ratio: 3.46:1

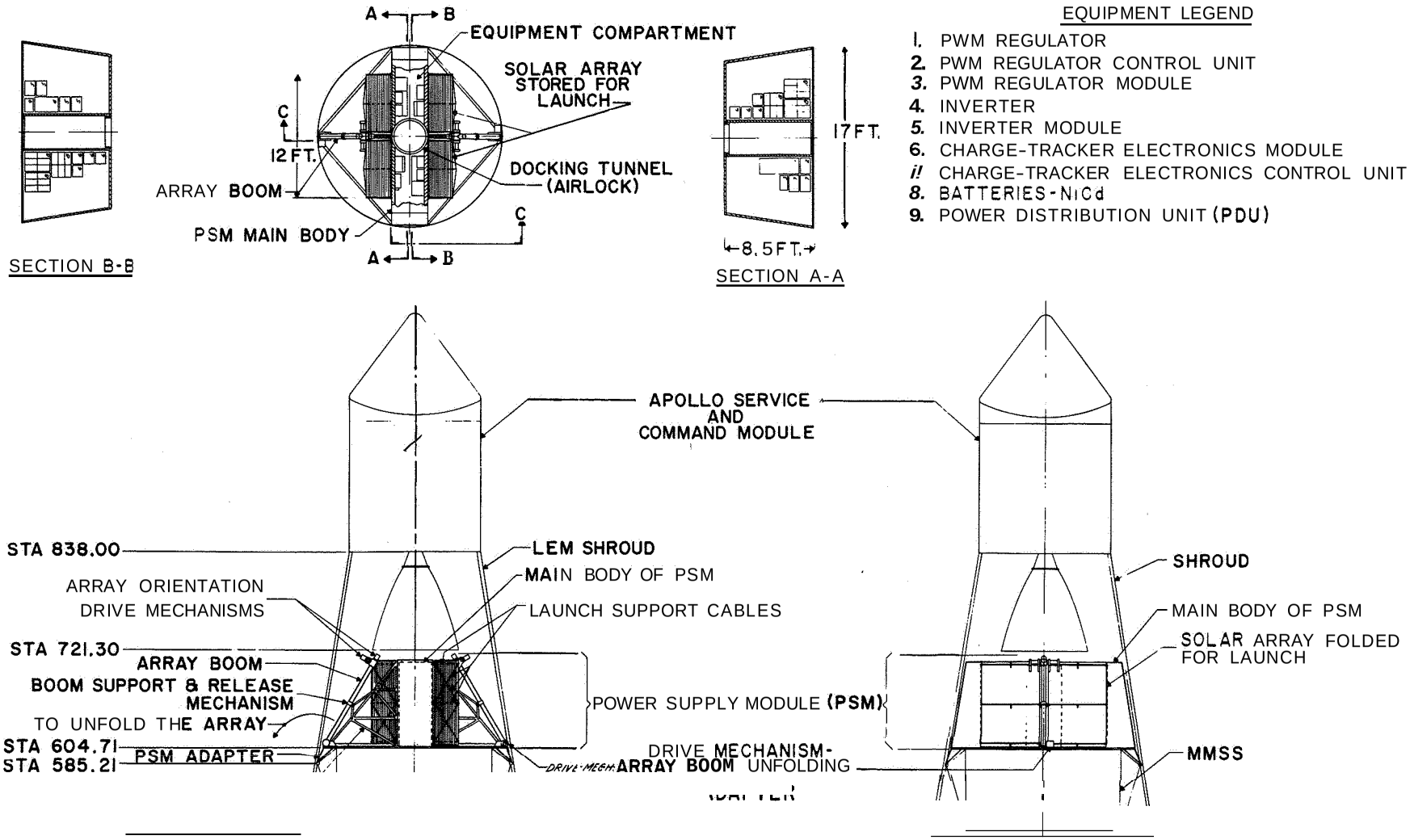


Figure 11-29. PSM Launch configuration (H Configuration Array)

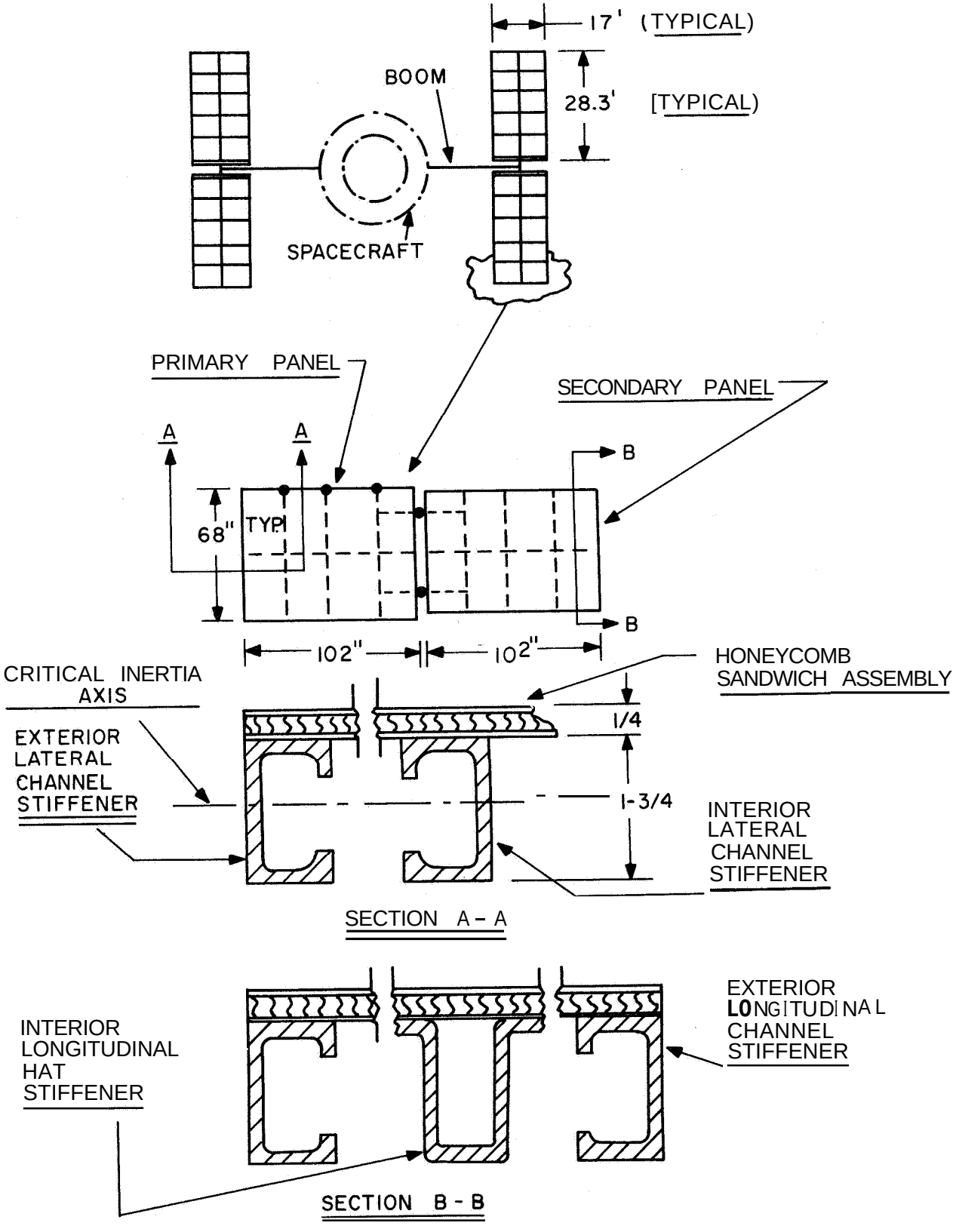
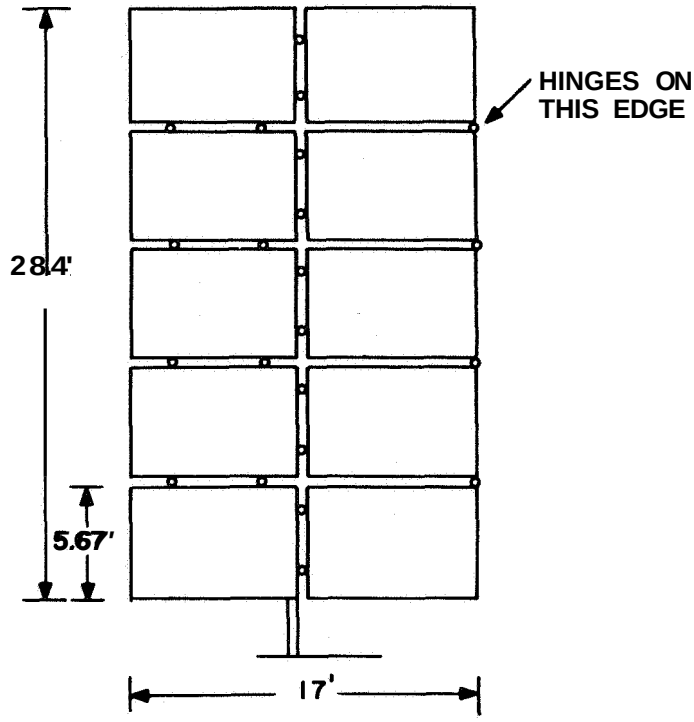
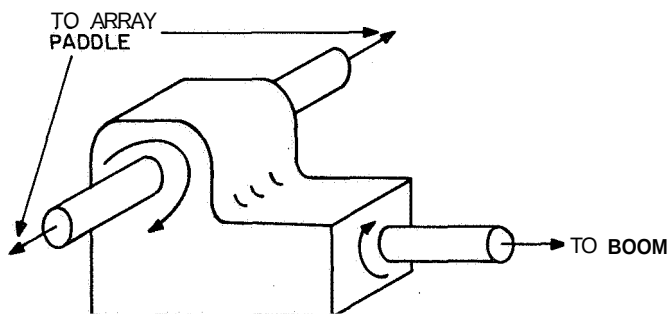


Figure 11-30. Vehicle Solar-Array Details, Normal Position



FINAL CONCEPT

Figure 11-31. Selected PSM Panel Concept



HOUSING CONTAINING MOTORS,
GEAR REDUCTION UNITS AND
SLIP RINGS

Figure 11-32. Schematic Representation of Orientation Drive Assembly

Retraction, when necessary, will be performed in a reverse sequence. The requirements for retraction must be further defined since the array can withstand SPS firing in the deployed configuration. The hinge lines though, will require spring loaded latching mechanisms with electro-magnetic unlatching for retraction.

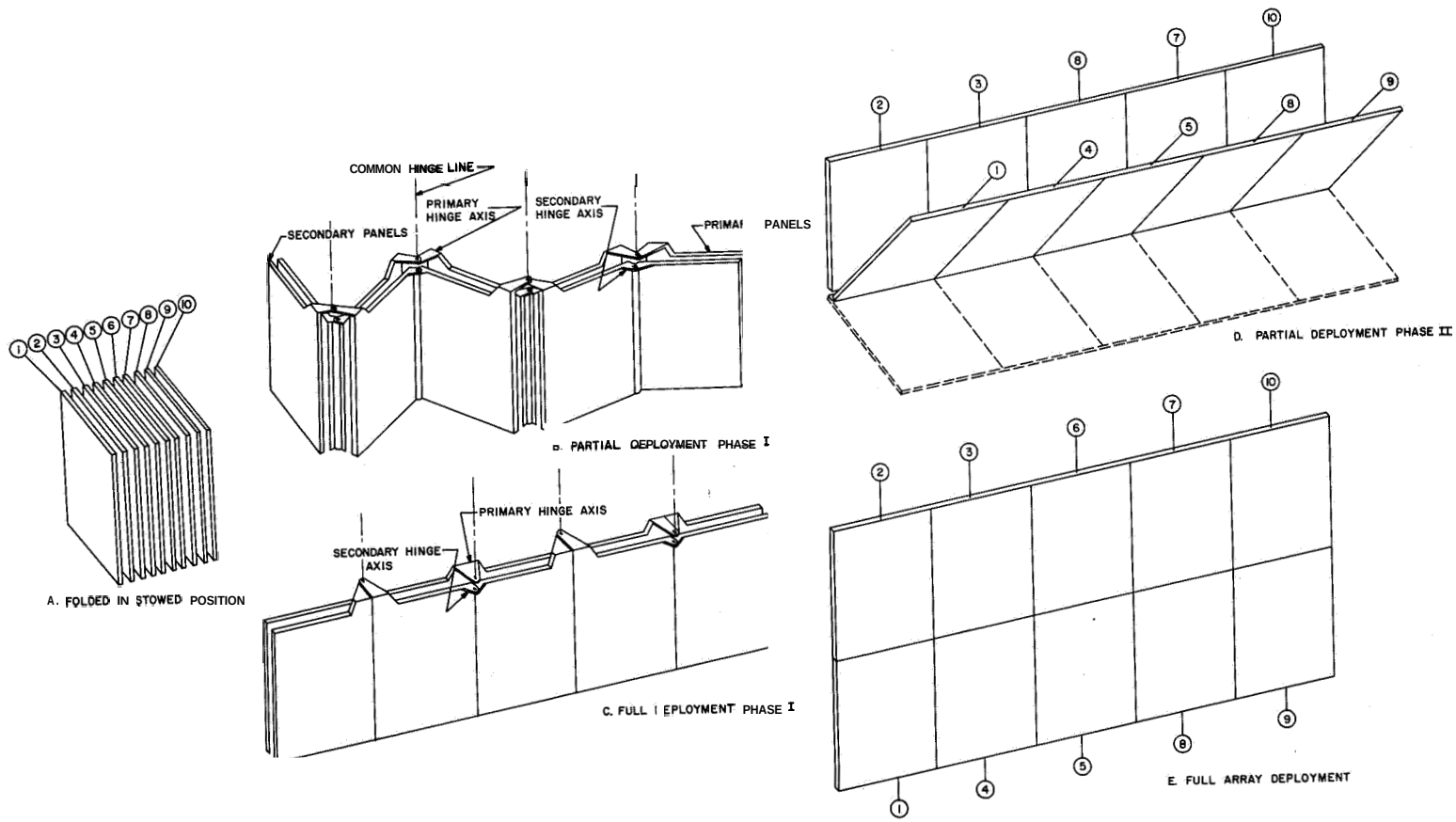


Figure 11-33. PSM Array Development Sequence

The orbiting configuration of the PSM is shown in Figure **II-26.** Command and Service Module docking to the orbiting vehicle can be accommodated with the array oriented in all three primary modes--tensile-compression mode, weak-axis bending mode, and strong-axis bending mode. The two latter modes, appear to be the most desirable since they afford the greatest clearance for docking.

SPS firing loads can be safely handled by the 'H' configuration array in the tensile-compression mode. The array boom has been sized for these loads. The boom will be a 4-inch-diameter by 1/2-inch-thick wall aluminum alloy tube.

Since battery and electronics modules will be mounted on the internal walls of the component housing assembly (as shown in Figure II-29), the outer surface of the component housing assembly will provide the radiating surface area for thermal control. Based on a comparison of two surface finishes, second-surface mirrors are recommended.

Table II-12 shows the results of a system analysis for thermal control. Since an active control system is required to maintain the batteries within the permissible temperature range of 15 to 30°C, several methods for active control have been considered; however, no recommendation can be made at this time. Passive temperature control may be acceptable for the lower orbits, if the battery permissible temperature range can be increased and if the vehicle may be rotated around its roll axis.

The power conditioning electronics and batteries'will require only 16 percent of the available 325 cubic feet available in the component housing assembly. This leaves adequate volume for other PSM-related equipment or for power-system growth.

TABLE II-12. RESULTS OF SYSTEM THERMAL ANALYSIS

ORBIT	VEHICLE OPERATING MODE*	POWER CONDITIONING ELECTRONICS (OPERATING TEMPERATURE LIMITS 0 TO 55°C)			BATTERIES (OPERATING TEMPERATURE LIMITS 15 to 30°C)					
		PASSIVE TEMP. CONTROL			PASSIVE	EMP. CONTROL		ACTIVE TEMP. CONTROL		
		AREA (FT^2)	MAX (°C)	MIN (°C)	AREA (FT^2)	MAX (°C)	MIN (°C)	AREA (FT^2)	MAX (°C)	MIN (°C)
200 NM POLAR ORBIT	I	76	55	15	110	30	-13	110	30	15
200 NM POLAR ORBIT	II	76	55	18	74	30	8	74	30	15
SYNCHRONOUS: 28 1/20 INCLINED	I	85	55	20	5	30	-10	5	30	15

NOTE: RADIATOR SURFACE FINISH SECOND SURFACE MIRRORS IN ALL CASES

*MODE I - NORMAL
MODE II - VEHICLE ROTATED AROUND ROLL AXIS OCCASIONALLY

The radiator area required for the particular mission studied is only 60 percent of that available, thus providing for additional equipment or power-system growth.

