GPO PRICE $ ____________

CFSTI PRICE(S) $ ____________

Hard copy (HC) 3.00

Microfiche (MF) .75

ff 653 July 65

FACILITY FORM 602

N65-33877
(ACCESSION NUMBER)

(THRU)

69
(PAGES)

1
(CODE)

CR-64889
(NASA CR OR TMX OR AD NUMBER)

03
(CATEGORY)

RE-ORDER NO. 65-323
Vol. I

DOCUMENT NO. 65SD4286
1 May 1965

SOLAR THERMIONIC FLIGHT EXPERIMENT STUDY

FINAL REPORT

VOLUME I - STUDY SUMMARY

PREPARED UNDER

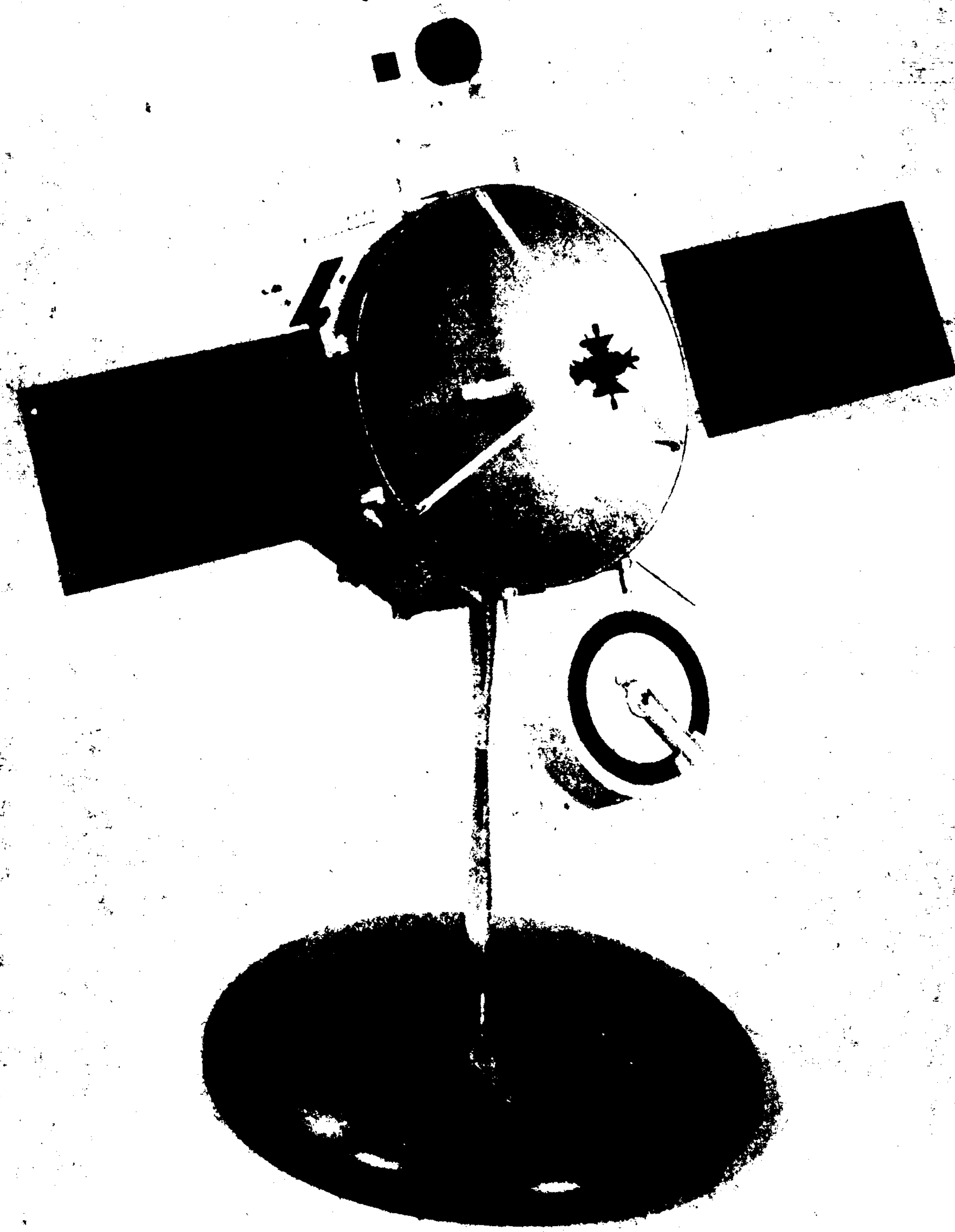
CONTRACT NO. 950852

FOR

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
4800 OAK GROVE DRIVE
PASADENA, CALIFORNIA 91103

This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, sponsored by the National Aeronautics and Space Administration under Contract NAS7-100.

GENERAL ELECTRIC
SPACECRAFT DEPARTMENT
A Department of the Missile and Space Division
Valley Forge Space Technology Center
P.O. Box 8555 • Philadelphia 1, Penna.



Model of Solar Thermionic Flight Experiment Spacecraft

VOLUME I

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF TABLES

VOLUME II

TABLE OF CONTENTS

TABLE OF CONTENTS (CONT.)

TABLE OF CONTENTS (CONT.)

VOLUME III

TABLE OF CONTENTS

Section Page

TABLE OF CONTENTS (CONT.)

TABLE OF CONTENTS (CONT)

RE ORDER

TABLE OF CONTENTS (CONT)

TABLE OF CONTENTS (CONT)

VOLUME IV

TABLE OF CONTENTS

TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

FORWORD

This final report covers a study program conducted from July 1964 to April 1965 by the General Electric Spacecraft Department for the Jet Propulsion Laboratory, under Contract No. 950852. The program was carried out under the direction of E.W. Williams who served as Program Manager and Project Engineer. The solar thermionic generator portion of the study was performed by Thermo Electron Engineering Corporation (TEECO) under subcontract to the General Electric Company. Dr. P. Brosens was responsible for the work at TEECO. The program was under the technical direction of R. Boring of the Jet Propulsion Laboratory.

Because of the many technical disciplines involved, engineers and scientists throughout the General Electric Missile and Space Division made contributions to this study. The principal contributors in each of the major technical areas are identified in the report.

Figures describing the Delta series of launch vehicles and their performance were provided through the courtesy of the Missile and Space Systems Division of the Douglas Aircraft Company.

RE-ORDER No. 61-323

ABSTRACT

33877

This report covers a study program to establish the governing parameters for an experimental flight test of a solar thermionic electrical power system. Three different missions are suggested for the flight experiment. Selection of the orbit, launch vehicle, appropriate secondary experiments and thermionic system size is made for each mission. Conceptual spacecraft designs are included for each of the suggested missions. Conceptual designs are developed for the primary and secondary experiments, and the major spacecraft subsystems. Drawings are presented for the resulting spacecraft configurations. The results of the study indicate that a successful thermionic flight experiment is within the present component state-of-the-art. In addition the study results show that a sun pointing spacecraft of the type required for conducting a solar thermionic flight experiment offers an attractive space platform from which many valuable scientific and engineering experiments could be performed.

Author

1. STUDY SUMMARY

1.1 INTRODUCTION

A thermionic converter is a static device which is capable of converting heat energy directly into electrical energy by utilizing the thermionic emission process. The thermionic converter is basically a simple device consisting of a hot emitter (1600°-1800°C) emitting electrons to a lower temperature collector (700°-1000°C). The electrons return to the emitter through an externally connected load resistance where they can be made to perform useful work. Thermionic converters may be used in conjunction with a variety of thermal energy sources such as:

Nuclear	Chemical
Radioisotope	Solar
Fossil Fuel	

This study deals with solar thermionics. In the case of solar thermionics the sun in conjunction with a parabolic solar concentrator is the source of thermal energy. The concentrator collects the solar energy and concentrates it up to 12,000 times into a cavity-absorber consisting of the emitter faces of several thermionic converters. A flight experiment designed to demonstrate the feasibility of solar thermionics as a means of supplying spacecraft electrical power is the object of this study.

In 1958 it became apparent that the power output and efficiency of thermionic converters could be greatly improved as a result of two developments. The first of these was the realization that by suitably arranging the work-function difference between the emitter and collector the converter power output and efficiency could be greatly increased. The second improvement was the development of effective methods of overcoming electron space charge. Some of the techniques employed to accomplish this are: (1) close emitter-collector spacing, and (2) the introduction of positive ions into the interelectrode space.

The most important development during this time period was the introduction of cesium vapor into the converter interelectrode space. This gas is absorbed on the collector surface producing the desired low collector work function. Also, because of its low ionization potential, cesium is readily ionized either by surface ionization at the hot emitter or by electron impact in the interelectrode space, and therefore, produces positive ions which help to overcome electron space charge effects.

Since 1958 rapid progress has been made in the development of thermionic converters. From the research laboratory devices of 1958, which produced milliwatts, have evolved converter designs suitable for engineering applications which at an emitter temperature of $2000^{\circ}K$ have demonstrated efficiencies to 17 percent with power densities of 20 watts per square centimeter. Engineering hardware converters are currently operating which have recorded over 6000 hours of continuous life testing without a measurable decrease in performance. Similar converters have been subjected to 20 g mechanical vibration levels and 100g shock levels with no sign of performance degradation.

NASA sponsorship of the development of solar thermionics as a potential spacecraft power source was initiated in 1961. The collaboration of several prominent industrial contractors was solicited in the development efforts. High performance and long life converters have been achieved as a result of this collaboration. Cavity-absorber designs have been developed and tested which minimize the thermal losses from the generator. The capability to fabricate light-weight solar concentrators of the required geometrical accuracy has been demonstrated. Several solar thermionic generators have been developed and ground tested. One of these generators is shown in Figure 1-1 undergoing solar testing. Thermionic generator power outputs as high as 138 watts at 6.1 percent efficiency have been obtained during solar testing of a five converter generator.

Solar thermionics has reached the stage of development where a flight experiment is both feasible and necessary if it is to be developed as a spacecraft electrical power source. A flight experiment is needed to accomplish the following:

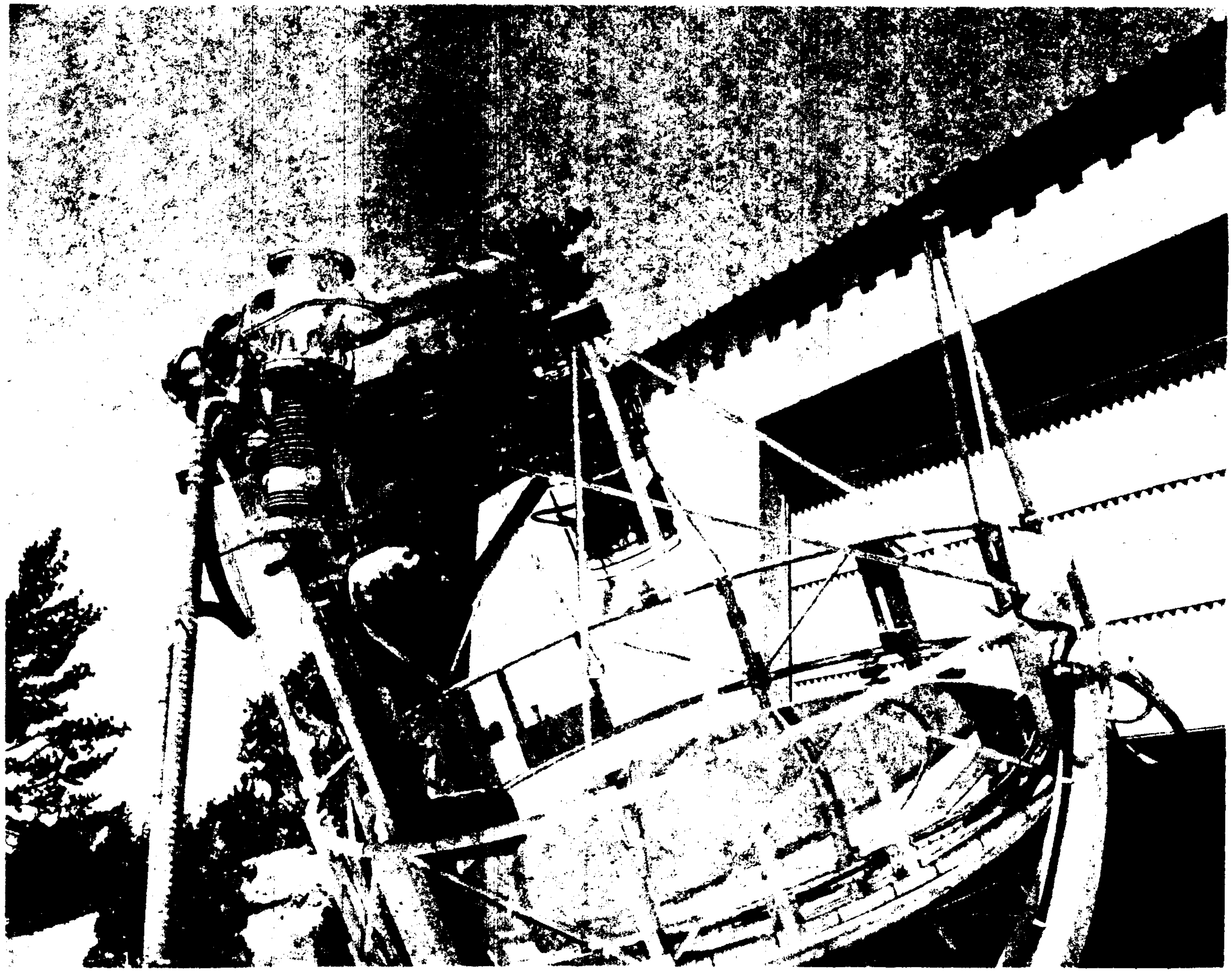


Figure 1-1. **Solar** Testing of JPL JG-2 Thermionic Generator

RE-ORDER No.

a. Demonstrate, in the space environment, the feasibility of solar thermionics as a means of supplying electrical power for space vehicles. This type of test is an essential step in the development of an operational system.

b. Determine the effects of the space environment on the solar thermionic system components, such as the solar concentrator reflectivity, thermal stability and micrometeoroid resistance. It is impossible to adequately evaluate, by simulated earth testing, the effects of such space environmental factors as:

   Temperature
   Pressure
   Micrometeoroids
   Electromagnetic Radiation
   Geomagnetic Fields
   Charged Particles
   Zero Gravity

   Simultaneous earth simulation of these quantities is impossible for two principal reasons: (1) The uncertainty in the space environment and (2) The difficulty in simulating effectively those characteristics known to be present. The problems of simulation become particularly difficult when the necessity for long duration tests and combinations of physical effects are considered.

c. Measure the thermionic system performance in the space environment and correlate this with the estimated performance. With existing facilities it is impossible to earth test a completely assembled solar thermionic system designed for space operation. The major difficulty arises as a result of the need to simulate the solar energy input available outside the earth's atmosphere. If the tests are conducted using the solar energy available at earth the energy density (watts/ft^2) is inadequate (approximately 90 watts per square foot maximum compared with the 130 watts per square foot available above the earth's atmosphere). On the other hand there is no known solar simulator available, or planned, that can provide the high degree of collimation required for testing a solar thermionic system. As a result it appears that the solar thermionic system components will continue to be tested separately and the total system performance estimated from the component performances. It also follows that the components will have to be tested under conditions different from those they will be expected to operate under and the performance estimates at the design point based on extrapolations of these data. It will be essential that actual system performance in space be correlated with estimated performance based on earth tests.

d. Demonstrate the attitude control subsystem's ability to maintain the high degree of sun pointing accuracy required (± 0.1 degrees). As experience on the OAO Program has shown, this is very difficult and expensive to accomplish on earth primarily because of the anisoelastic effects (effects of gravity on the non-rigid spacecraft structure).

1.1.1 OBJECTIVES

The objectives of this study were to:

a. Define the requirements for a solar thermionic flight experiment including determination of suitable solar thermionic system sizes and selection of appropriate orbits and launch vehicles.

b. Identify additional experiments that are of scientific and engineering importance that could be included, in a secondary capacity, as part of a solar thermionic flight experiment.

c. Develop conceptual designs for a spacecraft capable of conducting the primary solar thermionic flight experiment and the selected secondary experiments.

1.1.2 GROUND RULES

The imposed constraints are summarized here since they represented the starting point for the study.

Five basic types of orbits were specified for consideration:

a. Circular orbit with a 5 to 3 light to dark ratio
b. Sun-synchronous daylight orbit
c. Elliptical orbit with a high light to shadow ratio
d. Stationary earth synchronous orbit
e. Solar probe orbit.

In addition to the specified orbits, five ground rules were established at the outset of the study.

a. **The experimental thermionic system must generate a minimum of 100 watts.**

b. The thermionic generator would contain four converters.

c. The spacecraft power requirements would be supplied by a combination of silicon solar cells and rechargable nickel-cadmium batteries. To properly evaluate the performance of the solar thermionics portion of the experiment, it is necessary to have a separate spacecraft power supply; so there is no dependence on the power generated by the thermionic generator.* In this manner, if there were a degradation or complete failure of the thermionic generator, the spacecraft would still be operational and the reasons for the failure could be evaluated. In addition, failure of the thermionic generator would by no means put an end to the value of the experiment. Valuable information could still be obtained from the orientation and solar concentrator subsystems which will be instrumented to allow performance evaluation. Also, the secondary experiments would continue to yield valuable engineering and scientific data.

d. The number of ground stations needed for the telemetry, tracking and command functions and the demands on these stations were to be minimized in order to avoid priority problems and reduce the cost of data acquisition and handling.

e. The study was to be based on a spacecraft design life of one year. A design life on the order of one year is considered desirable for complete evaluation of such factors as the effects of the space environment on the life of thermionic converters and the performance of solar concentrators.

f. A 1967-69 flight period was assumed in formulating the spacecraft designs. This period is considered to be in keeping with the present rate of development of the solar thermionic electrical power system.

In addition to the specific ground rules outlined above, the governing philosophy was to design a solar thermionic experiment, based on present state-of-the-art capability, that will demonstrate solar thermionic system performance in the space environment. In addition, this experiment will provide the technical information needed for further refinement of this static energy conversion system thus achieving the original NASA goals of developing an attractive means for providing spacecraft electrical power.

---

*Although the experiment design proposed in this study does not include such a feature, it is possible to design the experiment so that during portions of the mission, the solar thermionic system could be used to supply all or part of the spacecraft power requirements.

The results presented in this report conform with the ground rules outlined here.

1.1.3 CONCLUSIONS

The results of this study indicate that a successful solar thermionic flight experiment is possible with present state-of-the-art capability. This experiment will allow complete evaluation of all the major components making up a solar thermionic electrical power system.

In addition, the study results show that a spacecraft of the type required for conducting a solar thermionic experiment offers an attractive space platform, principally because of its highly accurate sun pointing feature, from which many valuable scientific and engineering experiments could be performed.

1.1.4 STUDY APPROACH

To accomplish the study objectives the program was divided into two phases:

Phase I - Mission Analysis
Phase II - Spacecraft Design

The mission analysis considered various orbits, launch vehicles, solar thermionic system sizes and secondary experiments; and selected the most meaningful combinations of these elements for the first solar thermionic flight experiment. These selected combinations are referred to in this report as missions. Phase II of the study developed conceptual spacecraft designs for carrying out the missions defined in Phase I. These spacecraft designs include conceptual designs for the solar thermionic experiment, secondary experiments, vehicle instrumentation, TT&C Subsystem, attitude control subsystem, power subsystem and the structure necessary to integrate the experiments and subsystems into a spacecraft configuration.

1.1.5 REPORT ORGANIZATION

The Solar Thermionic Flight Experiment Study Report is divided into four volumes. These volumes break down along the lines of the Mission Analysis and Spacecraft Design phases described in the preceding paragraph. The contents of each volume are summarized in Table 1-1.

Table 1-1. Volume Breakdown

VOLUME	TITLE	CONTENT
I	Study Summary	Summary of Study Results
II	Mission Analysis (Phase I)	Mission analysis phase of the study including orbit selection, launch vehicle considerations, thermionic system performance estimates, evaluation of secondary experiments and spacecraft subsystem tradeoffs.
III	Spacecraft Design (Phase II)	Conceptual spacecraft designs for the missions defined in Phase I. Includes designs for the solar thermionic experiment, TT&C subsystem, attitude control subsystem, power subsystem, spacecraft configuration and identification of the instrumentation required.
IV	Appendices	Detailed calculations and supporting material that was not included in Volumes I, II or III.

The Study Summary presents only the study results with the details of how these results were obtained being included in Volume II and Volume III. These latter two volumes are divided into thirteen major sections. These sections are so arranged as to present a logical development of the spacecraft design when read in order.

1.2 SELECTED MISSIONS

Three potential missions were defined for a solar thermionic flight experiment. These missions are designated, in order of preference, as Missions A, B and C and their primary characteristics are summarized in Table 1-2. The only major difference between the selected missions is the orbit. Some minor component changes must be made for the different cases but the basic spacecraft structure is the same and a version of the same launch vehicle is used in all three cases. The primary solar thermionic experiment design is identical for each of the missions. The secondary experiments included on board the spacecraft vary somewhat between missions because of the different orbits employed, but the differences are small. A common spacecraft design offers the advantage of allowing the different missions to be flown with the same basic spacecraft, thus increasing the program flexibility and reducing the cost if more than one of the missions are undertaken.

The selected orbits and launch vehicles, and the spacecraft design are described in the following sections.

1.3 ORBITS

The advantages and disadvantages of the various orbits considered are summarized in Table 1-3. Based on these advantages and disadvantages the selected orbits are:

Mission A - Modified Sun-Synchronous
Mission B - Highly Elliptical
Mission C - Low Altitude Circular.

The stationary earth synchronous or solar probe orbits are not considered suitable for the first solar thermionic flight experiment. The stationary earth synchronous orbit was rejected primarily because of the cost and complexity associated with establishing and maintaining this type of orbit. The major objections to the solar probe orbit are the

Table 1-2. Summary of Mission Parameters

PARAMETER	MISSION A	MISSION B	MISSION C
Type of Orbit	Modified Sun-Synchronous	Highly Elliptical	Low Altitude Circular
Orbit Altitude, Nautical Miles	1000	25,000 apogee 200 perigee	325
Orbit Inclination, Degrees	101.84	45	30
Orbit Period, Hours	2.07	14.0	1.61
Orbit Maximum Dark Period, Hours	0.4	2.09	0.617
Launch Vehicle	IMPROVED DELTA DSV-3E	IMPROVED DELTA DSV-3E[a]	IMPROVED DELTA DSV-3H
Launch Site	WTR	ETR	ETR
Thermionic System Concentrator Diameter, Inches	50	50	50
Thermionic Generator Power Output, Watts	144	144	144
Thermionic System Efficiency, Percent	8.2	8.2	8.2
Secondary Experiments	See Table 1-7	See Table 1-7	See Table 1-7
Spacecraft Weight, Pounds	360	373	404

a - First Stage Floxed 30 Percent

Table 1-3. Advantages and Disadvantages of Various Orbits

ORBIT	ADVANTAGES	DISADVANTAGES
Low Altitude Circular (325 nm)	1-Can be obtained with a low cost, highly reliable launch vehicle.	1-Has a short period and low light-to-shadow ratio. This results in frequent thermal cycling and does not provide a long uninterrupted light period for evaluation of the solar thermionic experiment. 2-Of least interest for making scientific measurements because a relatively large number of experiments have already been conducted in this altitude range. 3-Because the boost capability required is relatively small, this orbit makes very inefficient use of any of the launch vehicles available. 4-Requires numerous ground stations for handling the TT&C functions.
Modified Sun-Synchronous Daylight (1000 nm)	1-Provides shadow free period for the early phase of the mission with shadow periods being introduced in the latter phase. 2-TT&C functions require a maximum of two ground stations. One station may be sufficient. 3-Can be obtained with a low cost highly reliable launch vehicle. 4-Altitude is of interest for making scientific measurements and ideal for conducting many of the secondary engineering experiments considered.	1-Falls in a severe region of the Van Allen radiation belt.
Highly Elliptical Orbit (200 by 25,000 nm)	1-High light-to-shadow ratio. 2-Scientific measurements of interest. 3-Can be obtained with a medium cost highly reliable launch vehicle. 4-Long orbit period reduces thermal cycling.	1-Requires numerous ground stations for handling TT&C functions. 2-Has lengthy shadow periods (2 hours). 3-Shadow history is erratic. 4-Has a long period which results in large quantities of data being collected per orbit.
Stationary Earth Synchronous (19,300 nm circular)	1-Can handle TT&C functions from one ground station. 2-Provides shadow free periods for the early phase of the mission with shadow periods being introduced in the latter phase.	1-Establishing and maintaining orbit requires a complex station keeping system. 2-Expensive launch vehicle (ATLAS-AGENA D) of relatively low reliability required. 3-Not very interesting for scientific measurements. 4-Has lengthy shadow periods (1.17 hours).
Solar Probe	1-Spacecraft always in the sun. 2-Secondary scientific experiments of great interest.	1-Large variation in solar intensity. 2-Effects of shadow cannot be evaluated. 3-Desirable orbit requires an [illegible] launch vehicle (ATLAS-AGENA [illegible]). 4-Launch vehicle relatively expensive and of low reliability. 5-High priority ground stations required (DSIF). 6-Requires directional antenna on board the spacecraft.

(a) This represents a disadvantage for the total spacecraft because of the potential radiation damage to the electronic components but is actually an advantage from the standpoint of testing the thermionic system and some of the secondary experiments under worst case conditions.

serious design problems resulting from the large change in solar intensity encountered throughout the orbit.

The selected orbits are discussed below.

1.3.1 MISSION A ORBIT

Primarily because it can provide an initial shadow-free period followed by the introduction of a shadow period in the latter phases of the experiment, the modified sun-synchronous orbit is far more attractive than any of the orbits considered for the thermionic experiment. The advantages provided during the initial shadow free period are:

a. An uninterrupted evaluation of the thermionic experiment can be made.

b. An uninterrupted evaluation can be made of all secondary experiments that require sun orientation.

c. With no shadow periods, the severe thermal cycling problems are eliminated and the solar thermionic system life and reliability increased.

d. Thermionic generator warm-up problems such as getting the cesium reservoir up to temperature after each dark period are eliminated.

e. The attitude control function is simplified because the sun reference is never shadowed.

f. The spacecraft thermal control problems are reduced.

g. The demands on the rechargable battery are greatly reduced.

The introduction of a shadow period in the latter phase of the mission gives the added advantage of allowing the effect of dark periods to be evaluated. Since the majority of space missions for which solar thermionics will be a potential power system would involve shadow periods, this information is needed. Other advantages are listed in Table 1-3 which contribute substantially to making this orbit the number one selection.

Since spacecraft weight is not a serious problem with the launch vehicles recommended, the solar cell shielding weight associated with the severe radiation environment encountered in this orbit is not considered a significant disadvantage. However, the degradation resulting from radiation induced surface effects on transistors and diodes will require further investigation. It appears that radiation induced surface effects could represent significant spacecraft design and/or life penalties but no firm conclusions can be drawn until this area is investigated in greater depth. To do so will require detailed circuit designs which are beyond the scope of this study. Since this represents the only apparent potential disadvantage of the modified sun-synchronous orbit this situation should be resolved.

1.3.2 MISSION B ORBIT

A highly elliptical orbit was selected as the second most suitable orbit for conducting a solar thermionic experiment. However, because it cannot provide a completely shadow free period it is considered far less attractive than the modified sun-synchronous orbit. Its major advantage, compared with the modified sun-synchronous orbit, is that the wide variation in orbit altitude provides a platform for performing valuable scientific measurements of such quantities as the proton and electron spectra. Although it does not offer a completely shadow free period, it does offer a high light to shadow ratio and a long orbital period. This assures long periods (12 hours or more) of daylight for operating the thermionics experiment and minimizes the thermal cycling.

Communication for the highly elliptical orbit is more difficult than it is with the modified sun-synchronous orbit and requires the use of more ground stations.

Also, the relatively long shadow period compared with the modified sun-synchronous orbit (2 hours compared with 0.4 hours) may create thermal control problems for the highly elliptical orbit.

1.3.3 MISSION C ORBIT

Of the five basic types of orbits considered, the low altitude circular orbit is ranked third. Although a solar thermionic experiment could be conducted from this type of orbit, it is not as attractive as the orbits selected for Missions A and B. Compared to the two orbits ranked above it, the low altitude circular orbit has no advantages and, as indicated in Table 1-3, it has numerous disadvantages. It was included in this study because it offers advantages for second or third generation solar thermionic flight experiments where the effects of orbital thermal cycling may be a major objective. This type of orbit would be of particular importance for a solar thermionic experiment involving thermal energy storage.

1.4 LAUNCH VEHICLES

The following were considered as potential launch vehicles for a solar thermionic flight experiment.

ATLAS D	THOR AGENA D
ATLAS AGENA B	TAT AGENA D (TAT-THRUST AUGMENTED THOR)
ATLAS AGENA D	DELTA
SCOUT	TAD (THRUST AUGMENTED DELTA)
THOR ABLESTAR	IMPROVED DELTA
THOR AGENA B	

From this list one of the IMPROVED DELTA launch vehicle configurations is recommended for each of thr three selected missions.

The basic IMPROVED DELTA configuration is shown in Figure 1-2. The IMPROVED DELTA, designated the DSV-3E, is presently under development. This vehicle is scheduled to be available in third quarter of 1965 and is basically an augmented DELTA (DSV-3C) with a larger improved second stage and larger payload fairings. The first stage is a modified thrust augmented DELTA (TAD) booster. The second stage vehicle consists of two major components; the AJ10-118E liquid propellant propulsion system and the guidance compartment structure. The third stage/spacecraft fairing attaches to the forward face of the guidance compartment structure and protects the third stage motor and spacecraft from aerodynamic heating during the boost flight.