The array stowage volume available provides for a 42-percent growth from the **1900** square feet of array required for the particular mission studied.

D. COMPUTER PROGRAM AND PARAMETRIC STUDIES

A computer program prepared during the study permits power-system Component sizing as a function of mission parameters, utilizing component parametric information as stored data. A generalized flow diagram is of this program shown in Figure **II-34.**

The mission requirements of load power, altitude flux, and resupply interval are programmable inputs. The stored component information is an input which is a relatively permanent record that can be periodically updated as better data are made available. Table **TI-13** lists the inputs which the user must specify each time a computer run is to be made.

The program outputs include the quantities which are the result of the computer calculations performed. These are printed out in the format shown in Table **II-14,** along with the input parameters selected by the user.

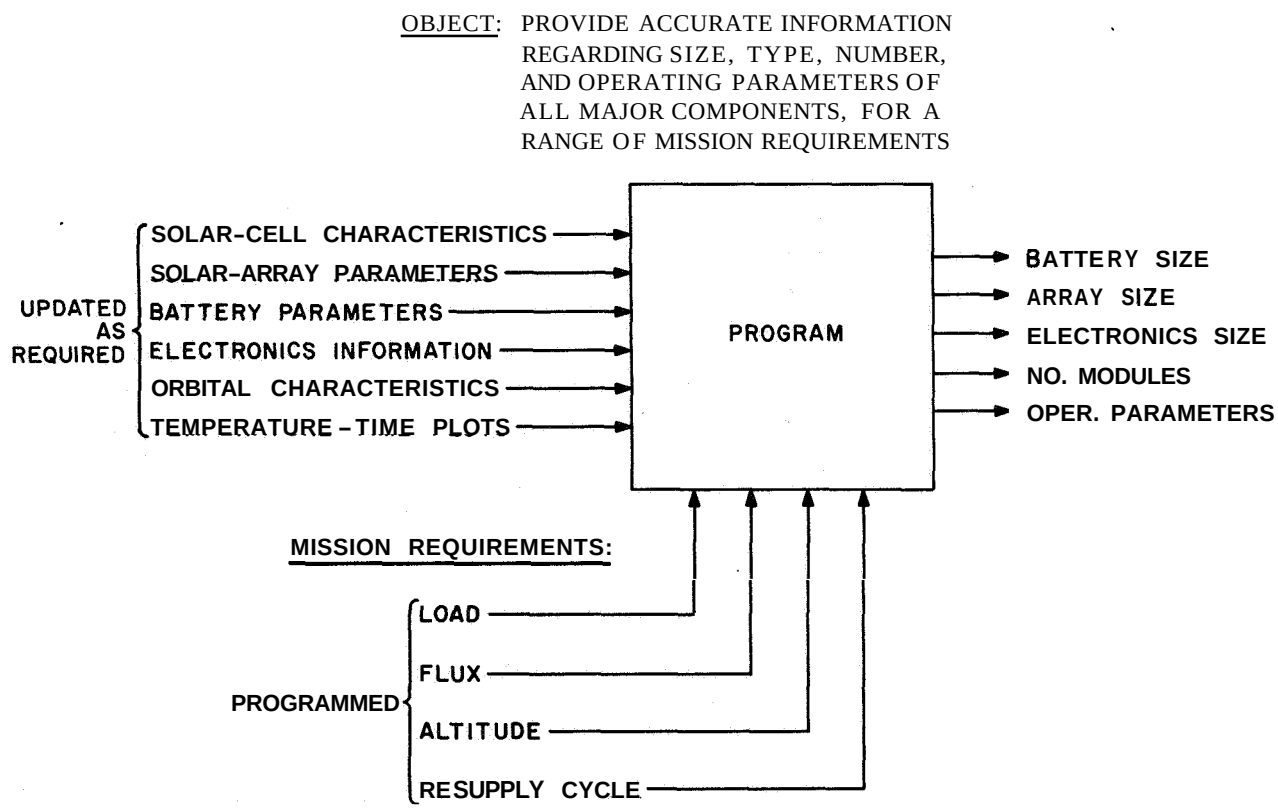


Figure 11-34. Computer Program Flow Diagram

TABLE II-13. COMPUTER PROGRAM INPUTS

N-Code Card No.	Numerical Value Punched
1	Daytime load in watts
2	Nighttime load in watts
3	Fraction of daytime load at dc
4	Fraction of nighttime load at dc
5	Resupply (fraction of a year)
6	Flux (1 MeV electrons). Three options exist, with numerical value punched on card No. 6 indicated to identify each option. See cards No. 12 and 13. (a) Value: -1.0. Calculate the equivalent 1-MeV electron flux from raw flux data stored. Value punched on card No. 12 is 1.0, to initiate STASH subroutine in the solar-cell program. Card No. 13 is punched 0.0. (b) Value: 1-MeV electron-flux value chosen for the solution, between limits of 0.0 and 10^{16}. For instance, 5.0 x 10^{14} is punched on the card as 500000000000000.0. PHI subroutine in the solar-cell program will not be initiated. Card Nos. 12 and 13 as in (a). This option is used if a 1-MeV flux value, other than that resulting from the raw flux data stored, is assumed. (c) Value: as in "b". This option is used if it is desired to use P_{max} -vs.-temperature-vs.-flux solar-cell data stored in the Main Program (see Paragraph X. E. 2. b(3)(c)1), rather than data calculated by the Solar-Cell Program. Card No. 12 is 0.0, to inhibit STASH; card No. 13 is 28.0 (special code corresponding to the serial number of the solar-cell data table stored in the Main Program).
7	Altitude, nautical miles

TABLE 11-13. COMPUTER PROGRAM INPUTS (Continued)

N–Code Card No.	Numerical Value Punched
8	Coverglass thickness, mils. If options 6b or c are exercised, the numerical value of the 1–MeV flux punched is assumed to be the correct value for the glass thickness punched on card No. 8.
9	Backshielding thickness, mils of aluminum. Same comment applies as for the glass thickness under options 6 b or c.
10	Solar-cell base resistivity, 1.0 or 10.0 ohm-cm cell data stored in either the Solar–Cell or the Main Program, depending on option selected on card No. 6.
11	Battery type. Punched value: 0.0, Ni–Cd; 1.0, Ag–Cd.
12	Value punched: 1.0, to initiate a new STASH solution; 0.0, to use the output of a preceding STASH solution, an option exercised to save computer running time in cases where no changes in solar-cell parameters are called for in a set of several consec–utive runs.
13	Source of solar-cell data. Punched value: 0.0, use output of STASH (solar-cell program); 28.0, use cell data stored in the Main Program.
14	Program selection. Four options exist: (a) Value: 1.0. Exercise the Solar–Cell Program only. Output is a single solution of the STASH subroutine. (b) Value: 2.0. Another Solar-Cell Program run, different data. Option used when more than one consecutive STASH subroutine solution is required. (c) Value: 3.0. Exercise the entire program, including Main. Single solution, then stop. (d) Value: 4.0. More than one solution in sequence, involving the entire program including Main.

TABLE II-14. COMPUTER PROGRAM OUTPUT FORMAT

	VALUE
ARRAY DESIGN PARAMETERS	
AVERAGE ARRAY POWER REQUIRED BY SYSTEM (KW)	15.287
PEAK USEFUL ARRAY POWER (KW)	19.008
AVERAGE ARRAY POWER USED (KW)	15.275
MIN. ARRAY POWER REQUIRED TO SATISFY DAY LOAD ONLY (KW)	9.704
PERCENT OF AVAILABLE ARRAY ENERGY USED	98.79
MINIMUM ARRAY POWER (KW)	14.488
MAXIMUM ARRAY POWER (KW)	23.85
ARRAY AREA (SQUARE FEET x 1000)	1.838
ARRAY WEIGHT (POUNDS x 1000)	1.367
NUMBER OF SOLAR CELLS REQUIRED	2.663
TOTAL EQUIVALENT 1 MeV FLUX USED IN THIS DESIGN	0.417314
BATTERY DESIGN PARAMETERS	
CAPACITY (AMPERE-HOURS)	610.9
MAXIMUM NUMBER OF C/D cycles	5740
MAXIMUM DEPTH-OF-DISCHARGE (%)	16
REQUIRED AVERAGE CHARGE RATE (HOURS)	5
CALCULATED AVERAGE RATE (HOURS)	5.01
EFFECTIVE C/D RATIO	1.162
BATTERY WEIGHT (POUNDS)	2761
BATTERY VOLUME (CUBIC FEET)	60.23
NUMBER OF BATTERY PACKS	5
ELECTRONICS VOLUME AND WEIGHT	
PWM + INVERTER + CHARGE ELECTRONICS WEIGHT (POUNDS)	488.5
VOLUME OF ABOVE (CUBIC FEET)	11.01
NUMBER OF MODULES	
NUMBER CHARGE MODULES	5
NUMBER OF PWM MODULES	4
NUMBER OF INVERTER MODULES	2
SUMMARY OF MAX. POWER AND CURRENTS	
PWM (WATTS) INVERTER (WATTS) CHARGE MODULES (AMPS)	
1370 1368 40.73	
TOTAL SYSTEM WEIGHT (POUNDS)	4617
ARRAY POWER VS. TIME CURVE AT 1 MINUTE INCREMENTS.	

The computer program is essentially two routines coupled together. The MAIN routine deals with system design and contains subroutines for battery, electronics, and solar-array calculations. The SOLAR CELL routine is used to determine solar-cell performance (as influenced by the environmental parameters encountered during a given mission) and contains subroutines for (1) converting charged-particle raw flux to an equivalent 1 MeV electron flux as a function of cover-glass and backshielding thickness (subroutine **PHI**), (2) degrading the solar-cell output characteristics (I-V curves) to account for the reduction of output power caused by the charged-particle flux (subroutine DEGRADE), and (3) operating on the solar cell I-V curve to account for all other degradation factors such as current, voltage and series resistance factors and temperature effects (subroutine STASH). The output of subroutine STASH is utilized as an input option in the MAIN program.

The **SOLAR** CELL routine may be used separately or in conjunction with the MAIN routine to determine the effective, degraded solar-cell output by employing either an assumed value of 1 MeV flux (in which case the **PHI** subroutine is by-passed) or by calculating the 1 MeV flux from raw **flux** data supplied as numerical values of the number of protons and electrons encountered in the radiation belts and the number of protons and alpha particles encountered in solar flares. In either case, the resultant cell data are printed out or fed to the MAIN program.

This power-system computer program has been developed as a preliminary design tool. It does not automatically select the best component type but permits sequential solutions which may be used to establish optimum component requirements. Many of the internal functions performed rely on weighted average values of component characteristics. A wide range of mission requirements can be accommodated, but the program has been written for a mission in which there are both daytime and nighttime power loads. Finally, the logic and numerical values utilized are based on a parallel-tracker power-system configuration. It may be utilized for other system configurations provided program modifications are made.

E. PARAMETRIC STUDIES

The computer program described has been utilized to provide data which would yield an appreciation of the effect of mission-parameter variations on power-system parameters. The variable parameters considered during this study were resupply interval, type of battery (Ag-Cd, Ni-Cd), nighttime load, fraction of load at DC, and ratio of daytime load to nighttime load. Individual and composite plots showing the effect of the above variables were generated. (Only baseline system requirements are considered. No redundancy is included in weight or volume magnitudes.)

All of the power-system trends and magnitudes of component size, weight, and volume show a linearly increasing value with nighttime load. For a given nighttime load, varying the fraction of load at DC between 0 and 1.0, or varying the ratio of daytime load to nighttime load from **1.3** to 1.7 produced a change in the total system weight of less than **16** percent and 8 percent, respectively.

The total power-system weight in the synchronous orbit for a given set of load conditions is only about 50 percent of that in the 200-n. mi. orbits primarily because higher energy-density Ag-Cd batteries can be used and charged at a lower rate, with a resultant reduction of both battery weight and solar-array weight.

F. CONCEPTUAL DESIGN

A power-system with a conceptual electrical design as shown in Figure 11-7 is recommended for low-altitude missions. The mechanical system is shown in detail in Figure 11-29, while the fuel penalties are shown in Table II-10.

This conceptual design was selected for the reasons outlined in the previous paragraphs. Some of the significant features of this conceptual design are:

(1) No array retraction is required for SPS firing.

(2) The total system weight will be approximately 6850 pounds (not including PSM structure, docking tunnel, or component housing assembly).

(3) The array (at the beginning of life) produces 14 watts per pound.

The recommendations outlined in Section IV are made with this power-system concept in mind.

G. TOTAL POWER SYSTEM WEIGHT

Although a computer program has been provided to calculate 'total system weight', a number of factors have not been included. The total weight indicated by the computer program is for a baseline power system which has not been reliability enhanced. In addition, no array-deployment mechanism or orientation-drive system have been included. Furthermore, for a PSM concept, no weight is allowed for the thermal-control system.

In order to show how a more realistic weight estimate may be made, an example of the additional calculations necessary to determine "Total Power System Weight" is included.

1. Hypothetical Problem

A manned laboratory is to be orbited for one year in a polar orbit at an altitude of 200 n. mi. The DC load power requirement is 3.34 kW during orbit night and 5.47 kW during orbit day. The AC power required is 1.66 kW during night and 2.75 kW during the day. No resupply is planned for the one-year mission.

Based on these constraints, the problem is to determine the total power-system weight for an IPS concept and for a PSM concept.

For the Integrated Power System, the physical properties of the orbiting vehicle and launch vehicle are such that an accordion-stowed, curved-panel solar array is utilized.

For the Power System Module, the flat-panel concept will be used.

2. IPS Solution

There are nine steps to be performed in computing the total power system weight for the IPS concept.

(1) Definition of Power System Elements

Based on the mission and vehicle description, specific power-system characteristics must be established. First, a parallel-tracker power system configuration will be assumed. Second, the RCA-generated radiation flux levels for 200-n. mi. altitude and an orbit inclination of **90** degrees will be used. Third, the solar array will be the IPS mechanical concept, with the solar-cell type being **1** ohm-cm base resistivity, N-on-P cell. Fourth, the coverglass will be 6-mil fused silica. Fifth, the backshielding (which is expressed in equivalent mils of aluminum) will be **15** mils. And, last, the battery type, selected from Figure **IC-12,** will be nickel-cadmium.

(2) Computer Program Inputs

The computer-program inputs prepared for this problem, will be as follows.

N-Code Card No.	Name	Value
1	Daytime load	8.22
2	Nighttime load	5.00
3	Fraction of daytime load at DC	.667
4	Fraction of nighttime load at DC	.667
5	Resupply (fraction of year)	1.00
6	Flux Option*	-1.0
7	Altitude	200.00
8	Coverglass thickness*	6.00
9	Backshield thickness*	15.00
10	Solar-cell base resistivity*	1.00
11	Battery-type option*	0.0
12	Use option*	1.0
13	Solar-cell data option*	0.0
14	Program-selection*	3.0

*See Table **IC-13** and Section **X** for limitations and program options.

(3) Computer Program Outputs

The computer will print out the results of the run in the format shown in Table **E-14.** From this, the following information is excerpted:

Component	Number Required
Batteries	5
C/T Modules	5
PWM Modules	4
Inverter Modules	2

The total system weight, **4617** pounds, applies for a system without reliability enhancement and without fuel penalty.

(4) System Reliability Enhancement

From Table **E-5,** the number of component spares required **for** each system functional element is determined. These are as follows.

5 baseline batteries	require **3** spare battery modules
5 baseline C/T modules	require **3** spare C/T modules
4 baseline PWM modules	require **2** spare PWM modules
2 baseline inverters	require **4** spare inverters

One each of the spare PWM and inverter modules will be active; the remainder will be stored. All spare C/T and battery modules are stored. Both C/T and PWM regulator modules will require redundant control and failure-detect modules.

Storage-cell capacity is determined by dividing the total required ampere-hours of capacity (from the computer print-out) by the number of batteries. Therefore, the nickel-cadmium storage cells will be required to provide **610.9/5 or 122** ampere-hours each.

(5) Electronics Weight and Volume

The weight and volume of the system electronics, enhanced for reliability, must now be calculated. Table **11-15** provides the per-spare-module additional weight and volume factors. Utilizing these values and the spare requirements of Table **11-5,** the total additional electronics weight can be calculated, as follows:

C/T

1	Spare control and failure-detection module	**16**
3	Spare power modules **(3** x **25)**	75

Regulator

1	Spare control and failure-detection module	15
2	Spare power modules (2 x 30)	60

Inverter

4	Spares (4 x 70)	280
	Added Electronics Weight	446 pounds

The added volume is calculated from the specific volumes listed in Table 11-15.

C/T:	81 pounds x 0.021	1.7
Reg:	75 pounds x 0.028	2.1
Inv:	280 pounds x 0.0181	5.1
	Added Volume	8.9 cu. ft.

TABLE II-15. BATTERY AND ELECTRONICS MODULE WEIGHT AND VOLUME

Item	Component Weight (pounds)	Specific Volume (cu. ft. per pound)
Battery Module		
• (Ni-Cd	66	-
• (Ag-Cd)	55	-
Charger/Tracker		
o Control	16	0.021
• Power	25	
Regulator		
• Control	15	0.028
o Power	30	
Inverter	70	0.0181

The total weight and volume of the system electronics is then:

	Weight	Volume
Baseline	489	11.0
Added	446	8.9
Total	935 pounds	19.9 cu. ft.

(6) Battery Weight and Volume

The enhanced system battery weight and volume are calculated by a straightforward ratio method. Five batteries with six modules per battery is 30 battery modules for the baseline system. The three additional spare battery modules will provide an increase of 10 percent in the weight and volume (compared to the computer print-out values).

Therefore, the total weight and volume of the system battery are:

	Weight	Volume
Baseline	2761	60.2
Added	276	6.0
TOTAL	3037 pounds	66.2 cu. ft.

(7) Solar-Array Weight

The Solar-Array specific weight (pounds per square foot) programmed into the computer is 0.744; this includes only the basic array, and must be corrected to the actual value of the design (see Table II-16). (A simpler approach would have been to change the stored value in the computer before the run was made.) Then, for 1838 square feet of array, the new weight is (1838 x 0.823) or 1513 pounds. To this value must be added the weights of the boom and deployment mechanism and the orientation servo system. Using Table 11-16, the total array weight becomes:

Item	Weight
Basic Array	1513
Boom and Deployment	675
Orientation	105
TOTAL	2293 pounds

TABLE II-16. IPS WEIGHT SUMMARY

Item	Weight	
	lb per sq. ft.	lb
Solar Array (4900 sq ft)		
Solar Cells & Electrical Components	0.314	
Aluminum Structural Frame	0.144	
Aluminum Honeycomb - 1/4 in. thick	0,235	
Hinges	0.034	
Panel Fold-Out Actuators & Fittings	0.019	
Electrical Harness	0.050	
Miscellaneous Hardware	0.027	
Total	0.823	4033
Deployment Mechanisms		
Boom Assembly (2 req.)		380
Transition Panel (2 req.)		180
Drive Shaft & Bearings (2 req.)		50
Boom Actuator Assembly (2 req.)		10
Array Retaining Cables & Hardware (2 req.)		45
Electrical Connectors		10
Total		675
Servo Control System		105
Array Orientation Drive Assembly (2 req.)		
Electronics		
Sensors		

(8) Fuel-Penalty Calculation

Since the array area is 1838 square feet in a 'Z' configuration (IPS concept), the fuel penalty involved may be determined by reference to Table II-9. Two cases must be considered; with and without a momentum storage device (MSD). The total weight can be determined by summation of the values from the table.

Item	Weight Without MSD	Weight With MSD
Aerodynamic	1574	1260
Gravity Gradient	2310	910
Solar Pressure	25	10
Magnetic Dipole	55	11
MSD Weight (See Table II-11)	0	35
TOTAL	3964 pounds	2226 pounds

It must be recognized that the weight penalties involved are calculated for a PSM 'Z' orbiting configuration and thus are not necessarily true for the IPS 'Z' configuration due to the differences in the RCS moment arms and vehicle center of mass. However, for these purposes, it is felt that the error introduced will not significantly affect the total power system weight.

(9) Total Power System Weight

The total power system weight can now be determined.

Item	Weight
Battery weight total	3037
Electronics weight total	935
Solar-array weight total	2293
Fuel-penalty total	2226
TOTAL	8491 pounds

This then is the total system weight attributed to the power system for an IPS concept.

3. PSM Solution

The methods described in the preceding paragraph are also applicable to solving the problem for the PSM concept, with only two major differences. The solar array will be an 'H' configuration and will have a weight breakdown as shown in Table II-17. Also, the batteries will be packaged differently since they are no longer within the manned vehicle.

Battery weight for the PSM concept is calculated on the basis of the number of modules to be utilized. Since there are to be 33 modules at 66 pounds per module (from Table II-15), the total battery module weight will be 2178 pounds. The weight of the battery mounting hardware is included in the weight of the component housing assembly.

The solar array weight is calculated in the same manner that the IPS array weight was calculated except that values from Table II-17 are used. The total array weight will then be 1649 pounds.

Again referring to Table II-10 for calculating fuel penalty weights, the "H" configuration values are used to arrive at 1967 pounds without MSD and 1298 pounds with MSD.

The total PSM power system weight will then be:

Item	Weight
Battery weight total	2178
Electronics weight total	935
Solar-array weight total	1649
Fuel-penalty total	1298
Component housing/Thermal-control system	700
TOTAL	6760 pounds

Several PSM items are not included in this weight because estimates for these elements can be made only if realistic assumptions can be made for the docking tunnel and the PSM/manned vehicle mechanical interface. First, the weight of the docking tunnel (airlock) and its associated hardware has not been included. Second, the basic PSM structure weight has not been included.

TABLE It-17. PSM WEIGHT ESTIMATE

Item	Weight	
	Lb/sq ft	Lb
Solar Array, 1927 sq ft		
Solar cells and components	0.314	
Aluminum structural frame	0.178	
Aluminum honeycomb, 1/4 in. thick	0.235	
Hinges, locks, and array deployment mechanism	0.053	
Electrical harness	0.008	
Miscellaneous hardware	0.020	
Total	0.808	1558
Deployment Mechanism		
Boom assembly (2 required)		65
Boom actuator assembly (2 required)		8
Array retaining cable		25
Electrical connectors		6
Total		104
Thermal Control System		700
Servo Control System		60
Array orientation drive assembly		
Servo electronics		
Sensors		

SECTION III

CONCLUSIONS

SECTION III

CONCLUSIONS

Specific conclusions for each of the trade-off studies and analyses performed are presented in the appropriate sections of the technical discussion (Sections VI through X). In addition, a series of general system-related conclusions have become apparent.

(1) Solar-array power systems for multikilowatt manned orbital applications are practical within the present state of the art. No technological breakthroughs are required to implement a program that will lead to flight-qualified equipment for a 1970-to-1972 launch.

(2) The Power System Module (PSM) concept requires significantly less interfaces between the power system and the manned vehicle than the Integrated Power System (IPS) concept. However, only a direct comparison of a PSM with an IPS (in terms of complexity, cost, reliability, development time, and weight) can lead to the logical selection of the optimum system for a specific application.

(3) The dynamic interaction between the solar array and the orbiting vehicle requires an in-depth analysis before development of a multikilowatt solar-array power system is undertaken. Factors affecting this relationship are the solar-array mechanical characteristics, the orientation-drive system characteristics, the solar-array-to-vehicle mechanical interface, the vehicle mechanical characteristics, and the vehicle attitude-control requirements and characteristics.

Although there are other areas with unresolved questions, these are considered design challenges rather than major technical problems. Typical of these areas are the low-temperature capability of the solar-array, testing of array deployment in a 1g field, determination and evaluation of the characteristics of large-capacity batteries, and the implementation of the solar-array orientation-drive and power-transfer system.

(4) The major non-technical problem is the evaluation of the development-plan alternatives as a function of flight-unit availability, material-procurement risks, and component-manufacturer's capability. Two areas, in particular, are important because of the large quantities required (solar cells) or because of the design and production time required (batteries).

(5) The weight estimates for the PSM power system as a function of mission parameters have not been developed adequately to permit realistic comparison of an IPS and a PSM. The PSM functional and operational requirements are not adequately defined to permit stress analysis and structural design for the docking tunnel (airlock), the component housing assembly, and the basic PSM structure.

(6) The conceptual design for a 5-kW, 200-n. mi mission (as described in Section II) does not require any major mission or manned-vehicle constraints.

(7) Since the batteries are the largest contributor to total power-system weight for the range of missions studied, effort expended to reduce battery weight may yield a greater return in reducing the total power system weight than similar effort expended in the area of solar array or control electronics.

(8) The 39-month development plan (including one flight-qualified system) defined in Section V can be implemented for approximately 35 million dollars.

(9) Specific areas of further detail study can be undertaken prior to implementation of an equipment program. These studies may reduce the system design and development time; however, no detail "critical path" evaluation has been made. (See Section IV).

SECTION IV

RECOMMENDATIONS

SECTION IV

RECOMMENDATIONS

A. INTRODUCTION

Section V presents a development plan, schedule, and budgetary cost estimate which define the magnitude of effort required to provide a flight-qualified PSM power system in 39 months. Within that description are many problems of analysis, design, and component development which may be addressed independently and which can contribute to the progress of the PSM design. In addition to these specifically PSM-oriented problems, there are a number of tasks (necessary to the PSM development) which are oriented to the more general problem of the orbital performance of a large-area-array/orbiting vehicle configuration.

This section describes not only the RCA recommendations for a continuing effort on the PSM design, but also the effort applicable to the broader problem of a high-capacity, photovoltaic earth-orbiting power system.

B. PSM DESIGN EFFORT

1. PSM Structure and Thermal Control

The PSM concept developed during this conceptual study was defined to have minimum interface (i.e., minimum constraints) with the orbiting vehicle utilizing the power system. Specific and more detailed design requires further definition of the interfaces between PSM and orbiting vehicle. RCA recommends an iterative type of design program wherein initial interface definitions and assumptions are made, followed by a detail design of the PSM structure, docking-tunnel (airlock) interface and thermal control system. As the design proceeds and interface definitions are received to suit practical design limitations, the PSM configuration and its range of applications will become increasingly identified.

2. Computer Program Enhancement

The present computer program will provide component sizing (for each solution) and a total system weight (for a baseline system). In order to make this program a more useful design tool, the following modifications are recommended:

(a) Provide the necessary logic changes to permit (as an input option) computer solutions based on predetermined fixed-size components.

(b) Provide the additional storage and logic capability to incorporate reliability enhancement for each of the functional elements.

(c) Provide a subroutine which will modify the array power for the shadowing encountered during orbit.

3. Vehicle-Attitude Effect

For the present study, the vehicle was defined to have an attitude in which roll axis was coincident with the velocity vector and the yaw axis coincident with the local vertical. It is recommended that the effects of other vehicle attitudes be evaluated with respect to the thermal control system, array-orientation-axis selection, array-power output penalty, and attitude-control fuel penalty.

4. Applicability of Other Array Designs

A current NASA program is in process (at the Boeing Company) to develop a large-area lightweight solar array suitable for interplanetary-flight applications.

RCA recommends that this array design be evaluated for application to the orbital missions of this program. Utilization of this advanced array design could possibly shorten the time cycle required to provide a flight-qualified PSM.

5. Electrical System Breadboard

It is recommended that a parallel-tracker system configuration be assembled with a solar-array simulator and lead-acid batteries to demonstrate system performance. The results of this effort would facilitate equipment design of the PSM electronic components.

6. Solar-Array Development Model

It is recommended that an engineering model of the solar array be designed and tested to demonstrate deployment and retraction capability.

C. SOLAR-ARRAY/BATTERY SYSTEM DESIGN EFFORT

1. Analytical Problems

a. ARRAY/VEHICLE DYNAMICS ANALYSIS

A major technical problem described in this report is the determination of the dynamic interaction between the solar array and orbiting vehicle. It is recommended that a program be undertaken to define a mathematical model of the dynamic system and that a computer program be constructed for solution of the problem. The solution of the dynamic-interaction problem will yield the amount of fuel required in orbit to maintain attitude control as a function of the dynamic parameters of the orbiting vehicle solar array. In addition, array transient vibrations and reactions torques on the vehicle are obtained.

b. PERTURBATION ANALYSIS

Perturbations refer to the "steady-state" reactions on the array, such as solar pressure, magnetic dipole, aerodynamic drag, and gravity-gradient forces. During the course of this study, several iterations of the perturbation analysis were made.

Development of computer programs to perform these calculations as a function of orbit, array geometry and size is recommended so that fuel-penalty and MSD determination can be made rapidly for a variety of missions and orbits. This computer program would be used in conjunction with the Dynamic Analysis program to ascertain the complete effect of the array on the orbiting vehicle.

2. Equipment Problems

Investigations are recommended which are concerned with the various components utilized in a high-power photovoltaic power supply.

a. PLUME EFFECTS

It is recommended that an investigation of the plume effects of the RCS thruster and the SPS engine on sample solar-array panels be performed. Both thermal and particle damage should be investigated.

b. ARRAY DRIVE SHAFT-SLIP-RING DESIGN

The complex mechanism of the array drive shaft including boom-deployment mechanism, motor drive, slip rings, bearings and latches will require extensive development testing. Although rotating shafts and slip rings have been used in space (on the Nimbus satellite), testing will be required for the PSM application. A design and test program is recommended to provide materials and design information.

c. ARRAY LOW-TEMPERATURE QUALIFICATION

The low-temperature capability of the solar-array has a direct effect on system performance and efficiency and on array weight. Design and test of low-temperature (-200°C) solar-cell interconnection configurations is recommended.

d. BATTERY CHARACTERISTICS

Large-capacity, sealed nickel-cadmium and silver-cadmium storage battery characteristics have not been determined emperically with sufficient accuracy to permit a detailed system design to be undertaken. A program for determination of 100-to-150 ampere-hour cell configurations and characteristics is recommended.

D. SUMMARY

Recommended sequels to the present program can be categorized into three types of efforts. -

- PSM design problems
- Solar-Array system design-analytical problems
- Solar-Array system design-equipment problems

During the course of an overall program leading to a flight-qualified PSM, all of the problems described in each of these three categories will require resolution. However, if the PSM is not yet ready to be committed to design and equipment, fruitful and applicable results can be obtained by performance of those programs recommended under "solar-array design problems".

SECTION V

PROGRAM DEVELOPMENT PLAN

SECTION V

PROGRAM DEVELOPMENT PLAN

A. INTRODUCTION

Based on the results of a nine-month conceptual study, a program development plan has been defined for providing a space-qualified Photovoltaic Power System for a Manned Orbiting Vehicle. This development plan describes a 39-month, four-phase program for the design, test, and fabrication of the Power System Module (PSM) concept employing present state-of-the-art techniques.

The Power System Module (PSM) is a separate, self-contained unit (i.e. the power system components are not an integral part of the mission vehicle) capable of supplying power for a variety of vehicle configurations and mission requirements. The Power System Module comprises seven major equipment groups:

(1) Solar Array Sections (4 per system) – includes the solar-panel assemblies, hinge and latch mechanisms, and panel-storage hardware.

(2) Deployment and Orientation Assemblies (2 per system) – includes the boom, the 2-degree array orientation drive motors and gears, the orientation sensors, and the electrical slip-ring assemblies.

(3) Orientation Servo Assemblies (2 per system) – includes the error sensors and power amplifiers that power the orientation hardware.

(4) Power System Electronics – includes all regulation, conversion, inversion, power-management, and status-monitoring components.

(5) Storage Battery – includes all storage-cell assemblies.

(6) Electrical Component Housing (2 per system) – includes structural units to house the Orientation Servo Subsystem, the Power Control Electronics, the Storage Battery Subsystem with a thermal control system for each structural unit, and appropriate harness and junction-box provisions to interconnect the power-system components.