The first stage liquid propellant booster is powered by a gimballed main engine and initially augmented by three externally mounted solid propellant motors equally spaced around the perihphery of the stage. The three solid propellant motors are jettisoned simultaneously after solid propellant burnout. The AJ10-118E second stage pressure fed propulsion system utilizes IRFNA and UDMH as propellants. The thrust chamber assembly is mounted on a gimbal system for attitude control (pitch and yaw) during powered flight. Roll control during

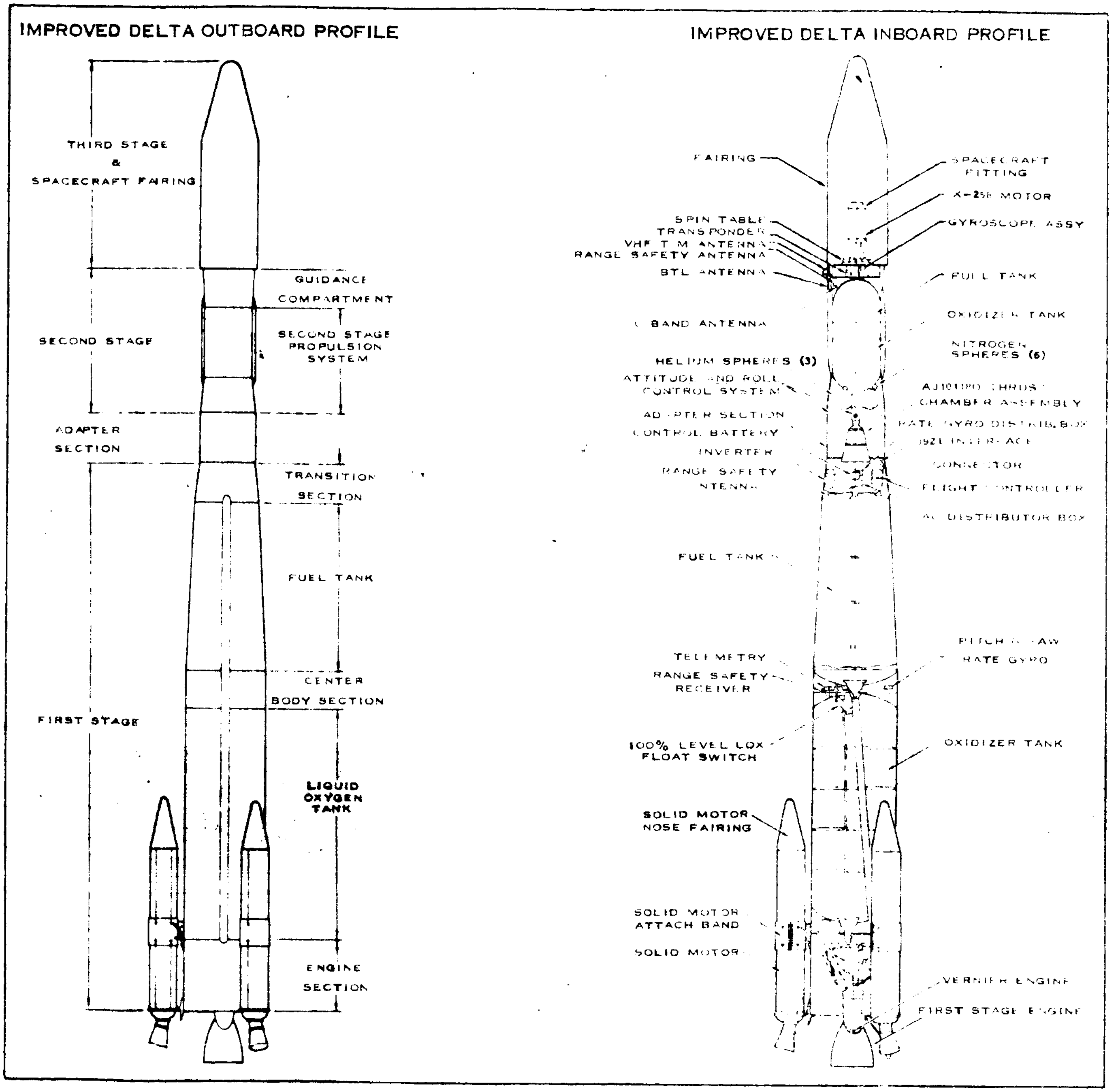


Figure 1-2. Improved Delta Launch Vehicle

powered and coast flight and pitch and yaw control during coast flight is achieved by the second stage cold gas system. The second stage guidance compartment structure houses the flight control system, the radio guidance system, the velocity cutoff system, instrumentation, range safety system, tracking and power systems.

The primary differences between the IMPROVED DELTA second stage propulsion system (AJ10-118E) and the present DELTA second stage propulsion system (AJ10-118D) are in the tankage and aft skirt assemblies. The tankage diameter has been increased from 32 inches to 54.7 inches, thus more than doubling the propellant capacity. The aft skirt has been revised to attach to the larger diameter tankage. The second-to-third stage interstage structure includes the spin table and is mounted at the forward end of the guidance compartment structure and supports the third stage vehicle. The third stage propulsion system is the ABL X258-E2 solid propellant rocket motor which is also used on the standard DELTA (DSV-3C). An attachment, from which the spacecraft is supported, is mounted to the forward shoulder of the third stage motor. The spacecraft attachment fittings include provisions for separating the spacecraft, at the spacecraft-support structure interface, from the expended third stage motor.

The larger diameter of the IMPROVED DELTA second stage allows larger diameter payload fairings than were possible with the standard DELTA (DSV-3C). The IMPROVED DELTA uses the standard AGENA D fairings which greatly increases its payload volume capability over the DSV-3C.

The IMPROVED DELTA represents a very versatile launch vehicle. It can be used in the configuration described above which constitutes a three stage vehicle with first stage augmentation. If the first stage augmentation is not needed the DSV-3E can be employed as a non-augmented three stage vehicle. Finally, for applications where an even smaller payload-orbit capability is required the third stage solid motor can be omitted and only the first two stages used. These various combinations have the following designations:

a. DSV-3E - Three stage vehicle with first stage augmentation
b. DSV-3F - Three stage vehicle without augmentation
c. DSV-3G - Two stage vehicle (omit X-258 solid) with first stage augmentation
d. DSV-3H - Two stage vehicle without augmentation

The IMPROVED DELTA then can be used as a two or three stage launch vehicle and with or without augmentation depending on the particular requirements. The recommended versions of the IMPROVED DELTA for the three selected missions are:

a. Mission A - IMPROVED DELTA DSV-3E
b. Mission B - IMPROVED DELTA DSV-3E (First stage floxed 30 percent)
c. Mission C - IMPROVED DELTA DSV-3H

The reasons for selecting a DELTA class vehicle are summarized below:

a. Based on actual flight records, the DELTA class of launch vehicles have demonstrated a significantly higher reliability than any of the other vehicles considered.
b. For equivalent payload-orbit capability, the DELTA series has the lowest cost of the vehicles considered.
c. The DELTA series of launch vehicles provide a wide range of payload-orbit capability. This minimizes the chances of paying for booster capability that is not needed.
d. The DELTA series were the only launch vehicles considered that are not engaged in orbiting classified military payloads. In addition, the DELTA series are NASA vehicles. These two conditions should minimize priority problems in obtaining a vehicle and scheduling launches.

The specific reasons for choosing the IMPROVED DELTA from the various DELTA configurations available are:

a. The payload fairings associated with the IMPROVED DELTA allow solar concentrator diameters essentially as large as those obtained with any of the vehicles considered and substantially larger than allowable with other vehicles in the DELTA series. These allowable concentrator diameters are large enough (57 inches maximum) to obtain the minimum acceptable thermionic generator output of 100 watts, with a sizable margin of safety (a concentrator diameter of 42 inches is required to obtain 100 watts). Having this safety margin simplified the spacecraft design and also permitted the use of a solar concentrator diameter of 50 inches, which is required if thermionic emitter areas of two square centimeters are to be used. Areas of two square centimeters are highly desirable since all of the existing experience has been with emitter areas of this size.

b. Since the IMPROVED DELTA can be used in four different configurations (DSV-3E, 3F, 3G and 3H), which cover a wide range of payload-orbit capability, it represents a very versatile launch vehicle. For example, assume a mission were designed around the DSV-3H vehicle as is the case with Mission C. If for some reason, such as a change in the planned orbit, an overweight spacecraft, etc., greater booster capability were required a switch to the DSV-3E launch vehicle could be made. Since all of the IMPROVED DELTA configurations are modifications of the same basic vehicle, and they all employ the same payload fairings, this change could be made with the minimum of difficulty. This might prove particularly attractive for second or third generation missions where a significant change in the spacecraft orbit could be made without a major redesign of the spacecraft.

There are three potential disadvantages with using the IMPROVED DELTA launch vehicle. First, since it is not scheduled to become operational until the third quarter of 1965, it is not now operational and therefore its performance has not been demonstrated. However, if the present schedule is held, the vehicle availability should not present a problem for a solar thermionic flight experiment. Also, this is not an entirely new vehicle since, in general, the IMPROVED DELTA will employ the same basic components as the current DELTA (DSV-3C), and is therefore expected to have the same high reliability.

Secondly, the DELTA class of vehicles does not at present have Western Test Range launch capability which is advantageous for achieving polar or near polar orbits. This capability is scheduled to be available in the first quarter of 1966 and, therefore, should present no problems for this program.

Finally, the DELTA series of vehicles have, in general, a more severe launch acceleration and vibration environment because of the solid stages employed. A detailed design analysis will have to be performed before the effects of the launch environment can be established, but based on past spacecraft design experience, no extremely difficult problems are anticipated.

These three potential disadvantages are not considered serious at this point.

1.5 SPACECRAFT DESIGN

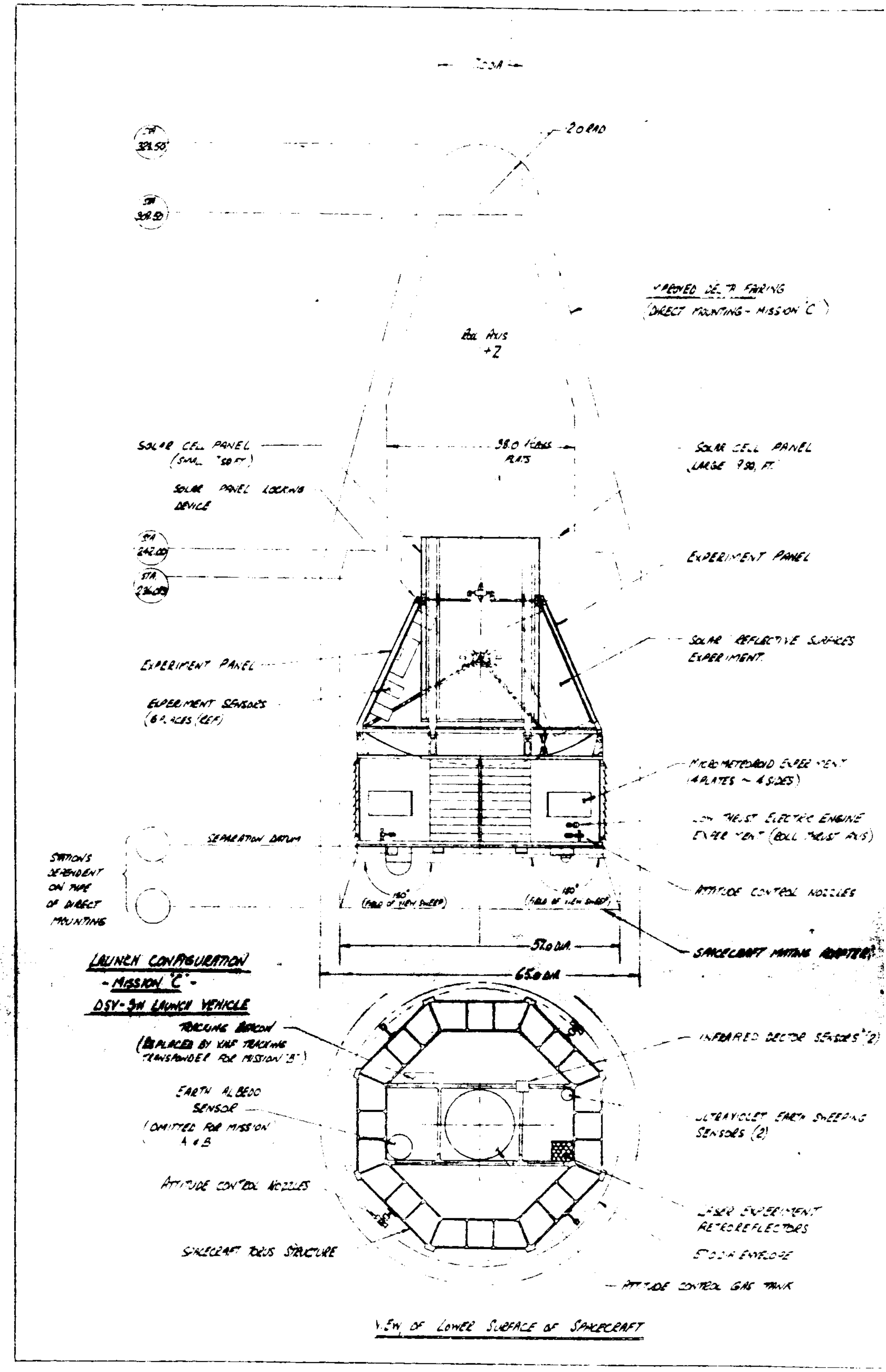
The proposed spacecraft design is shown in Figure 1-3. This same basic design is used for Missions A, B and C, with the minor component and secondary experiment changes required between missions indicated on the drawing. A detailed spacecraft weight breakdown for Missions A, B and C is given in Table 1-4. The Mission A, B and C spacecraft weights are 360, 373 and 404 pound respectively.

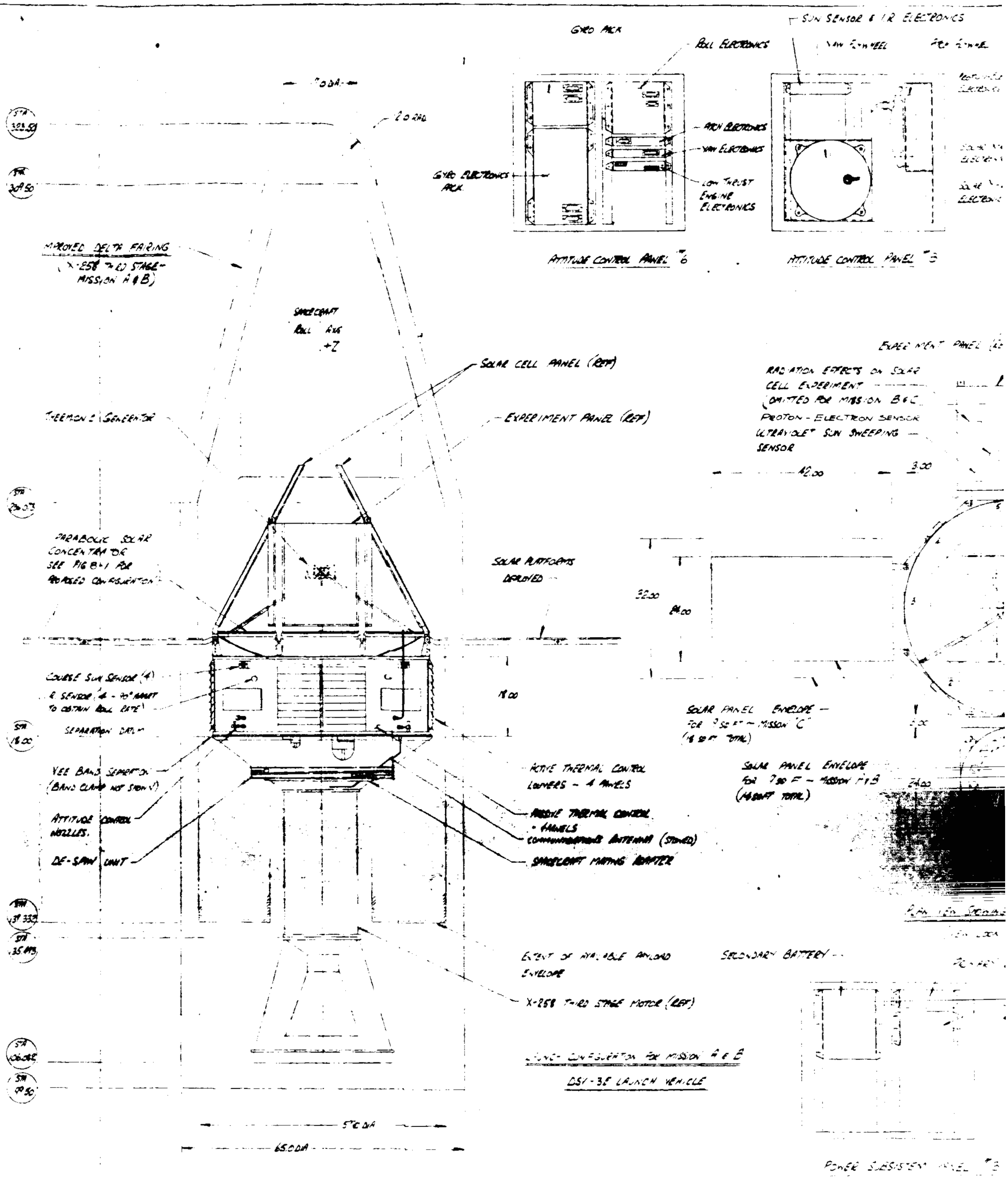
A description of the spacecraft configuration and the major subsystems is included in the following sections.

1.5.1 CONFIGURATION

The spacecraft body is an eight-sided prism with flat ends. The components are mounted in modular fashion on the eight side panels (see panel layouts Figure 1-3). As far as possible all of the components comprising a specific subsystem are located on the same panel. This simplifies assembly and checkout of the spacecraft. In order to obtain the desired center of gravity location and balance the moments of inertia, it was not possible to follow this rule in all cases. Four of the eight spacecraft sides have active thermal control. The remaining four sides are used for mounting the attitude control nozzles, the telemetry antenna, and hardware associated with the secondary experiments.

The lower octagonal face of the vehicle body is used for mounting the experiments and sensors that must see earth. This end of the spacecraft mates with the truncated conical adapter. The upper octagonal face of the vehicle body is used for mounting the solar thermionic system. Four deployable panels, hinged off this upper face, are used for mounting solar cells and the sun-pointing secondary experiments. The spacecraft body is built from two octagonal machined frames joined by vertical members at the eight corners. Hinge fittings for the four deployable panels and support fittings for the solar thermionic system are bolted to the upper machined frame.


20 RAD
IMPROVED DELTA FAIRING
(DIRECT MOUNTING - MISSION "C")
ROLL AXIS
+Z
SOLAR CELL PANEL
(SMALL 50 FT)
38.0 ACROSS FLATS
SOLAR CELL PANEL
LARGE 9.50 FT.
SOLAR PANEL LOCKING DEVICE
EXPERIMENT PANEL
EXPERIMENT PANEL
SOLAR REFLECTIVE SURFACES EXPERIMENT
EXPERIMENT SENSORS
(6 PLACES (REF)
MICROMETEOROID EXPERIMENT
(4 PLATES ~ 4 SIDES)
LOW THRUST ELECTRIC ENGINE
EXPERIMENT (ROLL THRUST AXIS)
SEPARATION DATUM
STATIONS DEPENDENT ON TYPE OF DIRECT MOUNTING
ATTITUDE CONTROL NOZZLES
180° (FIELD OF VIEW SWEEP)
180° (FIELD OF VIEW SWEEP)
57.0 DIA.
SPACECRAFT MATING ADAPTER
65.0 DIA.
LAUNCH CONFIGURATION
- MISSION "C" -
DSV-3H LAUNCH VEHICLE
TRACKING BEACON
(REPLACED BY VHF TRACKING TRANSPONDER FOR MISSION "B")
INFRARED DETECTOR SENSORS (2)
EARTH ALBEDO SENSOR
(OMITTED FOR MISSION A & B)
ULTRAVIOLET EARTH SWEEPING SENSORS (2)
ATTITUDE CONTROL NOZZLES
LASER EXPERIMENT RETROREFLECTORS
SPACECRAFT TORUS STRUCTURE
ATTITUDE CONTROL GAS TANK
VIEW OF LOWER SURFACE OF SPACECRAFT


SUN SENSOR & I.R. ELECTRONICS
GYRO PACK
ROLL ELECTRONICS
PITCH ELECTRONICS
YAW ELECTRONICS
GYRO ELECTRONICS PACK
LOW THRUST ENGINE ELECTRONICS
ATTITUDE CONTROL PANEL #6
ATTITUDE CONTROL PANEL #3
IMPROVED DELTA FAIRING
(X-258 THIRD STAGE - MISSION A & B)
SPACECRAFT ROLL AXIS +Z
SOLAR CELL PANEL (REF)
EXPERIMENT PANEL (REF)
RADIATION EFFECTS ON SOLAR CELL EXPERIMENT
(OMITTED FOR MISSION B & C)
PROTON - ELECTRON SENSOR
ULTRAVIOLET SUN SWEEPING SENSOR
PARABOLIC SOLAR CONCENTRATOR
SEE FIG B-1 FOR PROPOSED CONFIGURATION
SOLAR PLATFORMS DEPLOYED
COURSE SUN SENSOR (4)
SEPARATION DATUM
VEE BAND SEPARATION
(BAND CLAMP NOT SHOWN)
ATTITUDE CONTROL NOZZLES
DE-SPIN UNIT
ACTIVE THERMAL CONTROL LOUVERS - 4 PANELS
PASSIVE THERMAL CONTROL PANELS
COMMUNICATIONS ANTENNA (STOWED)
SPACECRAFT MATING ADAPTER
SOLAR PANEL ENVELOPE
SECONDARY BATTERY
X-258 THIRD STAGE MOTOR (REF)
DSV-3E LAUNCH VEHICLE
65.00 DIA
POWER SUBSYSTEM PANEL #3

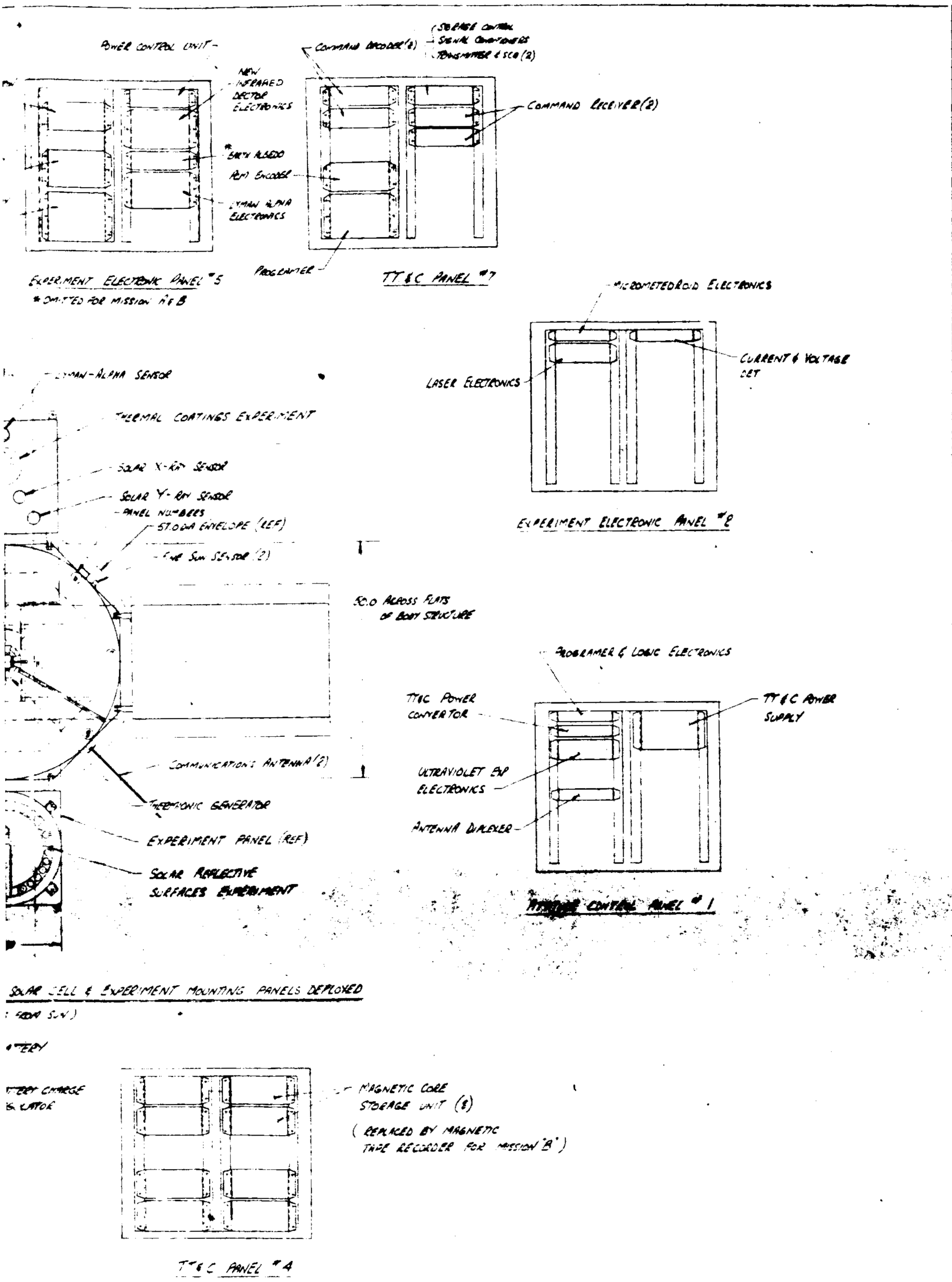


Figure 1-3. Spacecraft Configuration Missions A, B and C

Table 1-4. Spacecraft Weight Breakdown

Subsystem and Component	Mission A Weight (Lb)	Mission B Weight (Lb)	Mission C Weight (Lb)
ATTITUDE CONTROL	(65.9)	(59.2)	(75.5)
Momentum Flywheel (2)	11.0	9.0	17.0
Roll, Pitch, Yaw Gyros and Electronics	11.0	11.0	11.0
Power Supply (AC and DC)	11.5	11.5	11.5
Sun Sensors and Electronics	2.0	2.0	2.0
Programmer and Logic	2.5	2.5	2.5
Attitude Control Electronics (3)	7.0	7.0	7.0
IR Sensors and Electronics	1.0	1.0	1.0
Thermal Sensors	0.2	0.2	0.2
Pneumatic System	19.7	15.0	23.3
Gas (Nitrogen)	3.0	0.9	4.5
Tank	4.1	1.5	6.2
Fill and Check Valves and Filters	2.2	2.2	2.2
Pressure Transducer (2)	0.8	0.8	0.8
Regulator	2.5	2.5	2.5
Solenoids (7)	2.8	2.8	2.8
Nozzles and Orifices (8)	1.0	1.0	1.0
Relief Valve	.8	.8	.8
Lines, Fittings, Clips, Bracketry	2.5	2.5	2.5
POWER SUPPLY	(34.4)	(49.8)	(55.6)
Primary Battery	1.5	1.8	1.5
Secondary Battery	12.6	27.8	28.5
Charge Regulator	3.1	3.6	4.5
Power Control Unit	0.8	0.9	1.2
Solar Array Harness	0.8	0.8	1.0
Inflight Disconnect	0.5	0.5	0.5
Solar Array	15.1	14.4	18.4
Structure and Restraints	6.0	6.2	7.9
Solar Cells	9.1	8.2	10.5

Table 1-4. (Cont.) Spacecraft Weight Breakdown

Subsystem and Components	Mission A Weight (Lb)	Mission B Weight (Lb)	Mission C Weight (Lb)
TELEMETRY, TRACKING AND COMMAND	(43.8)	(46.8)	(48.8)
Antenna and Connectors	1.0	1.0	1.0
Diplexer (Antenna)	1.0	----	1.0
Command Receiver (2)	3.0	3.0	3.0
Command Recoder (2)	5.4	5.4	5.4
Magnetic Core Storage Units (2.5 lb ea.)	20.0	---	25.0
Storage Control Circuitry	1.0	---	1.0
PCM Encoder	3.0	3.0	3.0
TT&C Power Converter	2.0	2.0	2.0
Signal Conditioners	1.0	1.0	1.0
Transmitter and SCO (2)	1.4	1.4	1.4
Tracking Beacon	1.0	---	1.0
Programmer	2.0	2.0	2.0
Multiplexer (Antenna)	---	1.0	---
Magnetic Tape Recorder	---	20.0	---
Tracking Transponder	---	5.0	---
Coax Cabling	2.0	2.0	2.0
SECONDARY EXPERIMENTS	(82.8)	(82.3)	(87.3)
Solar Cell Radiation Effect	0.5	---	---
Thermal Coatings	2.0	2.0	2.0
Solar Concentrator Reflective Surfaces	27.0	27.0	27.0
Infrared Detector	3.0	3.0	3.0
Laser Experiment	10.0	10.0	10.0
Low Thrust Electric Engine	2.3	2.3	2.3
Micrometeoroid Detection	8.0	8.0	8.0
Proton and Electron Spectra and Direction	8.0	8.0	8.0
Solar X-rays	3.0	3.0	3.0
Solar γ Rays	8.0	8.0	8.0
Earth Albedo	---	---	5.0
Ultraviolet Radiation	5.0	5.0	5.0
Lyman-Alpha	6.0	6.0	6.0