(7) Power System Structure – consists of the main structural frame that will structurally integrate the Module Housing units, the solar-array subsystem, and the Deployment and Orientation Subsystem into the Power System Module (PSM).

Figure V-1 shows the interface that relate each of the seven PSM equipment groups to each other and to the mission vehicle. Specific definition of the detailed interfaces between the PSM and mission vehicle will be an important factor in the performance of the described development program. Although many portions of the development cycle (i.e. s component

development such as breadboards, mechanisms, etc.) may proceed with only preliminary interface information, the initiation of mechanical designs will require that all interface information be complete and firm if a space-qualified power system is to be provided within the defined schedule.

B. PROGRAM SCHEDULE

Figure V-2 is a master schedule chart summarizing the major milestones in the development program. Four major program phases have been defined; these are:

- Phase I, Program Definition and Preliminary Design: This phase extends for nine months and will concentrate on detailed preliminary design of each of the seven equipment groups. Completion of the preliminary design will require timely definition of the external system interfaces by the fifth month (see Figure V-2).

- Phase II, Engineering Model Development: This phase proceeds from the completion of Phase I and will concentrate on the fabrication and testing of engineering model assemblies. Fabrication releases for a complete hardware complement will be initiated during the first month of this phase. Starting with individual component tests, a completely integrated engineering model of the PSM will be assembled and tested. Figure V-2 indicates the flow of hardware buildup. Prior to incorporating live components in the component housing, the unit will be assembled with mass

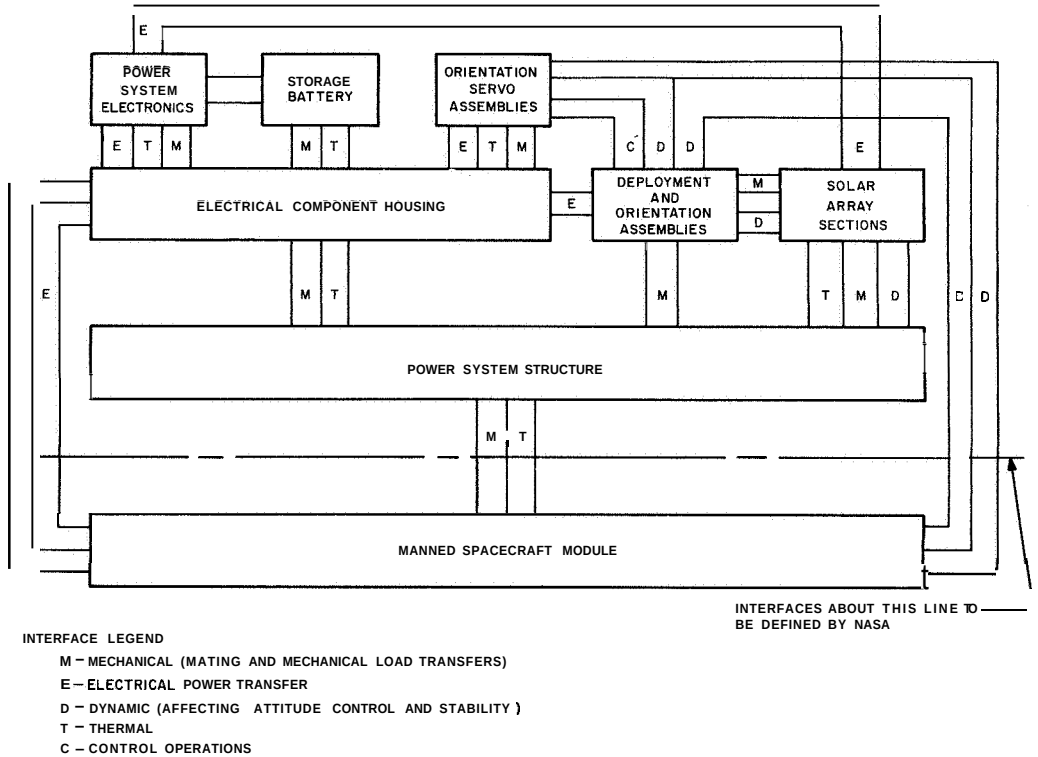
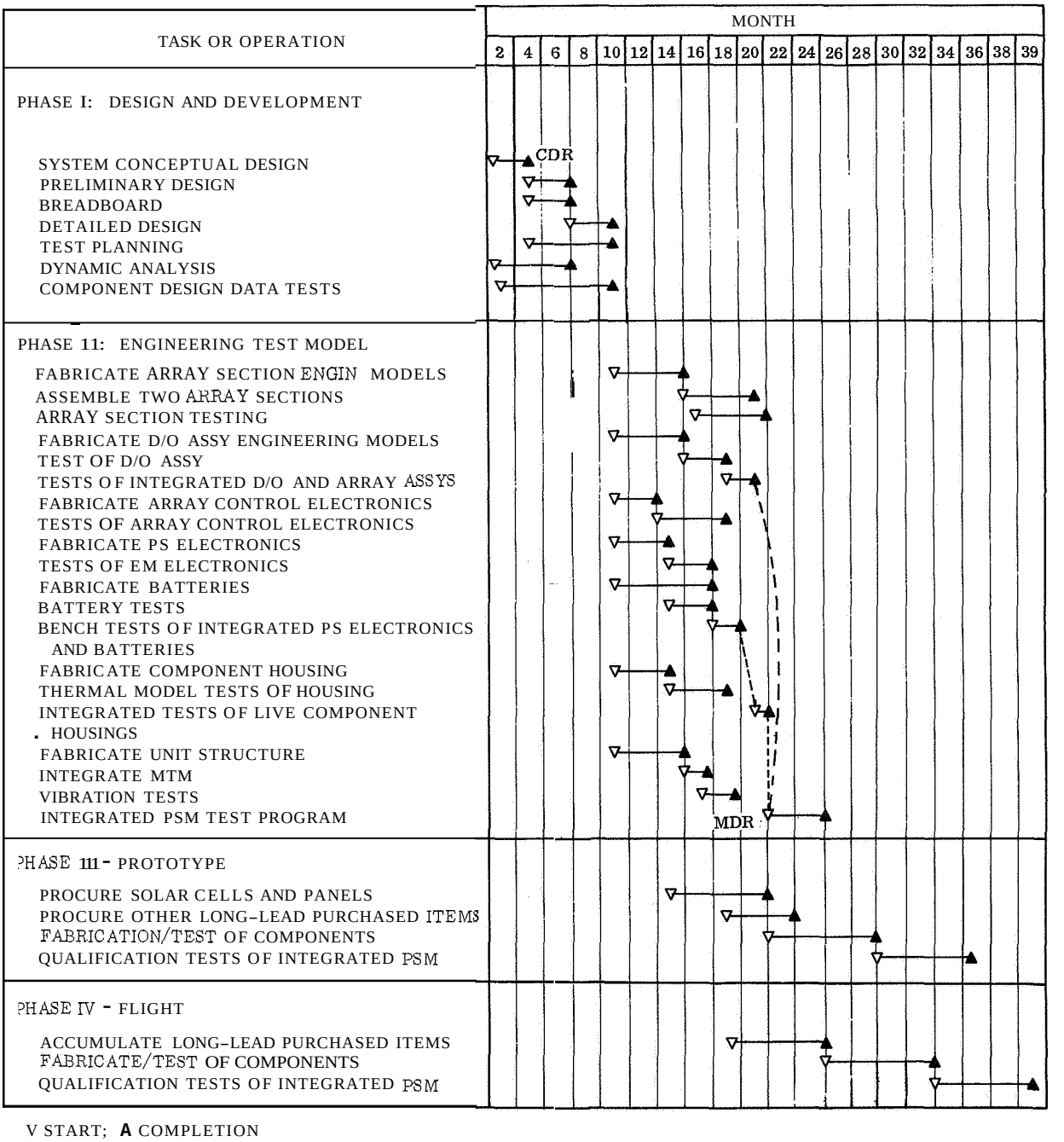


Figure V-1. Equipment Interfaces



V START; A COMPLETION

CDR CONCEPTUAL DESIGN REVIEW
MDR: MAJOR DESIGN REVIEW

Figure V-2. Development Program Master Schedule

dummies and tested for vibration responses and to design force levels. The unit will then be assembled with thermally heated dummies and operated in vacuum as a thermal test model. The Power System structure will likewise be tested in vibration with mass dummies, and later will be integrated with a deployment and orientation assembly and with an array section for further vibration tests. Major testing will be completed at the end of the 20th month and a major design review will be scheduled. Tests of the integrated PSM willcontinue to the 24th month in order to gain additional testing experience, particularly in areas such as RFI*, failure mode operations, etc.

During this phase, a complete complement of handling and support equipment will also be fabricated for use with the engineering models. Separate test articles will be fabricated and tested as follows:

(1) Scaled deployment model – tested for repetitive operations in thermal vacuum; consists of full-scale panel mechanisms with foreshortened array sections to permit deployment in restricted spaces.

(2) Slip-ring assembly – continuous testing for life data.

(3) Storage Cells – continuous cycling for life data.

(4) Servo-drive components – continuous testing for life data.

(5) Thermal controller – continuous testing for life data.

- Phase III, Prototype Fabrication – In the master schedule (Figure V-2), some prototype part procurement is shown starting four months after the engineering model phase begins (in month 13 of the overall program) to permit component deliveries of the large-volume items (such as solar cells, storage cells, panel substrates, etc.) to be accomplished in a timely manner. Other long-lead items of a more special nature can be ordered as late as the 18th month, thus permitting prototype hardware fabrication and assembly to begin in the 21st month after the major design review. A rather extensive manufacturing and test cycle is envisaged, with prototype qualification attaining 'completion in the 35th month.

- Phase IV, First Flight Model – As with the prototype/ETM** overlap, some first-flight-model long lead items can be procured in parallel with portions of the prototype cycle, starting in the 17th month. Fabrication and assembly is shown beginning at the completion of the prototype assembly and all ETM testing. The flight acceptance test program would start four months after initiation of the prototype test program and reach completion in the 39th month.

*Radio-frequency interference
**Engineering test model.

In review, initiation of the first phase of the program early in **1967** would provide the first flight system for spacecraft integration early in **1970** (assuming program continuity and phasing overlaps as indicated).

Many variations of the basic schedule may be made by altering the overlaps between the major system hardware phases and by extending the overall program length at the option of "hard" program requirements. However, RCA feels that the schedule, as shown, realistically demonstrates the minimum time in which qualified flight hardware can be made available for vehicle integration.

It is appreciated that other hardware components will be required for function and fit-check operations. These are not programmed in the schedule since they are not pertinent to the main stream of the hardware development. If significant numbers of components are required for these needs, they may have some impact on the overall schedule. That impact, however, cannot be identified at this time.

C. ENGINEERING PLAN DETAIL

The four phases of the master schedule define the engineering activities to be performed during the program. The primary engineering activities to be performed during each phase of the development program are outlined in the following paragraphs.

1. Program Definition and Preliminary Design (Phase I)

The output from this phase will be preliminary design information that will provide the basis for fabricating engineering test models (ETM) in Phase II for the engineering model development program. System design requirements must be established during the initial months to permit definition of preliminary component specifications for initiating design activities for each equipment group. Initial effort will be directed toward the development of system block diagrams, component block diagrams, layout studies, and the gathering of component part data. Work will be initiated to set up the problem of analyzing the dynamic coupling and interactions between the vehicle and the solar array. The analysis will consider both the array perturbation and control systems interplay during deployment and during normal sun tracking, in addition to vehicle attitude impulse corrections. A concept review of the system design will be scheduled at the end of the third month. The remainder of the engineering task in Phase I will be design and analysis activities, supplemented by breadboarding of circuits, and minor component development activities relating particularly to the solar-cell module design, storage-cell parameters, orientation-system components such as motors, gearing, and slip rings, etc. Final mechanical and dynamic interface data, to permit initiation of meaningful detailed mechanical design, will be required by the third month. All other interfaces will require firm definition by the fifth month.

2. Engineering Model Development (Phase II)

The engineering activities during Phase II will concentrate on developing the final equipment designs leading to the fabrication of prototype and flight hardware. The equipment designs generated during Phase I will be released for first-piece fabrication. Thermal, mechanical, and functional performance as well as the ferreting out of manufacturing and assembly problems are the principal aims of this phase. Design changes will be generated in areas indicated by the engineering model test program. A major design review is scheduled for the 20th month, following completion of all major mechanical tests. Further testing of the integrated system will continue to the 24th month to accumulate system experience.

3. Prototype Fabrication (Phase III)

Prototype equipment (considered to be of flight-hardware quality) is to be tested to levels higher than flight levels in order to demonstrate design integrity. The program emphasis is thus shifted from design and development to production and design verification. The engineering activity during this phase will be directed to guidance and support of fabrication and test cycles.

4. Flight Equipment Fabrication (Phase IV)

As in the prototype fabrication phase, engineering activities during Phase IV will be directed toward support of the fabrication and test activities of the flight equipment.

D. TESTPLAN

. The testing program, encompassing both major and minor test articles, extending from ETM through the flight hardware, is based on the philosophy of simulating all launch and orbit operations to the utmost extent practical. Table V-1 summarizes the operations and simulations that have been considered.

1. Phase I Test Plan

During Phase I, the emphasis in the test plan is heavily in the direction of establishing design criteria and verifying design performance as in breadboard testing. Table V-2 summarizes the major elements of testing proposed during Phase I. Figure V-3 summarizes the Phase I planning.

TABLE V-1. TEST SIMULATIONS

Operation	Possible Simulated Environment
Launch	• Launch environment • Pre-launch, temperature vibration, shock, and humidity
Parking Orbit	o Low-temperature stabilization prior to system activation
System Activation	• Array deployment • Sun acquisition and tracking • Load turn-on sequence • Vehicle dynamic interface during acquisition
System Operation	• Load profile cycling, temperature extremes • Array temperature cycling, worst-case orbits • Battery cycling, efficiency, and life o RFI and system transients • Power System Control operations • Array sun tracking, accuracy, life, and vehicle dynamic interface
Special Operations	• Deactivation • Array retraction • Modified Activation • Docking • Vehicle maneuvering

TABLE V-2. PHASE I DESIGN TESTING

Item	Test
Solar-Cell Module Solar-Cell Module	• Thermal cycling of proposed solar-cell assembly designs to verify mechanical integrity
Storage Cell	• Charge-discharge cycling to accumulate parametric design data for charge-control circuits
Regulation and Power Control Circuitry	• Breadboard testing to measure and verify design parameters such as logic, gain, impedance, stability
Array Control System	• Electromechanical breadboard tests to measure transfer-function, stability, and accuracy parameters
Array-Deployment Equipment	• Tests on proposed component parts to accumulate design data

2. Phase II Test Plan

During Phase II, an extensive series of tests will be performed, beginning with low-level component assemblies and ending with tests on a completely integrated engineering model.