Table 1-4. (Cont.) Spacecraft Weight Breakdown

Subsystem and Components	Mission A Weight (Lb)	Mission B Weight (Lb)	Mission C Weight (Lb)
PRIMARY THERMIONIC EXPERIMENT	(38.2)	(38.2)	(38.2)
Generator	5.7	5.7	5.7
Parabolic Concentrator Surface	9.8	9.8	9.8
Generator Mounting Ring	0.3	0.3	0.3
Truss Tubes (3)	0.7	0.7	0.7
Tube Fittings (3)	0.1	0.1	0.1
Support Torus	2.2	2.2	2.2
Support Link (3)	0.1	0.1	0.1
Yoke (Includes Bearings) (3)	0.3	0.3	0.3
Support Fitting (Includes Bearings) (3)	0.3	0.3	0.3
Tie-In Ring	0.2	0.2	0.2
RTV Bond	0.5	0.5	0.5
Control Electronics	6.0	6.0	6.0
Generator Electrical Leads	7.0	7.0	7.0
Reflectometer	5.0	5.0	5.0
DIAGNOSTIC INSTRUMENTATION	(2.8)	(2.8)	(2.8)
Current Detectors	1.1	1.1	1.1
Voltage Detectors	0.5	0.5	0.5
Temperature Sensors	1.2	1.2	1.2
THERMAL CONTROL	(22.0)	(22.0)	(22.0)
Active Thermal Control (12 ft^2)	15.0	15.0	15.0
Insulation (4.7 lb/ft^3 - 1/2 in thick, 0.20 lb/ft^2)	5.4	5.4	5.4
Paint and Grease	0.6	0.6	0.6
Heaters	0.5	0.5	0.5
Misc	0.5	0.5	0.5
HARNESSING	(28.4)	(30.4)	(32.0)

Table 1-4. (Cont.) Spacecraft Weight Breakdown

Subsystem and Components	Mission A Weight (Lb)	Mission B Weight (Lb)	Mission C Weight (Lb)
STRUCTURE	(41.8)	(41.8)	(41.8)
Side Panels (Honeycomb) (8)	9.6	9.6	9.6
Top and Bottom Fill Panels	4.8	4.8	4.8
L's, T's (Framing)	9.2	9.2	9.2
Bulkheads (Internal)	3.7	3.7	3.7
Tank Support Structure and Fittings	1.6	1.6	1.6
Gussets and Misc. Structure	0.3	0.3	0.3
GSE Fittings	0.8	0.8	0.8
Experiment Panels and Hinges (2)	3.3	3.3	3.3
Solar Platform Locks	0.8	0.8	0.8
Separation Ring and Band	3.9	3.9	3.9
Hardware	3.8	3.8	3.8
TOTAL SPACECRAFT WEIGHT	360.1	373.3	404.0

RE-ORDER No. 65 - 323

The octagonal ends of the body are partially closed by light honeycomb panels. Four faces of the body have temperature control louvers mounted to the exterior faces. The remaining four faces are covered with a multilayer superinsulation, except for the cut-outs required for sensors, attitude control nozzles, etc.

The attitude control gas tank and associated hardware is mounted on a trusswork in the center of the vehicle body. This location is chosen to minimize the center of gravity shift with gas usage.

All machined parts of the body are magnesium, and all body panels are aluminum.

As shown in Figure 1-3 the solar array consists of two rectangular hinged panels providing approximately 14 square feet of solar array area for Missions A and B and 18 square feet for Mission C. The solar cells are mounted on aluminum channels, each channel constituting a solar array module. These channel sections are supported and stiffened by two sheet metal members running lengthwise, at 90 degrees to the longitudinal axis of the submodule channels, and terminating in magnesium hinge fittings.

The two solar array panels and the two secondary experiment panels are folded upward and attached to each other to form a rigid structure during the launch phase. By means of a linkage system, only two pyrotechnic devices, one on each solar cell panel, are required to withdraw the four locking pins. The four panels are deployed independently by means of springs at their hinge points and lock positively in place upon reaching the deployed position.

The smaller secondary experiment panels are constructed in the same manner as the solar array panels. However, in general heavier gage materials would be used because of the greater weight of the experiments they must support.

The proposed active thermal control system* is essentially the same scheme used on the Mariner C spacecraft. The high heat dissipating equipment is mounted on the faces of the spacecraft body that have active thermal control capability and reject their waste heat to space by radiation. The amount of heat rejected is controlled by variable louvers attached to the outside of the panel. Each louvered panel is controlled independently by a bi-metal spiral spring that winds up and unwinds according to the temperature it sees, thus opening or closing the louvers. The surface area exposed by opening the louvers is coated with a high emissivity coating to promote radiation heat transfer.

In addition to the active thermal control system the spacecraft thermal balance is maintained by the use of superinsulation and coatings on the other exposed surfaces.

Thermal control of temperature sensitive equipment mounted outside the spacecraft body is achieved passively by the use of thermal coatings and insulation.

Two different truncated conical adapters are shown in Figure 1-3. Which one of these adapters is used depends on whether the IMPROVED DELTA is used with or without the X-258 third stage. The spacecraft release and separation system is similar for both cases. The lower octagonal frame of the spacecraft body is machined to have a circular flange approximately 55 inches in diameter. This flange forms one-half of a marman-clamp attachment and matches a similar ring on the adapter.

The two flanges are clamped together by a series of vee-shaped shoes and a band clamp. The band clamp would be split at separation by a bolt cutter or similar pyrotechnic device, and the spacecraft would separate under the action of four matched springs located at four of the eight corners of the vehicle body.

---

*This needs to be investigated in detail. If the required spacecraft thermal environment could be provided by a passive system the spacecraft design could be simplified slightly and the weight probably reduced.

The adapter for direct mounting to the IMPROVED DELTA second stage is a simple sheet metal truncated cone that grows from the 55-inch diameter at the spacecraft, to approximately the 60-inch diameter that is necessary to mate with the DSV-3H's second stage.

The adapter for the X-258 third stage decreases from the 55-inch diameter at the spacecraft to the 18-inch diameter necessary to mate with the X-258. The yo-yo despin system, required with the solid X-258 motor, is mounted on the adapter. Other equipment, such as the timer, squib battery, terminal boards, and balance weights are also mounted on the adapter. The balance weights are necessary, since in this configuration the spacecraft/X-258 assembly must be balanced to achieve the close center of gravity and principal axes alignments required as a result of having to "spin-up" this assembly on the IMPROVED DELTA second stage.

1.5.2 SOLAR THERMIONIC EXPERIMENT DESIGN

The solar thermionic system configuration is shown in Figure 1-4 and the performance parameters are summarized in Table 1-5. The system performance would be evaluated by means of the instrumentation outlined in Table 1-6.

Once the spacecraft was in orbit, the solar thermionic experiment would operate as described below.

Under normal conditions the four thermionic converters would be connected in series and the load adjusted so that at design temperatures the converters would operate at their peak efficiency point.

When the spacecraft emerged from the earth's shadow* the attitude control subsystem would orient the solar concentrator to the sun with the desired $\pm 0.1^{\circ}$ accuracy. To accomplish this would take approximately 2 minutes. No auxiliary heat would be used to warm up the

*In the Mission A case, the spacecraft would not encounter any shadow periods for the first four to six months of the experiment.

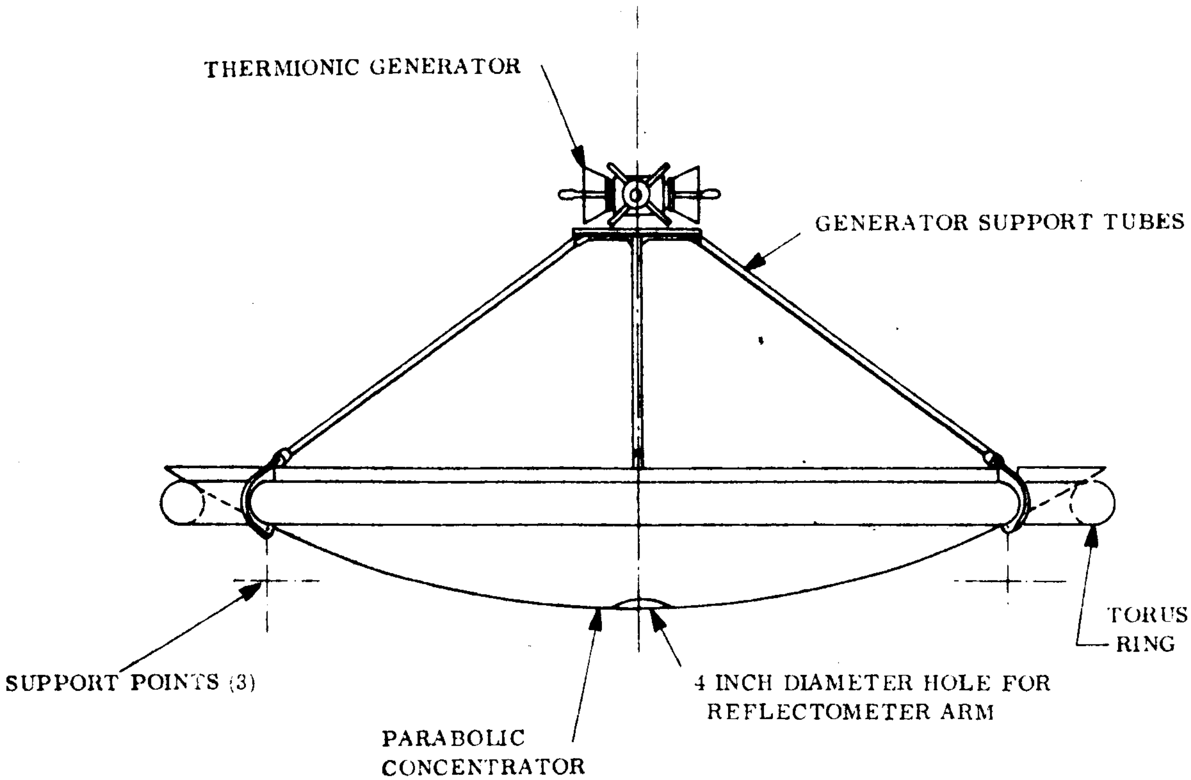


Figure 1-4. Solar Thermionic System

cesium reservoirs and no controls would be employed to limit the solar concentrator input to the generator cavity. As a result of the latter the emitter temperatures would reach approximately 2400°K for a short period of time. In an estimated 16 minutes after acquiring the sun the thermionic generator would reach the design operating point and steady state operation would follow.

During steady state operation the cesium reservoir temperature would be adjusted to the optimum value by small electrical heaters (0.1 watt maximum capacity).

The heater control circuit would be designed so that it would attempt to maximize the converter power output for the given load and emitter temperature. During the daylight portion of the orbit, the temperatures and voltages outlined in Table 1-6 would be monitored, the data stored, and when the spacecraft was over an appropriate ground station telemetered back to earth. The desired frequency of sampling the measured quantities requires reading

Table 1-5. Thermionic System Performance Parameters

PERFORMANCE PARAMETERS	
Converter Emitter Temperature	2000°K
Converter Electrode Spacing	2 Mils
Converter Emitter Material	Rhenium
Converter Sleeve Thickness	0.007 Inches
Converter Emitter Area	2 cm^2
Converter Operating Point	Peak Efficiency
Converter Power Density	18 Watts/cm^2
Converter Voltage Output	0.85 Volts
Converter Efficiency	17.4 Percent
Concentrator Diameter	50 Inches
Concentrator Rim Angle	60 Degrees
Concentrator Geometric Error, 3σ	12 Minutes
Concentrator Reflectivity	90 Percent
Concentrator Blockage Factor	5 Percent
Concentrator-Absorber Efficiency	67 Percent
Thermionic Generator Aperture Diameter	0.71 Inches
Thermionic Generator Efficiency	12.2 Percent
Thermionic Generator Power Output	144 Watts
Thermionic Generator Voltage (Four Converters in Series)	3.4 Volts
Thermionic System Efficiency	8.2 Percent

Table 1-6. Primary Experiment Instrumentation[a]

Measurement	Number of Measurements		Range of Output Signal	Comments
	Voltage	Temp		
A - Converter Performance				
Voltage Output Each Converter	4		0-1v	
Voltage Drop Across Load	1		0-4v	
Emitter Temperature Each Converter		4	0-35 mv	W-5 Re/W-25 Re Thermocouples
Collector Temperature Each Converter		4	0-35 mv	Chromel-Alumel Thermocouples
Fin Temperature Each Converter		4	0-30 mv	" " "
Seal Temperature Each Converter		4	0-35 mv	" " "
Cesium Reservoir Temperature Each Converter		4	0-15 mv	" " "
B - Generator Structure				
Temperature Generator Back Cover		2	0-30 mv	Cromel-Alumel Thermocouples
Temperature Outer Edge of Shield Cone		2	0-35 mv	W-5 Re/W-25 Re Thermocouples
Temperature Drop on One Generator Block Support		2	0-30 mv	Cromel-Alumel Thermocouples
C - Concentrator Performance				Measurements along two radial lines 90° apart made with resistance thermometers.
Temperature Distribution on Concentrator Surface		10	0-3v	
Concentrator Reflectivity	1		0-5v	
D - Orientation System Performance				
Thermocouples Located 90 degrees Apart Around the Aperature Opening		4	0-35 mv	W-5 Re/W-25 Re Thermocouples
Two Fine Sun Sensors (Pitch and Yaw)	2		0-5v	
Two Pair of Course Sun Sensors (Pitch and Yaw)	2		0-5v	

Notes.

a - A telemetry accuracy of one percent is adequate for all of the instrumentation in this table.

65-323 2 of 2

each quantity every 15 seconds during the transient warm-up phase (approximately 16 minutes) and then once every five minutes throughout the remainder of the daylight period. The steady state mode of operation would continue until the spacecraft entered the earth's shadow.

Upon entering the earth's shadow the solar input to the generator would stop and the emitter temperature would start to drop. As the emitter temperature dropped the generator power output would decrease until at an emitter temperature of approximately 1750^{o}K the generator power output would become zero (this assumes the cesium reservoir heaters are able to optimize the reservoir temperature as the emitter temperature decreases and that the design point load is maintained). The instrumentation outlined in Table 1-6 would be sampled every 15 seconds during the initial 16 minutes (approximate) of the shadow period and every five minutes through the remainder of the dark cycle.

At periodic intervals a generator EI curve would be measured by varying the generator load resistance. This would be accomplished by ground command. In order to keep a check on the individual converters performance they would periodically be commanded into series with their individual load resistors and the resulting performance recorded.

If a thermionic converter were to fail it would be removed from the series circuit by ground command and the experiment continued with the remaining converters. If the thermionic generator performance were to degrade, the operating point could be altered by changing the load resistance. This would also be achieved by ground command.

In general, commanded changes in the generator circuit would be held to a minimum for reliability reasons.

The proposed solar thermionic experiment design would provide the data necessary to conduct a complete performance evaluation of the individual thermionic converters as well as the thermionic generator, solar concentrator and attitude control subsystems.

1.5.3 SECONDARY EXPERIMENTS

In addition to the primary solar thermionic experiment, numerous secondary experiments were considered and a recommended list established for each of the selected missions. The major factors considered in choosing the proposed secondary experiments were:

a. Compatibility with the solar thermionic experiment
b. Importance to the development of solar thermionics
c. Importance to the space program and interest to the scientific community
d. Ease of integration with the spacecraft
e. Size and weight
f. Demands on the telemetry, command and power subsystems.

Based on consideration of the above, the recommended secondary experiments are listed in Table 1-7, along with the major reasons for their selection.

The secondary experiments considered in this study are illustrative of those that could be included on a spacecraft whose primary objective is the evaluation of solar thermionics. As subsequent studies of the solar thermionic flight experiment are undertaken, it will be necessary to continually evaluate potential secondary experiments to insure that those finally selected represent the best possible choice. It will also be necessary to evaluate these experiments in greater detail than this study would allow before a final selection is made.

1.5.4 TELEMETRY, TRACKING AND COMMAND SUBSYSTEM

Ground traces for the first 24 hours after injection for Missions A and C are shown in Figures 1-5 and 1-6 respectively. Superimposed on these traces are the coverage circles for appropriate ground stations, within which the line-of-sight to the satellite is at least 15 degrees above the horizon. Figure 1-7 shows the ground trace during the first 40 hours for the Mission B elliptical orbit. Slant ranges at the 15 degree elevation angle limits and total

Table 1-7. Recommended Secondary Experiments

Secondary Experiment		Major Reasons for Selection	Mission Compatibility		
			A	B	C
1	Solar Reflective Surfaces	Results of major importance to the development of solar thermionics and other solar power systems employing concentrators. Also helpful in evaluating the primary experiment.	X	X	X
2	Micrometeroid Measurements	Important scientifically and to evaluation of the primary experiment. Also needed for complete evaluation of experiment No. 1	X	X	X
3	Proton and electron Spectra and Direction	Important scientifically and to evaluation of the primary experiment. Also needed for complete evaluation of experiments No. 1 and 8.	X	X	X
4	Ultraviolet Radiation	Important scientifically and to evaluation of the primary experiment. Also needed for complete evaluation of experiments No. 1, 5 and 8, and supplements experiments No. 10 and 11.	X	X	X
5	Thermal Coatings	Relatively simple experiment which yields needed information on thermal coatings to be used on space vehicles. Experiment package already developed. Experiment makes use of sun pointing feature of the spacecraft.	X	X	X
6	Laser Experiment	Important to the development of lasers as a means of space communication.	X	X	X

Table 1-7. (Cont.) Recommended Secondary Experiments

Secondary Experiment		Major Reasons for Selection	Mission Compatibility		
			A	B	C
7	Low-Thrust electric engine	Important to the development of low-thrust electric engines for use in attitude control systems and space propulsion in general.	X	X	X
8	Radiation effects on solar cells	Important to a more complete understanding of solar array degradation from radiation. Results needed to improve solar cell power supply design. Also makes use of the sun-pointing feature of the spacecraft.	X	-	-
9	Infrared Detector	Important to the development of better IR detectors for use in attitude control systems.	X	X	X
10	Solar X-Rays	Important scientifically and makes use of the sun pointing feature of the spacecraft. Also supplements experiments No. 4 and 11.	X	X	X
11	Solar γ-Rays	Important scientifically and makes use of the sun pointing feature of the spacecraft. Also supplements experiments No. 4 and 10.	X	X	X
12	Lyman-Alpha	Important scientifically and makes use of sun pointing feature of the spacecraft. Also supplements experiments No. 4, 10, and 11.	X	X	X
13	Earth Albedo	Important to a better measure of the earth's albedo. This information is of value scientifically and to the thermal design of space vehicles.	-	-	X

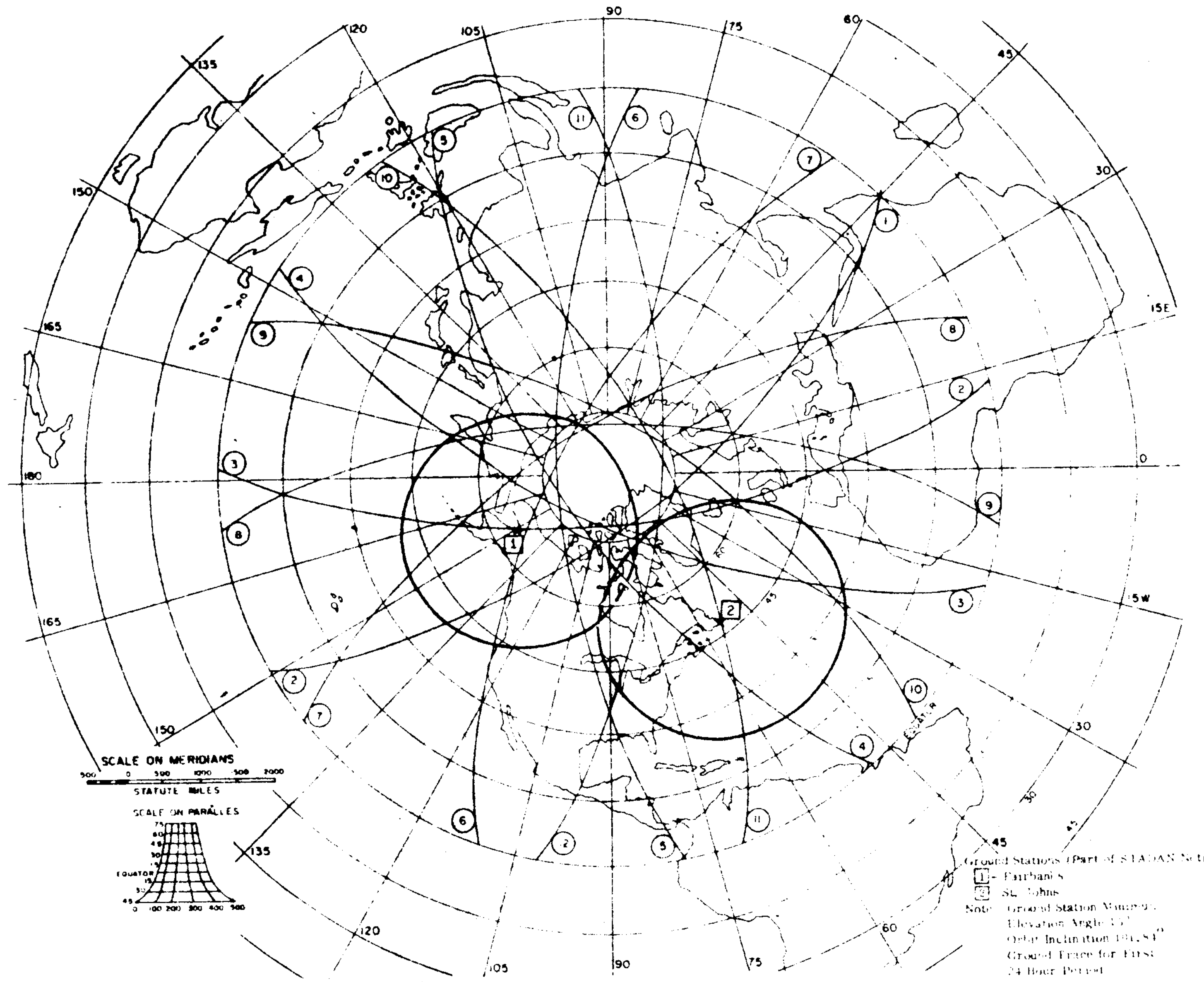


Figure 1-5. Ground Station Coverage for Mission A

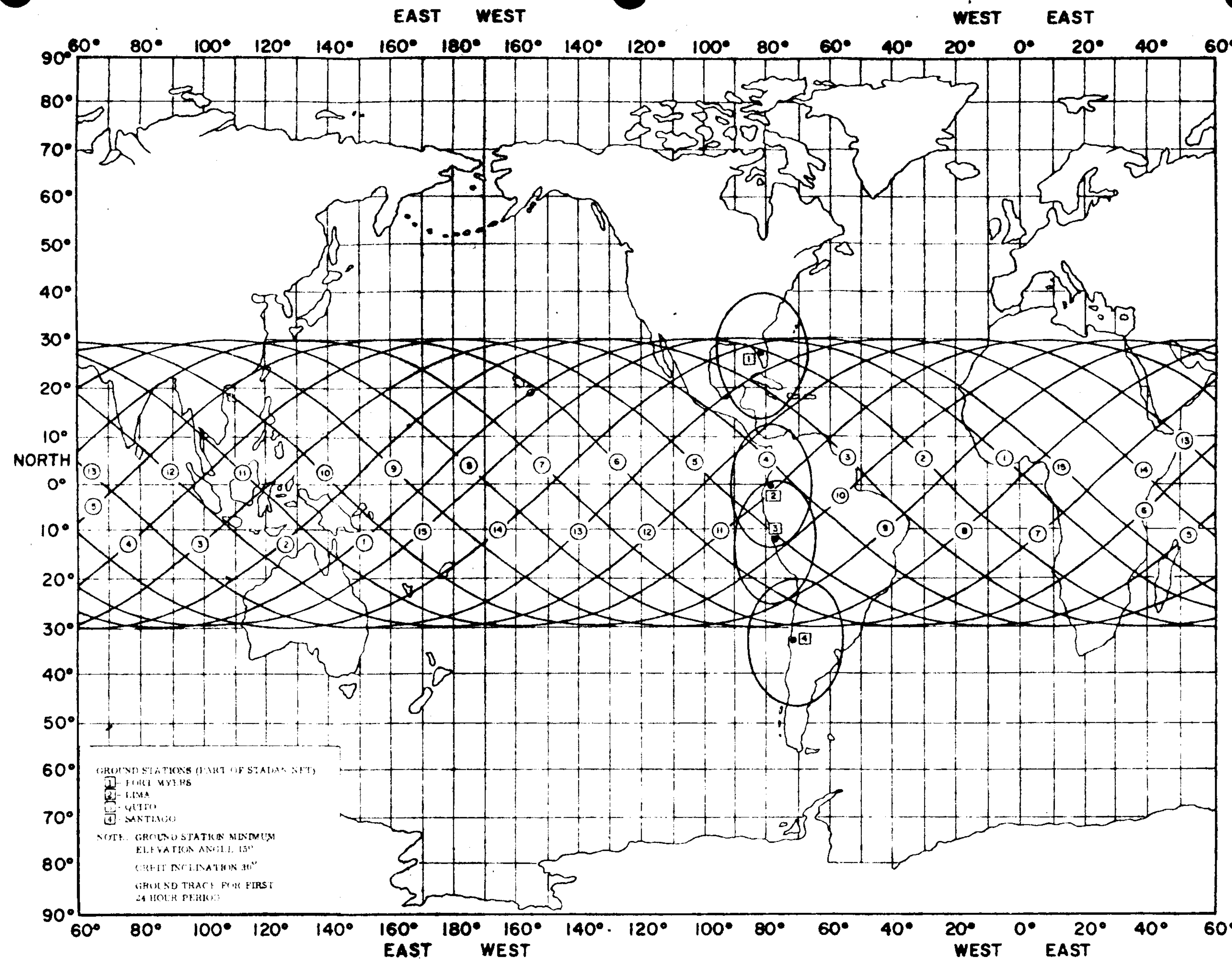


Figure 1-6. Ground Station Coverage for Mission C

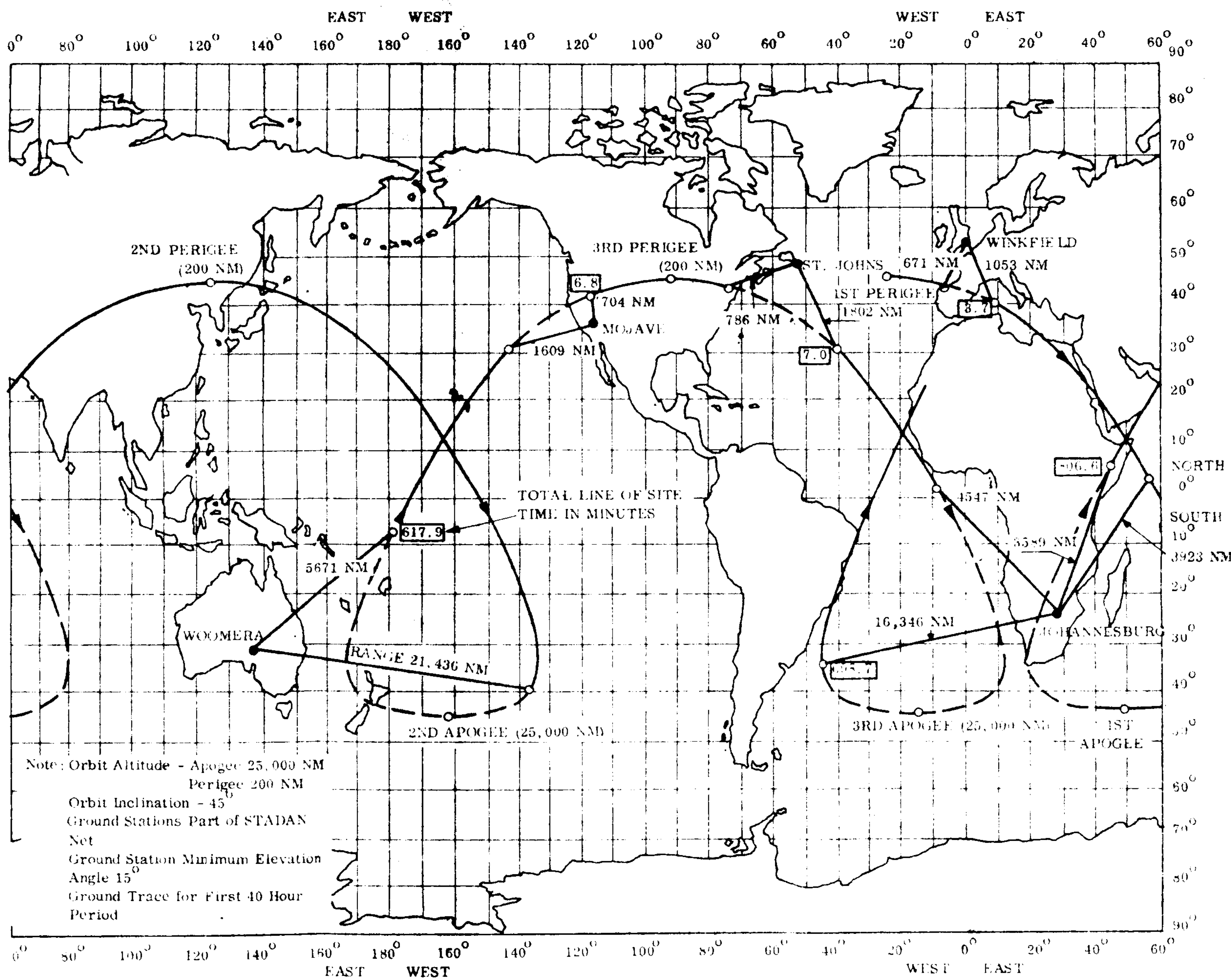


Figure 1-7. Ground Station Coverage for Mission B

line-of-sight times are indicated (coverage circles would have no significance for the elliptical orbit since the altitude is continuously changing). These figures indicate the location and number of potential ground stations required for handling the TT&C functions.