Figure V-4 is a flow plan for the fabrication and test activities in Phase II. Releases for all major pieces of engineering-model hardware will be made at the beginning of the phase to the drawings produced during Phase I.

a. SOUR-ARRAY SECTIONS

The engineering test program will verify the deployment and retraction capabilities of the array design for the various design conditions. The PSM design is configured as four stowed assemblies with the array sections deployed in pairs on either side of the spacecraft. Each of these assemblies includes 10 panels folded for stowage. The needs of the engineering test program will be fulfilled by two stowed assemblies (consisting of 20 panels) with the other two stowed assemblies represented by models of equivalent compliance for testing the

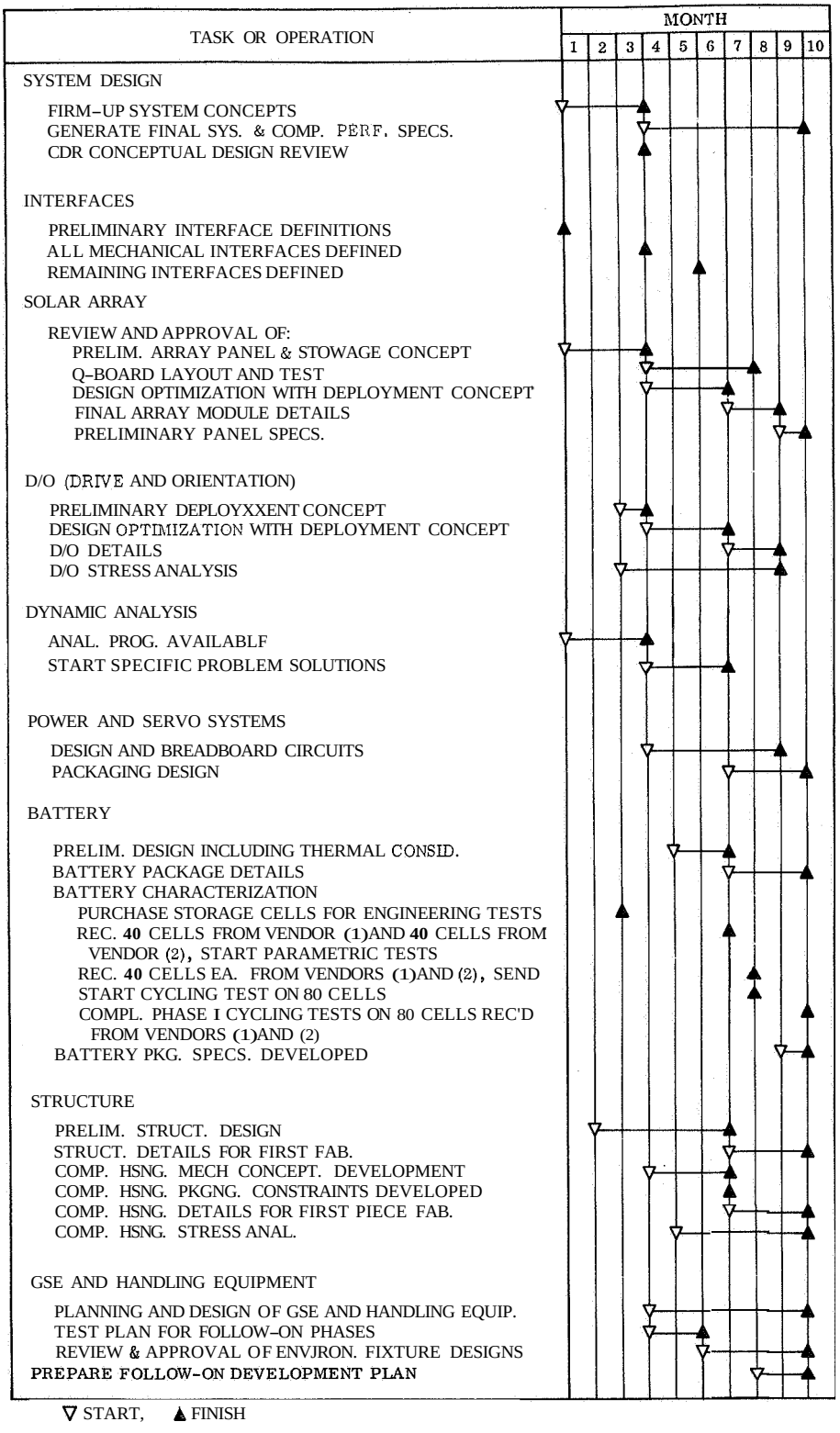


Figure V-3. Phase I Plan

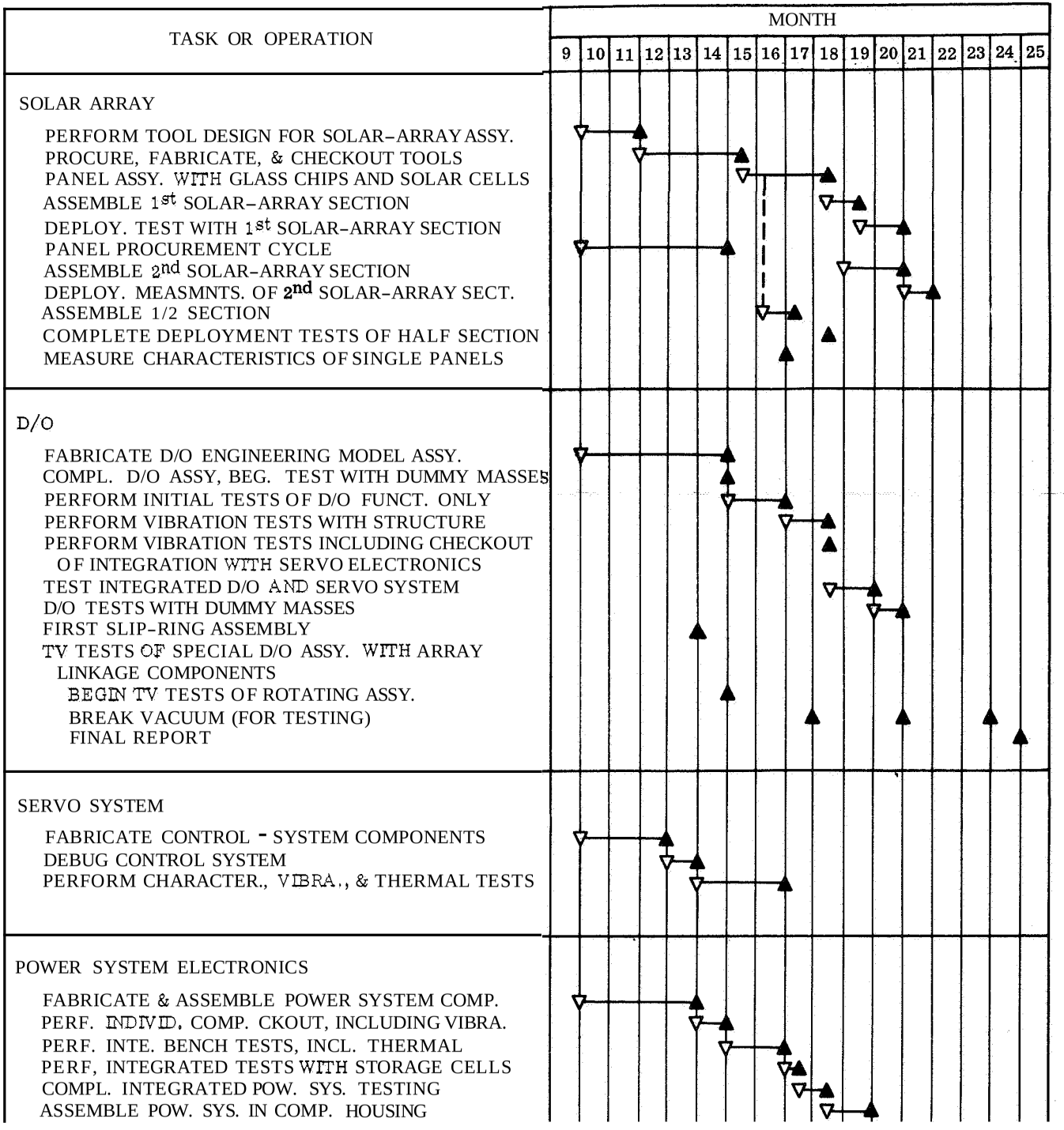
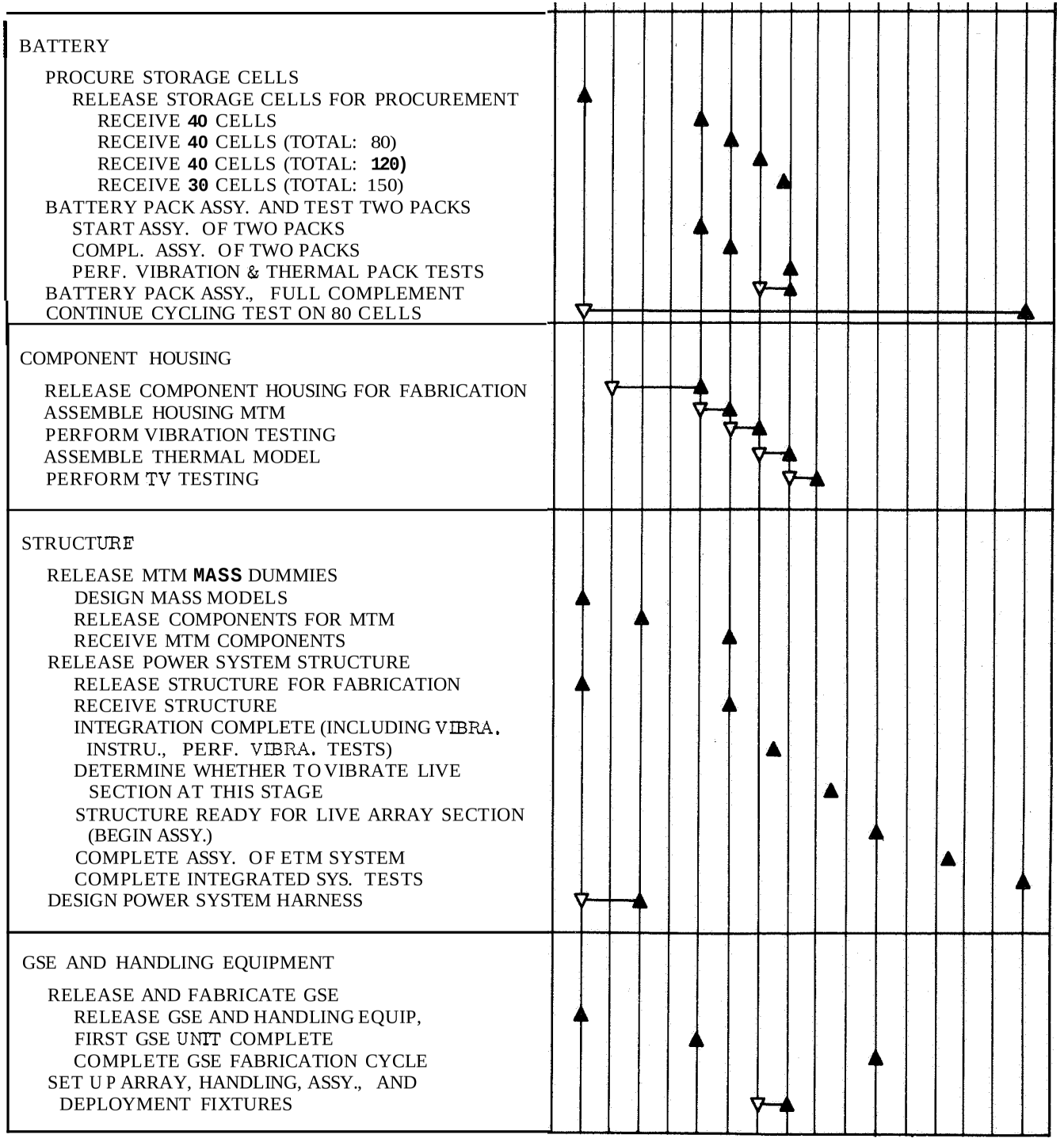


Figure V-4. Phase II Plan


BATTERY
PROCURE STORAGE CELLS
RELEASE STORAGE CELLS FOR PROCUREMENT
RECEIVE 40 CELLS
RECEIVE 40 CELLS (TOTAL: 80)
RECEIVE 40 CELLS (TOTAL: 120)
RECEIVE 30 CELLS (TOTAL: 150)
BATTERY PACK ASSY. AND TEST TWO PACKS
START ASSY. OF TWO PACKS
COMPL. ASSY. OF TWO PACKS
PERF. VIBRATION & THERMAL PACK TESTS
BATTERY PACK ASSY., FULL COMPLEMENT
CONTINUE CYCLING TEST ON 80 CELLS
COMPONENT HOUSING
RELEASE COMPONENT HOUSING FOR FABRICATION
ASSEMBLE HOUSING MTM
PERFORM VIBRATION TESTING
ASSEMBLE THERMAL MODEL
PERFORM TV TESTING
STRUCTURE
RELEASE MTM MASS DUMMIES
DESIGN MASS MODELS
RELEASE COMPONENTS FOR MTM
RECEIVE MTM COMPONENTS
RELEASE POWER SYSTEM STRUCTURE
RELEASE STRUCTURE FOR FABRICATION
RECEIVE STRUCTURE
INTEGRATION COMPLETE (INCLUDING VIBRA., INSTRU., PERF. VIBRA. TESTS)
DETERMINE WHETHER TO VIBRATE LIVE SECTION AT THIS STAGE
STRUCTURE READY FOR LIVE ARRAY SECTION (BEGIN ASSY.)
COMPLETE ASSY. OF ETM SYSTEM
COMPLETE INTEGRATED SYS. TESTS
DESIGN POWER SYSTEM HARNESS
GSE AND HANDLING EQUIPMENT
RELEASE AND FABRICATE GSE
RELEASE GSE AND HANDLING EQUIP,
FIRST GSE UNIT COMPLETE
COMPLETE GSE FABRICATION CYCLE
SET UP ARRAY, HANDLING, ASSY., AND DEPLOYMENT FIXTURES

orientation system. Panels and folding and stowage hardware will be fabricated for the engineering model array according to the required design. Since the emphasis in the model is on mechanical functioning, glass chips (of mass equivalent to the solar cells) will be bonded to all except one of the panels, to which active solar-cell modules will be bonded. (Demonstration of the mechanical integrity of the solar-cell assembly in temperature cycling will be performed in a small-scale test during Phase I.) This expedient permits checkout of array fabrication tooling and accumulates all of the required mechanical performance data on the solar panel assembly at minimum cost. Upon receipt of the panels, the glass chips and solar cells will be bonded and deployment hardware added to each panel. Electrical buses or cabling will be included to make a true representation of the mass and to test for potential interferences during deployment. The first panel to be bonded will be used in a measurements program to determine balance, center of gravity, panel deflection, and other characteristics. The third panel will be bonded with the 'live" solar cells and will be similarly measured. In addition, it will be measured for magnetic properties. Using the first five glassed panels, a partial array section will be assembled for first-generation deployment tests. When the remaining complement of panels (four more glassed panels plus the live panel) are available, the array section will be fully assembled and further deployment testing will be pursued. When the deployment and retraction operations are considered satisfactory, the array module will be vibration-tested in the PSM structure and deployed and unfolded to verify the design. A second section will be similarly assembled and tested for deployment performance.

This assembly will then be subjected to various measurements (for compliance, balance, etc.) and will be unfolded and retracted through many operations. The measurements will be useful to verify preliminary data used for the orientation system design. Since it is not practical to deploy the system in a vacuum chamber, a separate model will be fabricated (of perhaps a half-dozen panel-folding assemblies) and tested for repeated operation in vacuum after various thermal-vacuum conditioning programs, for example, low-temperature "soaking" to simulate orbital parking before deployment.

b. DEPLOYMENT AND ORIENTATION SUBSYSTEM

This assembly serves as the tie-down and positioning link between the array sections and the PSM. Of major importance in the engineering test program, will be the performance of the various linkages and rotating components. Although there will be important structural considerations, the bulk of engineering testing will be concerned with operations in vacuum over the predicted temperature regime. Wear-out phenomena are also of prime importance and separate test articles of motors, bearings, linkages, and slip rings will be placed on continuous test in vacuum early in the program.