Figure 1-8 shows a functional block diagram of the TT&C subsystem for all three missions. A moderate amount of redundancy has been included. Both command receivers and both decoders are powered continuously to guard against a command link failure. The spare magnetic tape recorder for the Mission B case has been included for use in the event of premature failure of this critical component. If both recorders are operative, the spare recorder may be used to accommodate data collection for a second orbit when necessary.

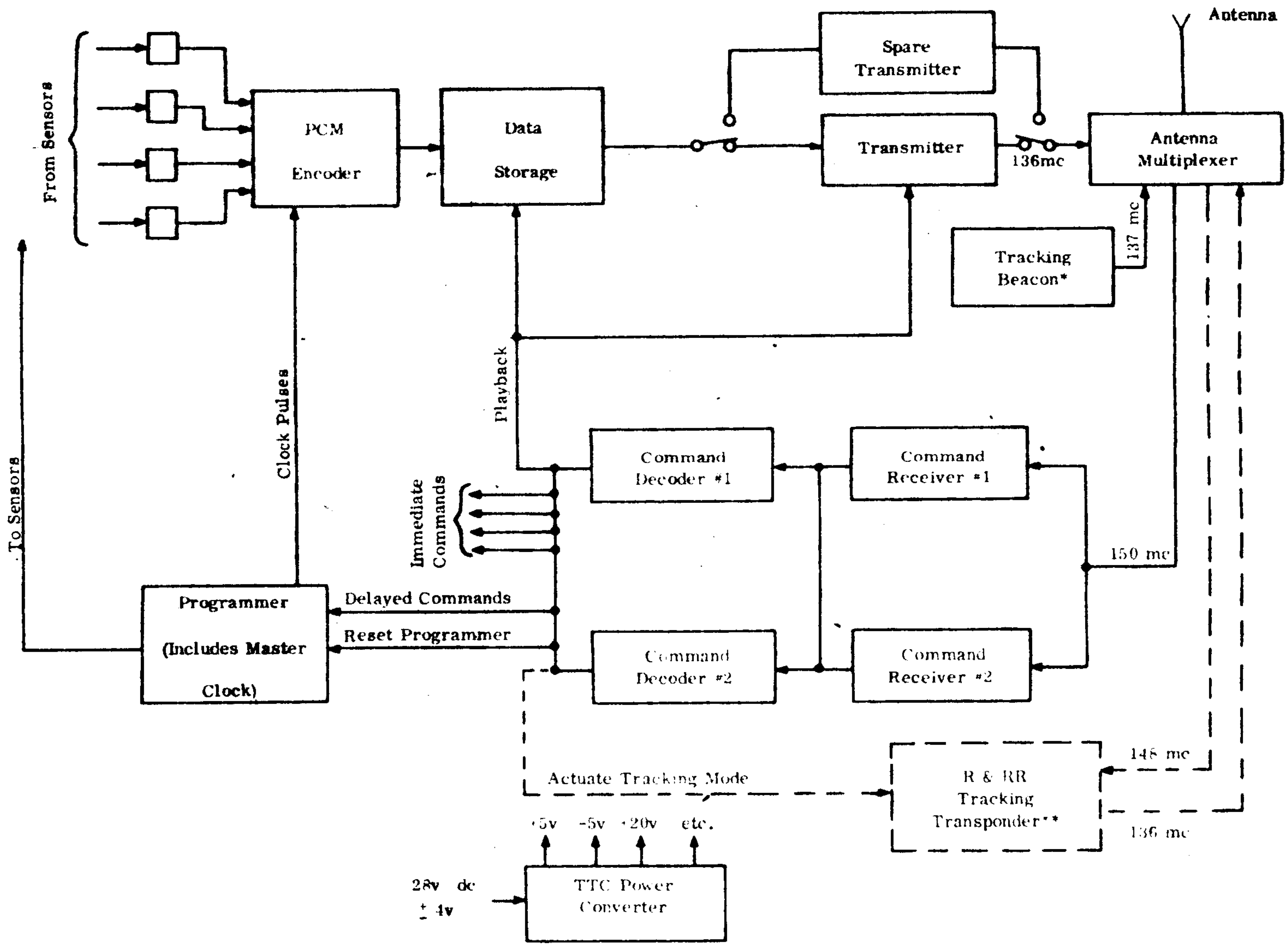
A tracking beacon has been included on the Mission A and C spacecraft for early orbit determination. It will be used only during the first few weeks of the mission. The tracking beacon can utilize the excess power available from the solar cells during this period.

A range and range-rate transponder has been included for the Mission B elliptical orbit, in order to permit an accurate ephemeris determination as soon after injection as possible. Tracking for a period of about five minutes per orbit should generally be adequate, although somewhat longer tracking periods during the first few orbits would be desirable.

The required transmitter output powers have been selected to provide the required effective radiated power through antenna nulls of -10 db.

The Mission A satellite would collect 240,000 bits of data each orbit and would store these in eight magnetic core storage units. After the orbit was well established, the tracking beacon would be turned off, and the telemetry carrier activated by command (either stored or real-time) to initiate the acquisition procedure.

Acquisition should be completed in less than two minutes, and the data would be read out upon command from either the Fairbanks or St. John's ground station at a rate of 400 bits per



*(Used only during first few weeks to assist in orbit determination.)
**(Highly desirable for Orbit II. Optional for circular orbits.)

Figure 1-8. TT&C Subsystem Block Diagram second for ten minutes, using PCM/FM/PM. Orbits where neither of these ground stations had a clear line-of-sight to the satellite would be extremely rare, and the loss of data on these orbits would be accepted.

The Mission B satellite would collect 800,000 bits of data each orbit and would store these in a magnetic tape recorder. A second tape recorder would be available as a spare and to provide storage for a second orbit of data on those occasions where a ground station was not available upon completion of an orbit. Up to two orbits of stored data would be transmitted near perigee upon command to the Mojave, St. John's, or Winkfield stations at a rate of 6680 bits per second for four minutes, using PCM/FM and noncoherrent reception. Frequency acquisition would not be necessary, and angular acquisition should be possible during the first few weeks before the solar cells have degraded and while surplus power is available.

The Mission C satellite would collect 150,000 bits of data each orbit and would store up to two orbits of data in ten magnetic core storage units. It would be acquired in the same manner as the Mission A satellite (in approximately two minutes), and the data would be read out upon command from the Fort Myers, Santiago, Lima, or Quito stations at a rate of 2500 bits per second for two minutes, using PCM/FM/PM. Occasions where it would be necessary to go for more than two orbits without reading out the data would be extremely rare, and the loss of data on these occasions would be accepted.

1.5.5 ATTITUDE CONTROL SUBSYSTEM

In the proposed design the solar thermionic system is rigidly attached to the main spacecraft body and the entire space vehicle is oriented to the sun. The solar thermionic system necessitates an orientation accuracy of ± 0.1 degree about the pitch and yaw axes. There is no position control requirement about the roll axis but vehicle roll rates are limited to avoid gyroscopic cross-coupling problems.

RE-ORDER No. [illegible]

The following sources of disturbance torques were considered in arriving at the attitude control subsystem design.

a. Gravity Gradient
b. Aerodynamic Drag
c. Solar Radiation Pressure
d. Micrometeoroid Impacts
e. Magnetic Field
f. Orbit Eccentricity

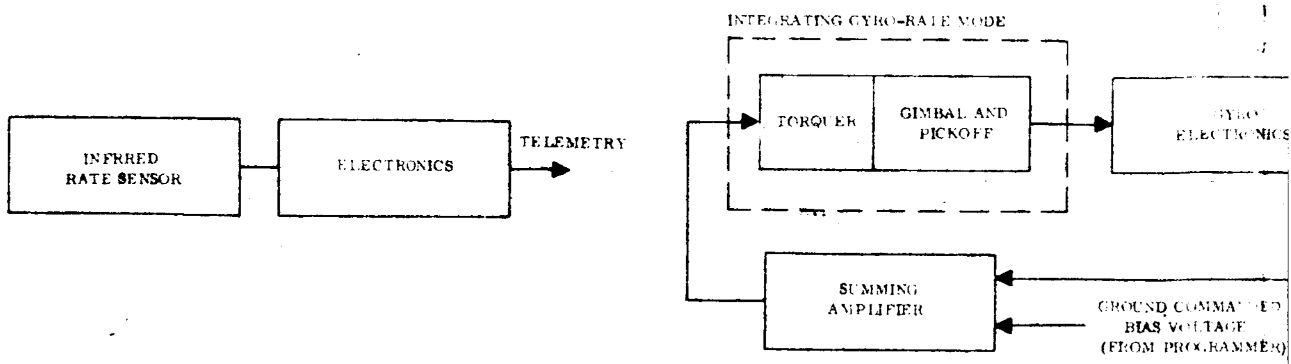
Relatively large momentum storage requirements result from the magnetic effects of the nickel solar concentrator. For example in the Mission C case the total momentum storage requirements in both the pitch and yaw axes is 2.0 ft-lb-sec/orbit and the magnetic effects on the solar concentrator represents 98 percent of this total. However, the proposed attitude control subsystem design is entirely capable of handling magnetic disturbances of this magnitude if they are present. If aluminum concentrators are developed and replace the nickel design proposed the magnetic disturbances could be greatly reduced and the weight of the attitude control subsystem decreased 10 to 20 percent. However, such a development is not necessary to a successful solar thermionic flight experiment. The thermionic converter electrical leads would be designed to cancel the potential magnetic disturbance torques resulting from current loops.

The study results indicate that the attitude control requirements can be met using existing techniques which have been flight proven.

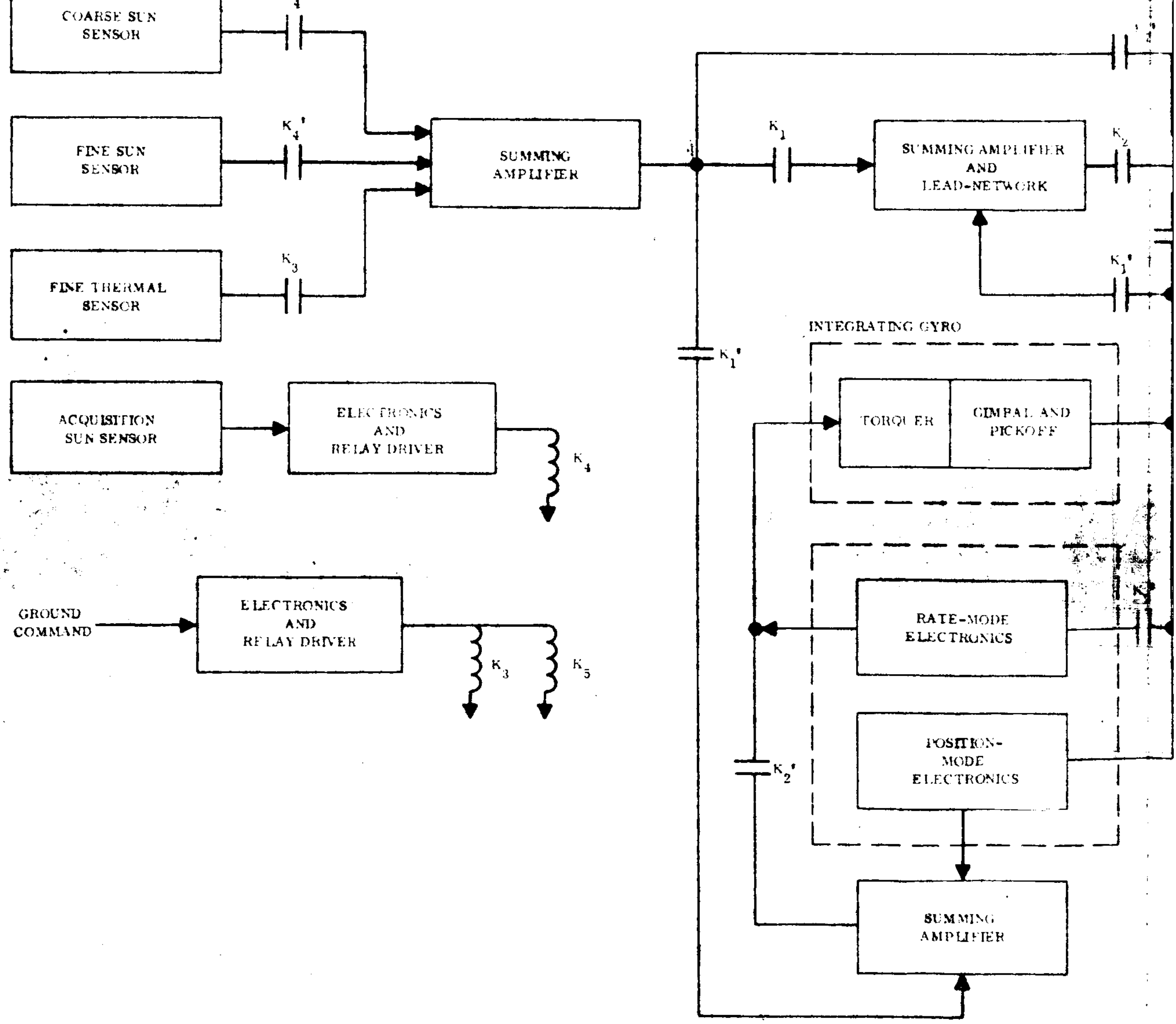
A block diagram of the attitude control subsystem is given in Figure 1-9. This subsystem design is based on the cold gas-flywheel attitude control technique presently in use on the Nimbus and OAO spacecraft. The attitude control subsystem must control the spacecraft under three primary modes of operation.