One engineering-model unit will be assembled and tested thoroughly under thermal-vacuum conditions. As indicated previously, this assembly will be subjected to vibration tests in the power unit structure and subsequently mated to the array for further vibration and deployment tests.

c. ORIENTATION SERVO SUBSYSTEM

This electrical servo assembly receives signals from the array positioning sensors and, through an electromechanical linkage, positions the array to optimize the solar input. Initial engineering tests will be of the servo loop alone in the packaged configuration. Tracking-error inputs and electromechanical compliance will be simulated for these tests. The unit will be tested over its full temperature range (non-vacuum conditions). After these tests, the assembly will be mated to the deployment and orientation assembly, together with array compliance masses, and additional loop tests will be conducted with simulated tracking-error inputs. The servo will subsequently be integrated into the component housing and tested in thermal vacuum prior to integration into the PSM structure for the final engineering-model system test series.

d. POWER SYSTEM ELECTRONICS

Each major component regulation, charge-control, and power-management unit will be tested by itself to its component specifications. Each unit will be vibration-tested and tested in a temperature (non-vacuum) chamber for evaluation of the detailed performance characteristics. At the completion of these tests, the components will be assembled in an appropriate buildup sequence (including mating with batteries) to form a complete power system. The array input and load programming will be simulated by ground test equipment. After a program of bench testing, the entire system will be assembled into the component housing and tested in vacuum over temperature extremes and load profiles for an appropriate time period (approximately seven days). Finally, the power system will be integrated into the PSM for the engineering-model system tests.

e. STORAGE BATTERY SUBSYSTEM

Although it is possible to obtain adequate engineering information from one or two battery packs, a complete system complement of batteries is proposed in order to provide a better systems-test program in areas such as load-sharing between batteries, etc. One pack will be subjected to a complete test program, including vibration and thermal vacuum, with particular attention given to measuring temperature gradients and hot spots within the battery pack at the various charge rates. Baseplate temperatures over the complete temperature range will be simulated. (Cycling performance will be initiated during Phase I and will be continued during Phase II.) After individual pack tests, the batteries will be integrated with the power-system electronics for further tests as indicated in Pharagraph V. B. 2. d.

f. ELECTRICAL COMPONENT HOUSING

Maximum system flexibility will be obtained for the PSM by approaching the mechanical design of the electrical system in a truly modular fashion; i. e. to provide a flexible, plug-in type of assembly. This approach is implemented by designing each of the component packages of the electrical system to plug in to a universal component housing, which includes mechanical features to permit simple assembly of electrical components to the component

housing, built-in harnessing to interconnect the various component packages, input-output junction boxes for the spacecraft interface, and with built-in thermal control. The approach can be though of as a bookcase, with electrical component as the equivalent of a book. The intent is to make the component housing, or bookcase, as universal as possible, permitting a wide variety of power-level requirements to be met by choosing an appropriate complement of books as needed.

A systems complement of component housing (two units) will be released for fabrication at the beginning of Phase II. The first unit received from fabrication will be assembled with mass equivalent mechanical dummies, and the integrated component housing will be subjected to vibration testing. The unit will be thoroughly instrumented to permit detailed mapping of the structural responses and transmissibilities. When the analysis of structural performance is complete, the measured responses will be incorporated into the test requirements for the electrical components. The component housing will then be reassembled with thermal dummies to simulate the thermal loads to be handled by the component housing thermal control. The unit, after checkout, will be tested in thermal vacuum to evaluate steady-state and time-dependent temperature profiles.

The dummy components will be removed from the component housing, and the 'live" engineering-model electrical components that have completed their testing will be assembled into the housing for the thermal-vacuum test program. The tested housings will then be integrated into the power unit to conduct the test program on the complete engineering model.

g. POWER UNIT STRUCTURE

The power unit structure is basically a structural frame that facilitates mechanical integration of the component housings, the array sections, and the deployment and orientation assemblies as a total power system, and provides the structural tie to the vehicle. An important interface will be that to the vehicle airlock (or docking tunnel), around which the power system components are to be assembled. It has been assumed that the airlock will be a GFE* item and that timely detail definition will be available, as well as suitable models.

A full-scle engineering-model structure will be fabricated for testing. Mass equivalents for the component housings, array sections, and deployment and orientation assemblies will be integrated to the structure and the entire structure will be tested for its response and strength characteristics. After completion of these tests, the structure will be available for further testing with the live engineering-model components, as previously described.

h. SPECIAL TESTS

A number of special tests (as follow) should be performed in addition to the tests outlined above.

*Government-furnished equipment.

(1) Acoustic Noise. Acoustic noise is principally effective against large area components, such as the solar panels, large structural webs, and flat plane radiators or radiating surfaces. During Phase I, considerable thought will be directed to planning the test articles and test methods to be used for acoustic-noise tests.

(2) Dynamic Interaction Testing. It is appreciated that the spacecraft is not a rigid, inelastic base for obsorbing the dynamic characteristics of the deployed array. Rather, it has a feedback reaction of its own, which will interact with both the structural and control system dynamics of the PSM. During Phase I, an important analytic effort will be directed to understanding these interactions and to account for them in the design. However, it seems quite important to perform some form of dynamic-simulation tests with a spacecraft simulator to establish steady-state loads, coupling factors, array reactions to impulses, etc. Ultimately, it would be prudent to provide a means for giving the astronauts some direct feel for spacecraft flight and control characteristics, with the large-area array. Again, a carefully thought out plan of testing will be generated during Phase I. For the purposes of this program, it is assumed that flight simulators will be available in the overall spacecraft plan for a test program with the engineering model PSM. If this proves to be a false assumption, it is recommended that a spacecraft mock-up, with the appropriate interaction parameters, be fabricated for test purposes.

3. Phase III Test Plan

The prototype hardware is defined as the first hardware built to flight-quality standards. It is to be subjected to prototype level testing to verify the design. Figure V-5 is a schedule showing the fabrication and test activities for the prototype (PTM) system. Components, major subassemblies, and the entire system assembly will be subjected to various test programs as appropriate. In order to effect a significant cost and schedule reduction, only one of the four array sections will be assembled with live solar cells. The other three will be completely fitted with actual hardware, except that the solar cells will be simulated with equivalent-mass glass chips.

The PTM schedule shows an approach to integration and testing that is similar to the sequence described for the ETM in Phase II. One important addition is the testing of individual live solar panels prior to assembly into the array section. After completing assembly, the panels will be illuminated to verify their total electrical performance. The panels will then be tested for approximately 10 orbit-simulating temperature cycles under thermal-vacuum conditions. Of particular importance during this test sequence will be the proper simulation of the appropriate rate of change of temperature on the panel (usually of the order of 10 to 20 $^{\circ}$C per minute). This test is particularly effective in pinpointing poor solder joints that might prove troublesome in orbit. After the thermal-vacuum test, the panels will again be illuminated and the results compared with the previous test data. The individual panels will then be assembled as an array section, deployed through a number of operations and subjected to further test and assembly operations.

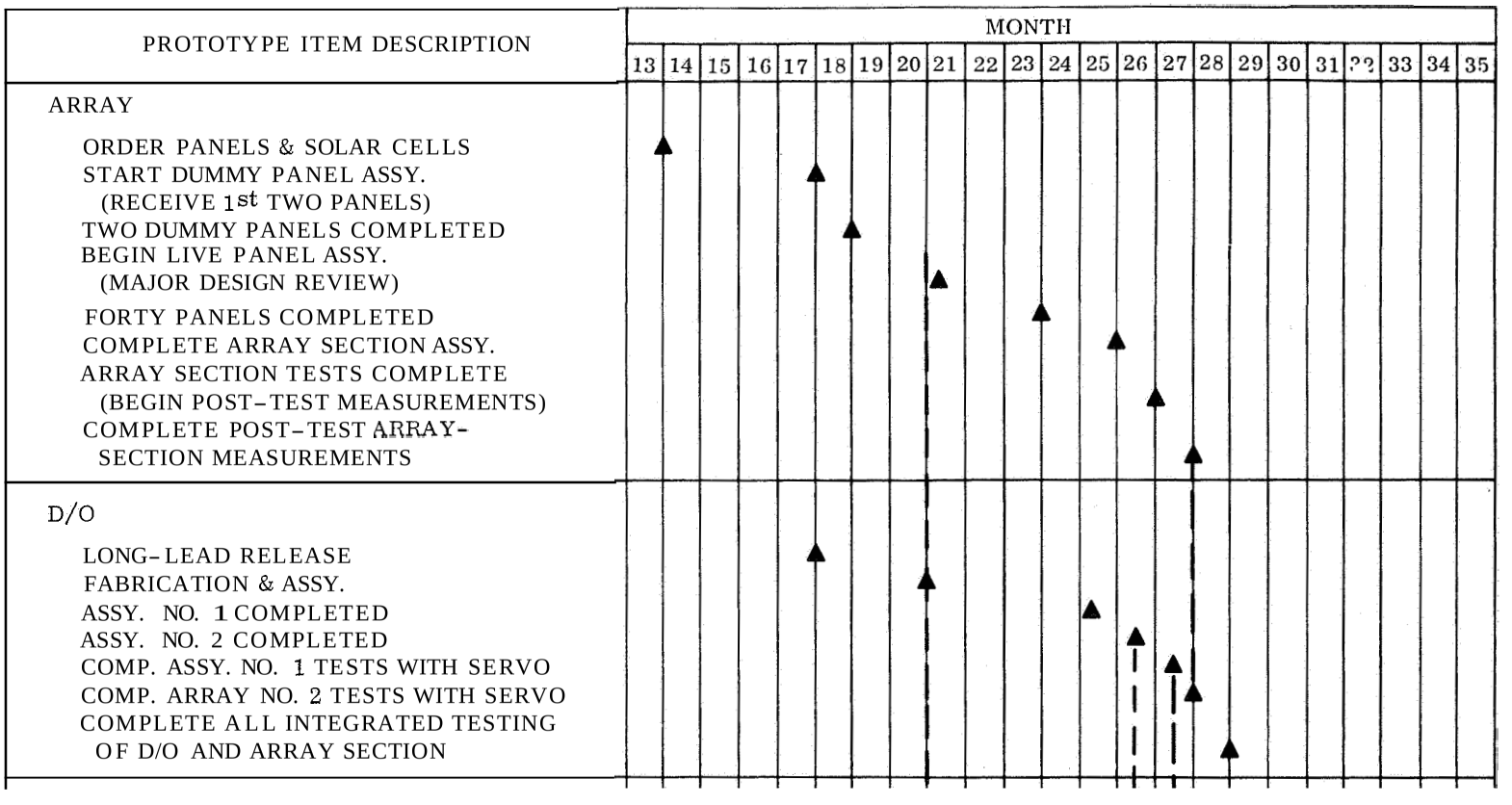

PROTOTYPE ITEM DESCRIPTION
MONTH
13 14 15 16 17 18 19 20 21 22 23 24 25 26 27 28 29 30 31 ?? 33 34 35
ARRAY
ORDER PANELS & SOLAR CELLS
START DUMMY PANEL ASSY.
(RECEIVE 1st TWO PANELS)
TWO DUMMY PANELS COMPLETED
BEGIN LIVE PANEL ASSY.
(MAJOR DESIGN REVIEW)
FORTY PANELS COMPLETED
COMPLETE ARRAY SECTION ASSY.
ARRAY SECTION TESTS COMPLETE
(BEGIN POST-TEST MEASUREMENTS)
COMPLETE POST-TEST ARRAY-
SECTION MEASUREMENTS
D/O
LONG-LEAD RELEASE
FABRICATION & ASSY.
ASSY. NO. 1 COMPLETED
ASSY. NO. 2 COMPLETED
COMP. ASSY. NO. 1 TESTS WITH SERVO
COMP. ARRAY NO. 2 TESTS WITH SERVO
COMPLETE ALL INTEGRATED TESTING
OF D/O AND ARRAY SECTION

Figure V-5. Phase III Prototype Test Model (PTM) Pian

4. Phase IV Test Plan

The flight hardware will be complete in every respect, with solar cells and all other components. The flight-hardware program is shown in Figure V-6. The program is similar to the prototype program except that the test levels will be reduced from prototype to flight-acceptance levels.

E. GROUND SUPPORT EQUIPMENT (GSE)

Ground support equipment requirements have been generated on the basis of analysis of the fabrication and test schedules previously presented. Tables V-3 through V-8 summarize the various pieces of handling and test equipment necessary for supporting the outlined program. Figures V-7 through V-13 illustrate various pieces of the test equipment.

Of special importance is the testing of the solar array deployment mechanism in a zero-g field. In order to simulate a zero-g environment for the test, a fixture similar to that shown in Figure V-13 will be required; the deployment of the primary panels is shown in view A. The weight of the panels is relieved by cords attached to the centers of the panels and to the centers of support members overhead hinged at distances coincident with those of the panel. These members would be attached to a support structure by means of roller bearings. During the deployment sequence, a force would be exerted just equal to that required to overcome the friction in the bearings and hinges.

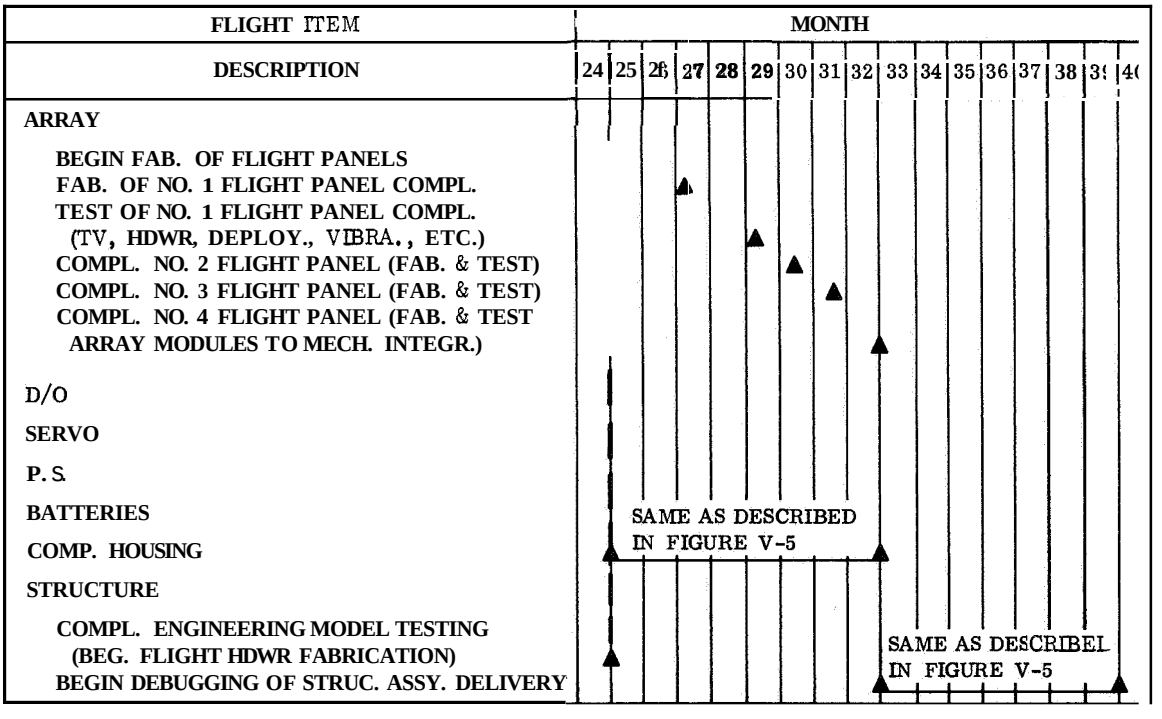


Figure V-6. Phase IV Flight Model Plan

TABLE V-3. ASSEMBLY, HANDLING, AND TRANSPORT EQUIPMENT

Equipment	Function
Solar-Panel Handling and Transport Dolly (Figure V-7)	To facilitate handling of individual solar panels; can be used for solar-panel assembly, assembly of panel hardware, transport to test areas, etc.
Array Section Assembly (Figure V-8)	To facilitate assembly of individual solar panels into a one-panel array section.
Array Section Transport and Deployment Support Device (Figure V-9)	(1) To facilitate transport of array sections to higher assembly stations. (2) To permit manual, supported deployment of an array section, as for illumination testing in horizontal position.
Power System Structure Dolly (Figure V-10)	To facilitate transport of the structure during integration and testing activities.
Component Housing Assembly and Transport Device (Figure V-11)	(1) To facilitate assembly and wiring of the component housing. (2) To provide a means for transport during integration and test.
Boom and Array Drive Assembly, Test and Transport	(1) To facilitate assembly of the array deployment and orientation components. (2) To provide a means for manually assisted deployment to check fits. (3) To provide a means of transport.

The deployment of the secondary panels is shown in view B of Figure V-13. After primary panel deployment, additional cords would be assembled through the centers of the secondary panels and attached overhead through the support structure to a pressure-equalizing unit designed to allow the support cords to unwind while maintaining the proper pressure on the cords to relieve the weight of the panels. An alternate to this approach for secondary-panel deployment would require rotation of the deployed primary panels with secondary panels folded to such a position that the rotational axis of the secondary panels is vertical. This would be, at best, an unwieldy operation and would require a very large vertical-clearance facility.

TABLE V-4. MONITOR AND CHECKOUT, MECHANICAL OPERATIONS

Equipment	Function
Zero-g Development Fixture, with Control and Monitor Console (Figure V-13)	To permit deployment tests of the solar array sections, simulating zero-g conditions; console displays measurements of deployment time spatial characteristics, etc.
Orientation System Checkout	(1) MODE 1 — to test orientation servo circuit operating with simulated position signals and equivalent compliance of orientation drive and panels. (2) MODE 2 — to test orientation servo and drive, with equivalent panel compliance loads on drive system; position signals can be simulated, or obtained from sun sensors illuminated with sun source.

TABLE V-5. MONITOR ANI) CHECKOUT, ELECTRICAL PERFORMANCE

Equipment	Function
Battery Tester (Figure V-14)	To test individual storage cells and/or packaged batteries.
Electrical Component Testers (Power System and Panel Orientation)	To test individual electrical components, such as regulators, converters, inverters, battery chargers, telemetry monitors, etc. The exact tester design will be a function of final package configurations.
Ordnance Circuit Tester	To test firing circuits for deployment initiators.
System Tester (Figure V-15)	To support integrated testing; includes solar I-V simulator (electrical analog), programmed loads, and monitor equipment.
Solar Panel Illuminator (Figure V-16)	Tungsten test set to permit illumination tests of individual panels and panel circuits; includes programmed load for producing I-V plot.
Assembled Array Test Unit (Figure V-12)	Permits sunlight or artificial testing w/o disassembly of panels from array section; includes a "collimating" box, sized to fit one panel; opaque panel covers to shade panels not under test; used in conjunction with array section transport and deployment support device (Table V-3).

TABLE V-6. ENVIRONMENTAL TEST FIXTURES

Test Article	Required Fixtures			
	Shock	Vibration	Thermal-Vacuum	Temp. - Humidity
PSM Structure, Integrated		x	x	
Component Housing, Integrated		x	x	x
Orientation Servo Assembly	x	x		x
Power System Electrical Components	x	x		x
Storage Batteries	x	x		x
Deployment and Orientation Assembly	x	x	x	x
Array Section	x	x		x
Solar Panels			x	x

TABLE V-7. FIXTURES FOR CHARACTERISTICS MEASUREMENTS

Fixture	Function
Weight and Balance	Required for all major assemblies.
Deflection/Compliance	To measure deflection and compliance parameters required for orientation system and dynamic interface analyses.

TABLE V-8. SHIPPING CONTAINERS REQUIRED

Solar Panels with Assembly Hardware
Array Section without Panels
Electrical Component Housing
Deployment/Orientation Components
All Electrical Components
Storage Batteries
Deliverable Ground Support Equipment

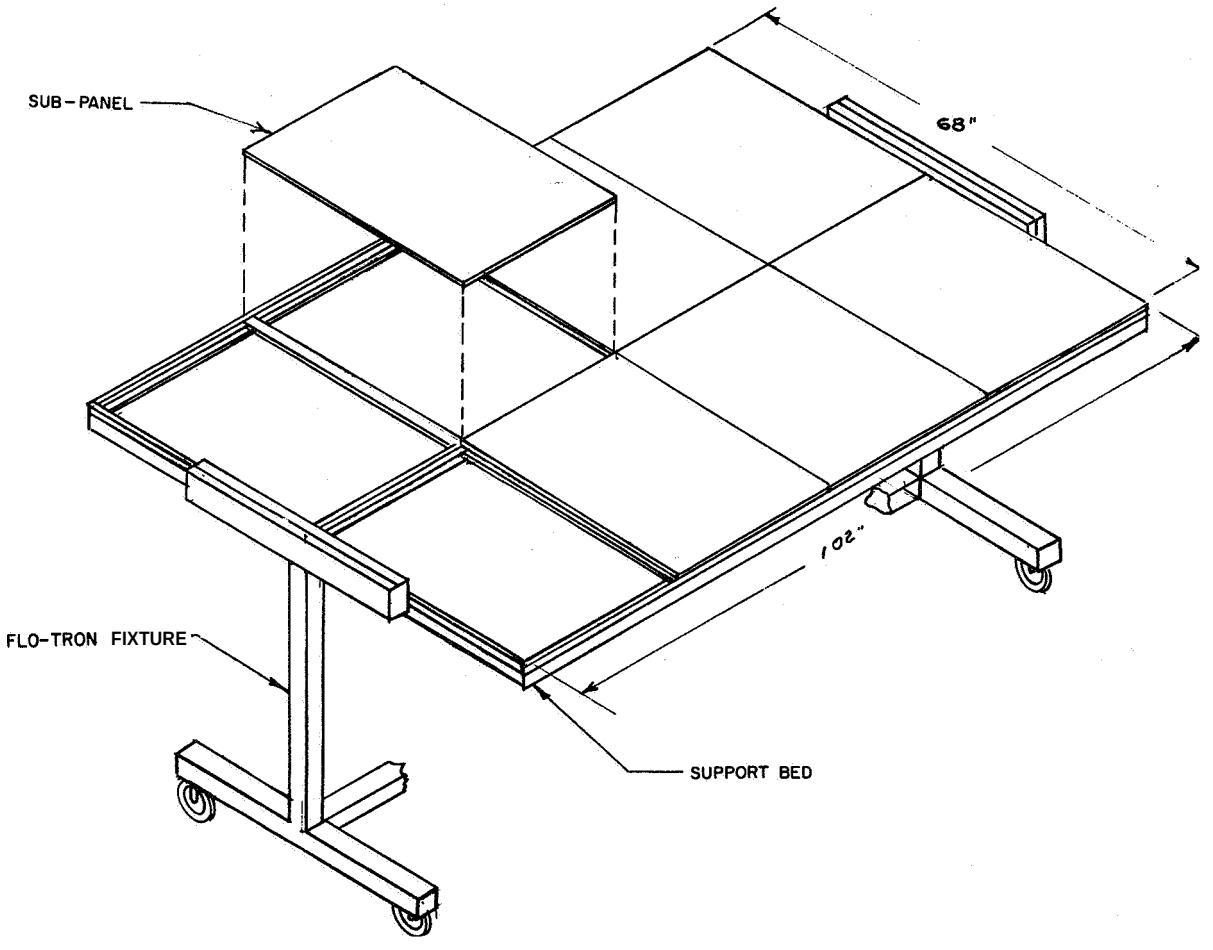


Figure V-7. Solar-Panel Handling and Transport Dolly

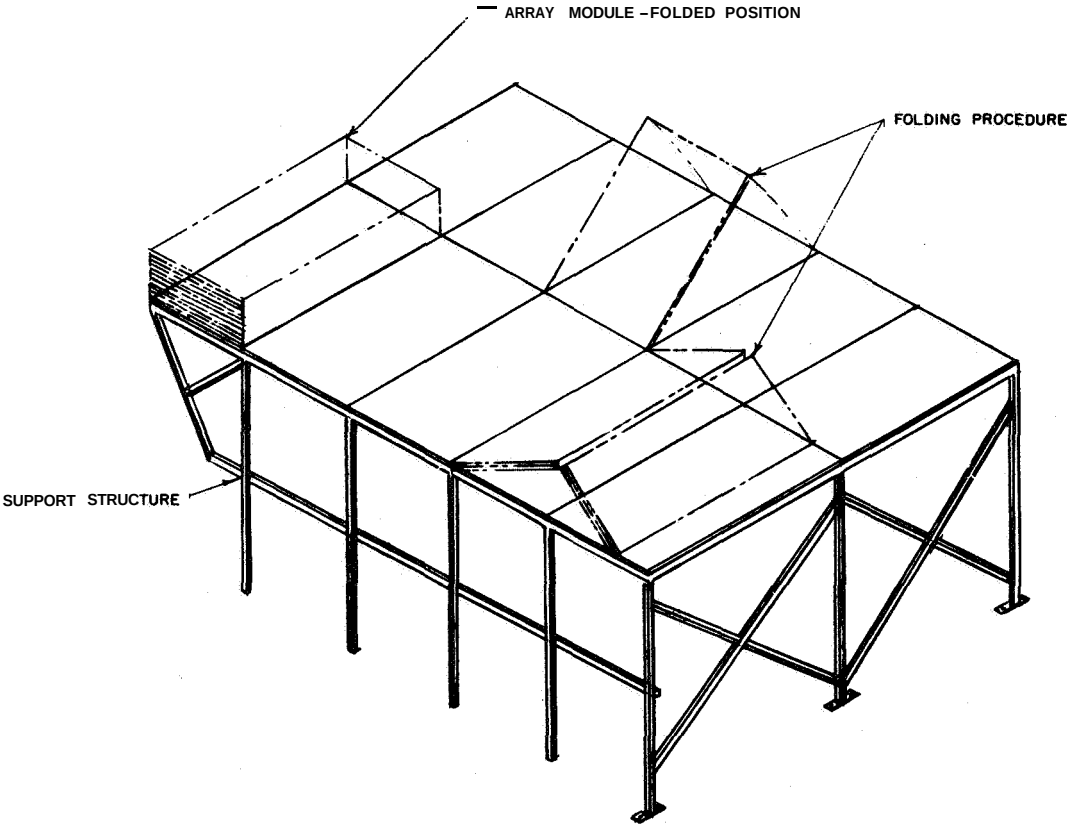


Figure V-8. Array-Section Assembly Fixture

A spacecraft simulator has not been included in the GSE complement. However, it is recommended that combined tests between a simulated deployed array with an active drive system and a spacecraft simulator that can react to dynamic inputs to be performed for the reasons previously cited.

F. MANUFACTURING AND FACILITIES PLAN

The fabrication, assembly, and test operations for the PSM hardware will require careful organization and planning because of the very large number of small, fragile component parts that will be processed (for example, approximately 250,000 2 cm by 3 cm solar cells per flight array) and the many moving mechanisms requiring careful assembly, alignment, and handling. These factors, combined with relatively short production cycles, will require that manufacturing and facility planning and implementation start early in Phase I.

1. Manufacturing

Beginning with Phase II, production will be initiated in quantities large enough to require that "hard" tooling be developed for the engineering-model hardware. Possible exceptions might be the engineering-model structure, (which could be assembled with prototype tooling), although the proposed overlap in schedule with the prototype and flight hardware would require timely acquisition of permanent tools.

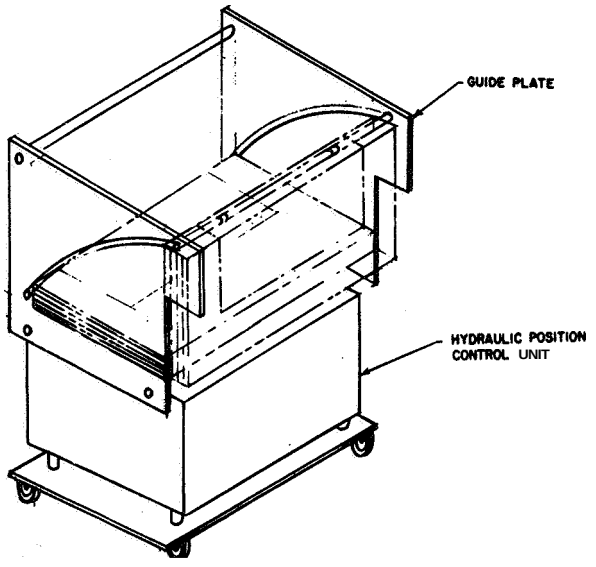


Figure V-9. Array Section Transport and Deployment Support Device

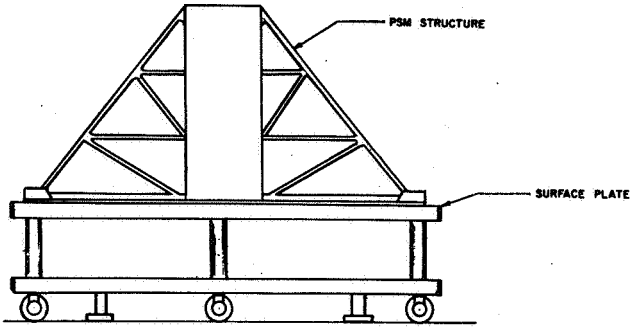


Figure V-10. Power System Structure Dolly

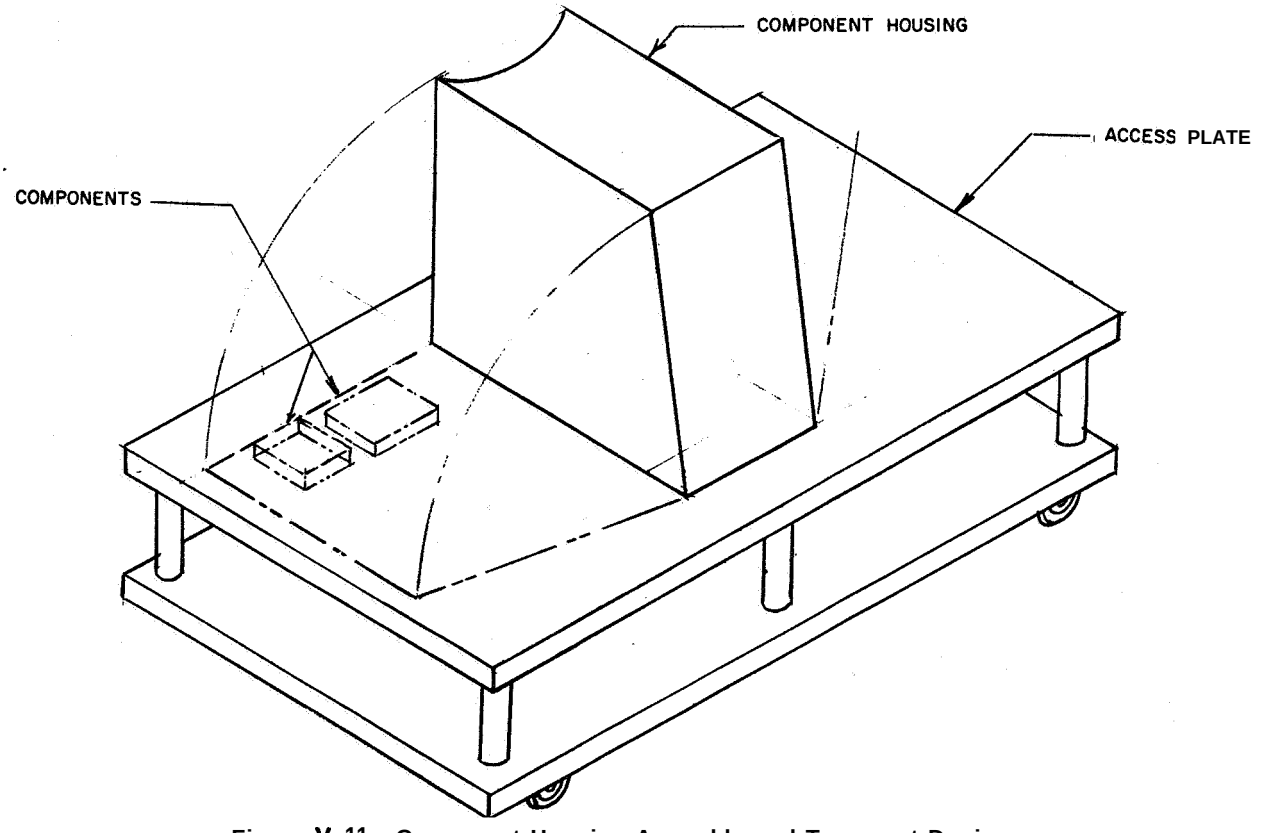


Figure V-11. Component Housing Assembly and Transport Device

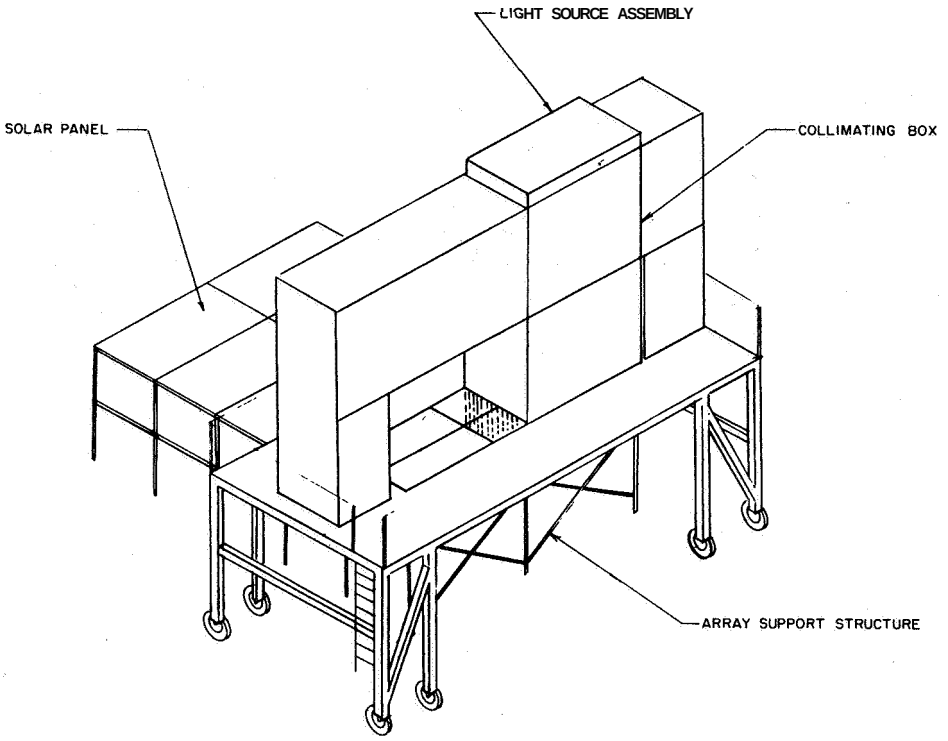


Figure V-12. Assembled Array Test Unit

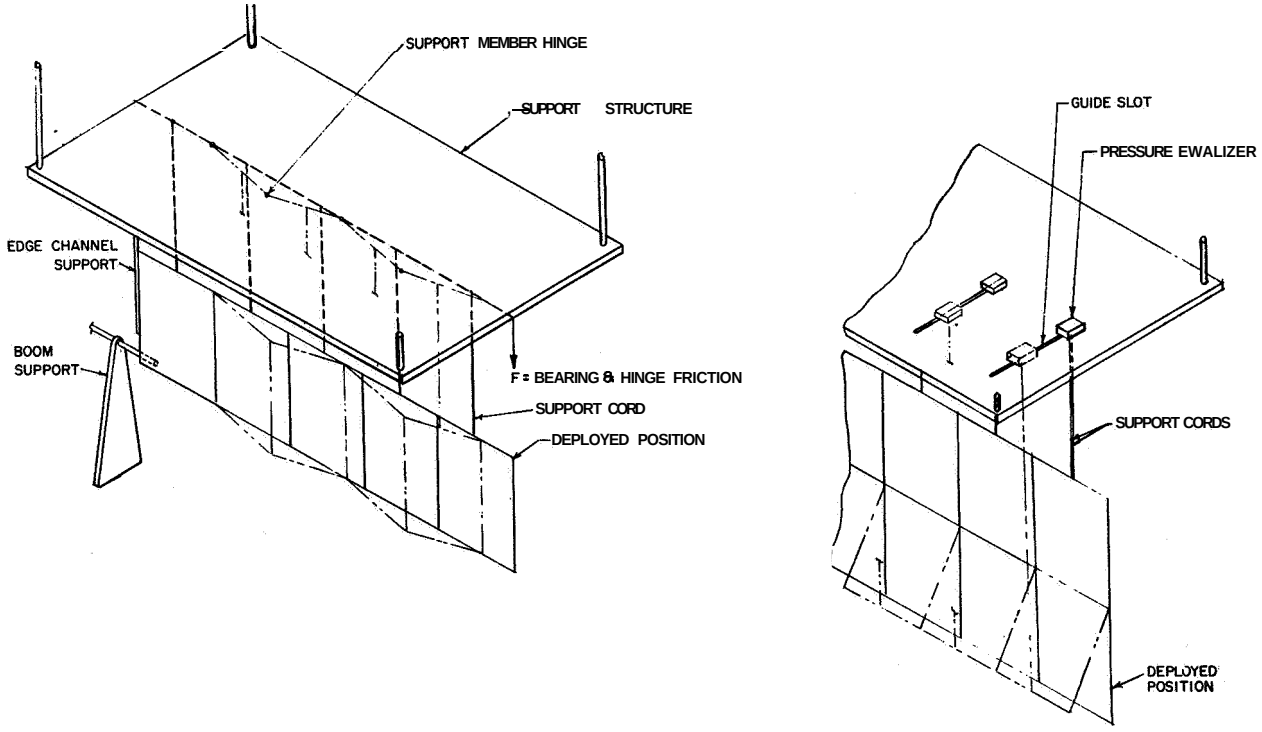


Figure V-13. Zero-g Deployment Fixture

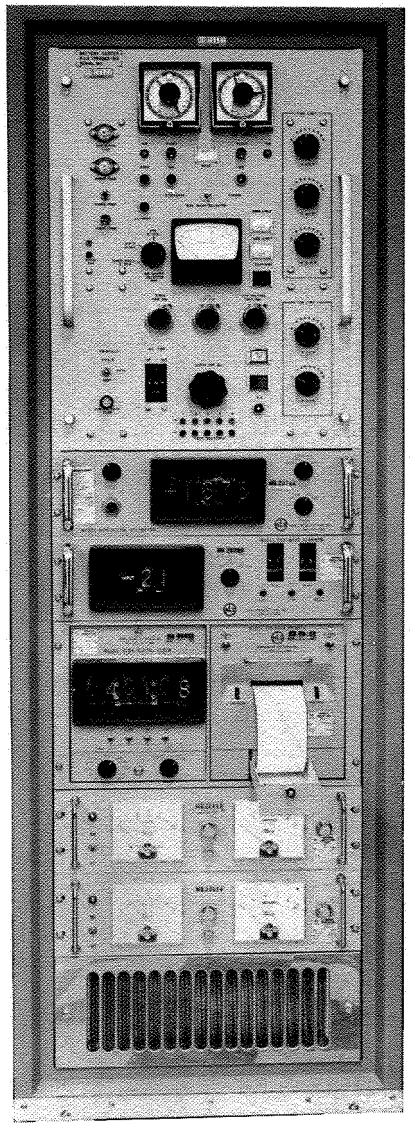


Figure V-14. Typical Battery Test Station

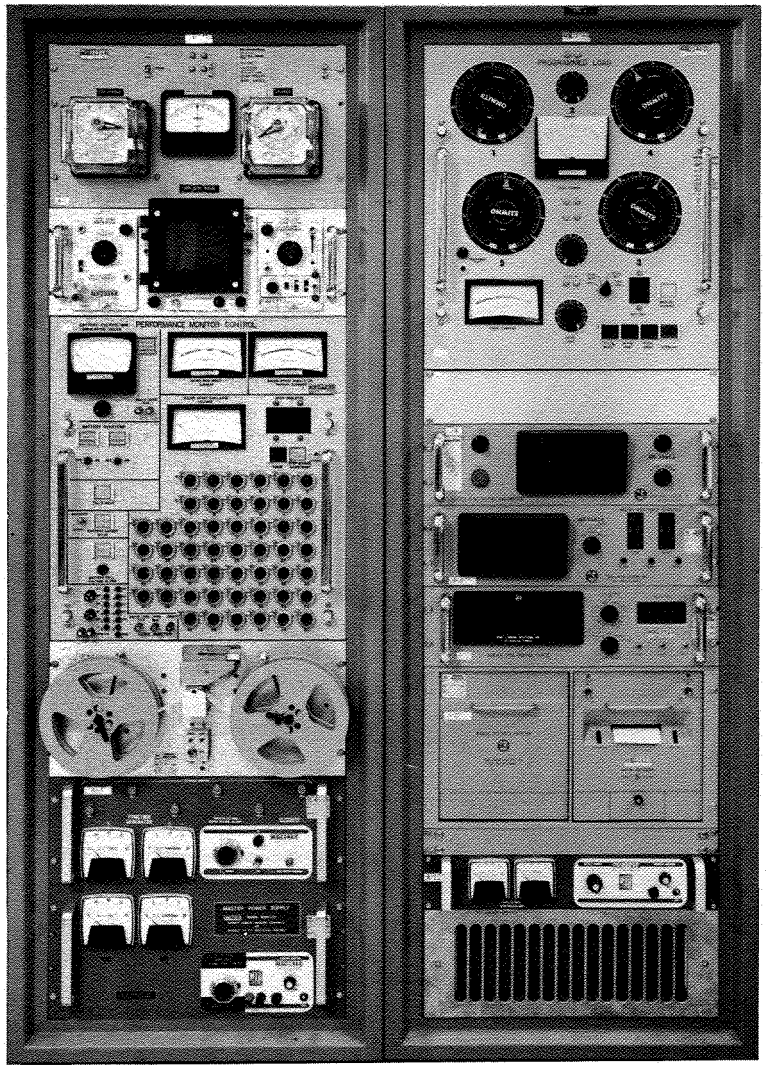


Figure V-15. Typical System Tester

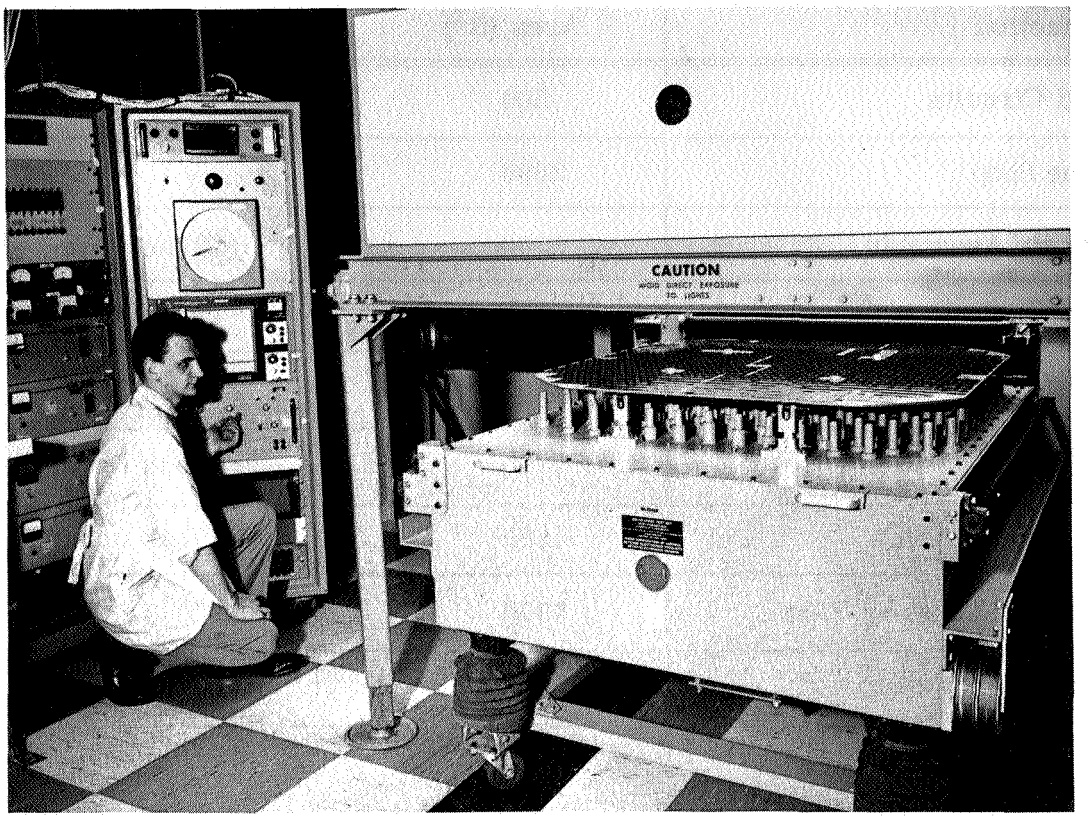


Figure V-16. Typical Solar-Array Illuminator

No state-of-the-art problems are anticipated in any of the proposed materials, manufacturing, or processes. Therefore, the principal problem will be to evaluate the required production rates and to design the production process accordingly. To assure maximum reliability, automatic or semiautomatic operations will be designed to the maximum extent possible.

In support of the actual manufacturing, a well-planned purchased-materials inspection and in-process inspection must be integrated into the manufacturing flow.

2. Facilities

Table V-9 is a breakdown of the facility space requirements for the fabrication and test of the power supply module. In summary 16,500 square feet of clean-room area will be required for the fabrication and test of the sub-assemblies (array, batteries, etc.), 10,000 square feet of high-bay area will be required for the integration of the sub-assemblies and system test, and 10,000 square feet of environmental-test area will be required for development and qualification testing. The environmental facilities required are shown in Table V-10; the vibration facilities shown would be utilized to verify the analytical model of the system

TABLE V-9. ESTIMATED SPACE REQUIREMENTS

Function	Area (ft^2)	Type
Printed Circuits	500	Clean Room
Machine Shop	2000	
Electronic Assembly	1500	Clean Room
Conformal Coating	500	Clean Room
Array Fabrication	4000	CleanRoom
Sub Panel Integration	2200	CleanRoom
Module Integration	4800	CleanRoom
Electronics: Battery Test	1000	Clean Room
Power System Integration	10,000	High Bay
Vibration Test, Shock	5600	Environmental High Bay
Thermal-Vacuum Testing	4000	Environmental High Bay
,Temp. Humidity	500	Environmental High Bay
TOTAL	39,500	

only. If it is necessary to test to actual anticipated loads, a hydraulic shaker of 50,000 to 100,000 pounds of thrust would be required. The frequency range would, however, be limited with such a shaker.

G. BUDGETARY COST ESTIMATE

A budgetary cost estimate has been prepared for the entire program from initial design through tested flight hardware. The program details are in accordance with those described in the previous paragraphs, as summarized in the following.

TABLE V-10. TYPICAL ENVIRONMENTAL FACILITIES REQUIRED

Test	Facility
Thermal-Vacuum, System	25-ft internal diameter by 20 ft high thermal-vacuum chamber
Thermal-Vacuum, Panels	12-ft internal diameter by 4 ft long thermal-vacuum chamber
Thermal-Vacuum, Comp. Housing and Boom Assembly	12-ft internal diameter by 15 ft long thermal-vacuum chamber
Vibration-Shock	Cluster of four 15,000-lb-thrust shakers
Temperature-Humidity	14 by 14 by 17 ft temperature humidity chamber
Component Tests	Variety of small environmental facilities, such as 5000 lb. shakers, bell jars, etc.

1. **Phase I**

Generation of a complete set of 'build-to" drawings and specifications for the first piece (engineering test model or ETM) and for supporting test equipment. Effort will include design, supporting analysis, breadboard and other special tests to accumulate design information.

2. **Phase II**

Fabrication and test of a complete engineering test model and supporting test equipment, except that the solar array will consist of 19 panel substrates with bonded glass chips and one "live" panel. Special test articles will be fabricated and tested to accumulate long-term performance data that it is not possible to acquire from the integrated engineering model. Final drawings and specifications for prototype and flight hardware will be produced.

3. Phase III

Fabrication and test of a complete system for prototype tests, including additional items of supporting test equipment. The solar array will consist of a full complement of 40 panels, except that only 10 of the 40 will be assembled with solar cells.

4. Phase IV

Fabrication and test of a complete system of flight hardware. No costs are included for special building construction, engineering support, or special equipment models that may be required for spacecraft-integration support. Equipment delivery is assumed as FOB, Princeton, New Jersey.

Table V-11 shows the budgetary estimate by major cost element and by phase. The cost figures for each phase are broken into engineering labor and associated costs for materials, subcontracted services, facilities, and fabrication labor. The figures shown may vary upward or downward depending on the final program details as agreed upon by NASA.

TABLE V-11. BUDGETARY COST ESTIMATE

Item	Cost (Thousands of Dollars)								
	Phase I		Phase II		Phase III		Phase IV		Total Program
	Labor	Hdw.	Eng. Labor	Hdw.	Eng. Labor	Hdw.	Eng. Labor	Hdw.	
Power System Hardware									
System	170	-	320	-	-	-	-	-	490
Solar Array	1110	80	810	1930	80	4260	10	6150	14430
Battery & Electronics	660	50	790	610	80	670	10	670	3540
Structured/Comp. Housing	1960	280	270	300	80	200	10	200	3300
Integration & Test									
Mech. Integ,	680	-	680	140	190	460	190	-	2340
Elect. Integ.	810	-	530	140	2380	700	1540	-	6100
Environmental Test	300	-	710	-	280	600	80	260	2230
Program Control/Documentation	460	70	390	-	280	-	130	60	1390
Column Total	6150	480	4500	3120	3370	6890	1970	7340	33820
Per Phase Total	6630		7620		10260		9310		33820