a. Fine pointing control during the sunlight portion of the orbit

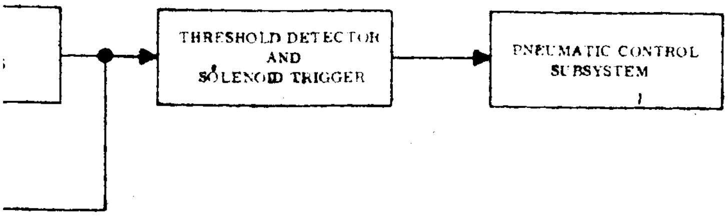
ROLL CONTROL LOOP



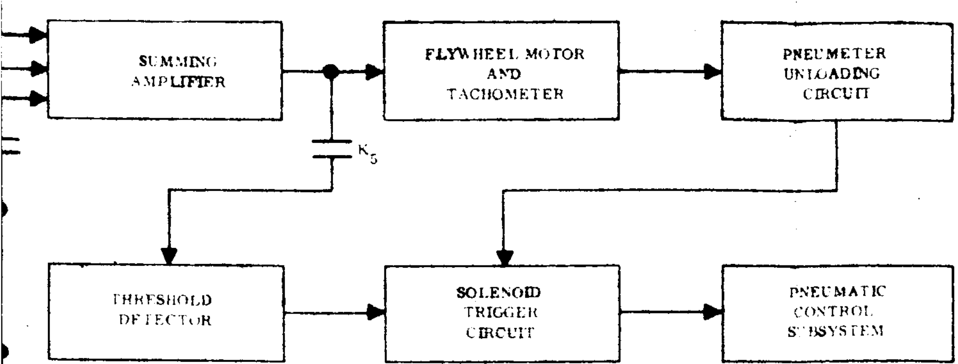
PITCH OR YAW CONTROL L



RE-ORDER NO. 67-323
2 C Z



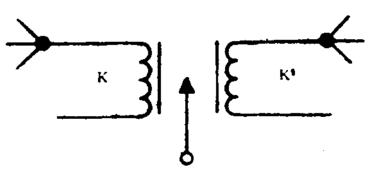
-OOP



MODES OF OPERATION

	CLOSED	OPEN
INITIAL STABILIZATION AND ORIENTATION:	K_2', K_4, K_5	K_2, K_1, K_3
FINE POINTING:		
(a) ALL DAYLIGHT	K_4', K_2, K_1	K_4, K_2', K_3, K_5
(b) DAY-NITE	K_1', K_2, K_4'	K_1 K_2', K_4, K_3, K_5

RELAY NOTATION:
(NON-LATCHING RELAYS)



CURRENT EITHER IN COIL K OR K', NEVER IN BOTH AT SAME TIME

Figure 1-9. Attitude Control Subsystem Block Diagram

b. Initial stabilization and sun acquisition

c. Dark period operation and sun reorientation

Each of these operating modes is discussed further in the following sections.

1.5.5.1 Fine Pointing Control

The fine point mode of operation represents that portion of the mission when the spacecraft pitch and yaw attitude errors must be held to within ±0.1 degree. The pitch and yaw fine pointing control systems are functionally identical. A fine sun sensor is used as the primary attitude error sensing device. A backup sensor is provided in the form of a thermal device that senses the position of the solar concentrator focal point with respect to the thermionic generator aperture. The fine sensor output is used to update a rate-integrating gyro. This is accomplished by feeding a biasing current to the coils of the gyro torquer that is proportional to the difference between the gyro output and the fine sensor output. In this manner, the fine sensor cancels the inherent drift of the gyro during the daylight portion of the orbit. When the vehicle enters the shadow period the gyro will continue to control its attitude, although its drift will no longer be cancelled by the fine sensor output. Thus the attitude error when the spacecraft emerges from the shadow will be directly proportional to the gyro drift.

Fine attitude control about the pitch and yaw axes would be maintained by the method of momentum-interchange between the spacecraft body and the flywheels. Since the spacecraft would be a free body in space, any momentum that the spacecraft absorbed from external torquing forces would be maintained in an inertial sense. The component of momentum about a given spacecraft body axis would be transferred to the flywheel by changing the steady-state speed of the flywheel whose spin axis was aligned parallel to the body axis. The attitude error and rate information provided by the sun sensor and gyro would control the flywheel speed. The acceleration of the flywheel is accomplished by exerting a torque on the wheel through changes in its control phase excitation. This torque would be reflected in the opposite sense on the spacecraft body, thereby resulting in a transfer of momentum from the spacecraft body to the flywheel.

This momentum storing technique operates most efficiently on vehicles that experience cyclic disturbance torques. The flywheel momentum storing capability is made larger than the maximum momentum imparted to the spacecraft by the cyclic torques, thus no external torques need be applied by the mass expulsion devices to counteract these disturbances.

In addition to the cyclic torques on the spacecraft, there would also be cumulative disturbance torques. As the momentum due to the combination of cumulative and cyclic torques absorbed by the flywheel increased, the wheel speed would increase to its no-load speed. At that point the flywheel could not absorb any additional momentum and it is referred to as a "saturated flywheel". Once saturated, the flywheel could not respond to further increases in error signal and therefore could not control the spacecraft attitude until desaturated (slowed down). Momentum would be removed (or dumped) from the flywheel by the pneumatic reaction system. The action of expelled gas would apply an external torque to the spacecraft which would subtract from the total momentum about that axis. As the spacecraft rotated due to this external torque, a change in position would be noted by the fine sensor; this would decrease the signal to the control phase of the flywheel and it would decelerate. The resultant torque due to the flywheel deceleration would reduce the spacecraft body rate. Operation of the control loop would result in an equilibrium condition of zero body rate with the spacecraft holding the steady pointing error necessary to maintain the flywheel speed corresponding to the spacecraft momentum component about the body axis.

1.5.5.2 Initial Stabilization and Sun Acquisition

The spacecraft would be despun in the Mission A and B cases prior to separation from the final booster stage by the yo-yo technique. Despin would not be required in the Mission C case because the final launch vehicle stage is not spin stabilized. The residual angular rates about the pitch and yaw axes after the despin maneuver would be removed by a combination of cold gas torquing and momentum storing. The residual rates about the sun pointing or roll axis would be controlled by cold gas torquing alone.

RE-ORDER No. 64-323

1.5.5.3 Dark Period Operation and Reorientation

During the orbit dark period, the roll axis control would function exactly as it does during the daylight portion since it does not depend on a sun reference. The pitch and yaw control loops would remain in the fine pointing mode throughout the dark period. Since there would be no sun information available, the fine sun sensor would relinquish its part as the prime control sensing device to the position sensing gyro. Due to the fact that the sun sensor would no longer be biasing out the gyro, the gyro output may drift at the maximum short-term rate of 0.1 degree per hour. Therefore, for the worst case condition of a two hour dark period (Mission B), the vehicle would be 0.2 degrees away from null upon emerging from the dark. At this point the fine sun sensor would again resume its prime sensing function.

1.5.6 SPACECRAFT POWER SUBSYSTEM

An example of the spacecraft power requirements is given in Figure 1-10. This particular power profile is for the Mission A case but the Mission B and C cases have similar power requirements.

A schematic of the power subsystem is presented in Figure 1-11 and is the same for each of the missions.

The solar array, using N/P silicon solar cells, supplies power to the load and charging power to the battery when the vehicle is in the daylight portion of the orbit. A rechargeable nickel-cadmium battery supplies the power requirements when the vehicle is in the earth's shadow and short term peaks during the daylight portion of the orbit. The power requirements during initial stabilization, before the solar panels are sun-oriented, would be supplied by a non-rechargeable silver oxide-zinc battery.

The battery charge regulator controls the rate at which the battery is charged. It is a switching type unit which maintains the required average battery charging current and operates at a high efficiency. With this arrangement, the battery also provides coarse voltage

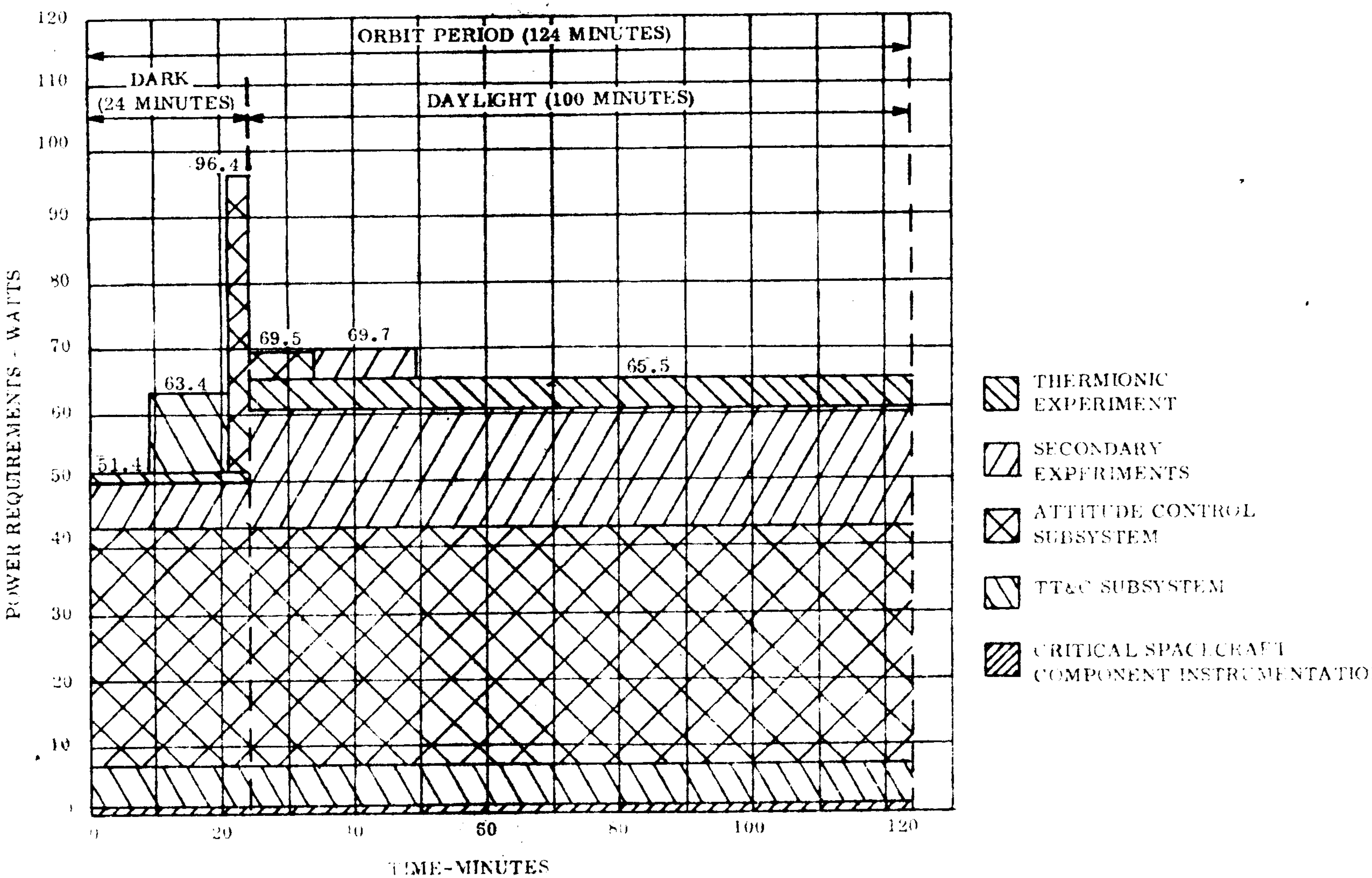


Figure 1-10. Power Requirements for Mission A

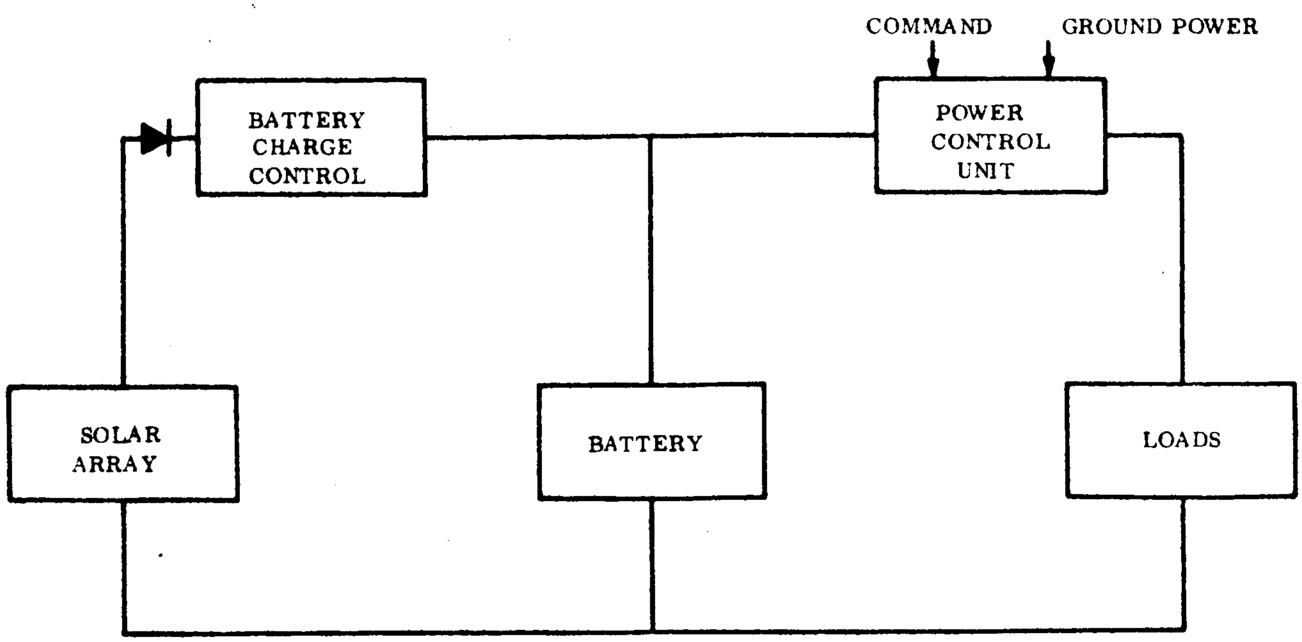


Figure 1-11. Power Supply Subsystem Block Diagram

regulation on the main bus (the bus voltage varying between battery charge and discharge values) with an expected variation of approximately ±15 percent. Each load would provide its own voltage level and regulation.

The power control unit provides for switching of various components according to command and/or programmed inputs, contains input connections from ground power, and may also provide some circuit protection.

1.6 RECOMMENDATIONS

This study establishes a conceptual design for a solar thermionic flight experiment that could be developed with present state-of-the-art capability. The following are proposed as the next logical steps in developing a solar thermionic flight experiment.

STEP 1

Perform a detailed mission analysis and preliminary design of the spacecraft, and estimate development schedules and program cost.

STEP 2

Direct the current development efforts in solar thermionics toward the development of a solar thermionic system of the general design defined in this study. The emphasis should be on developing flight qualified hardware.

These two steps could be undertaken in parallel.

This study identified the following as the major areas requiring further study and/or development effort.

a. Development of a control to optimize the cesium reservoir temperature for a given thermionic converter load and thermal input.

b. Determine if the suggested thermionic generator replacement scheme, or some alternate scheme, is feasible.

c. Determine the maximum emitter temperature the thermionic converters will be exposed to if no flux controls are used during the generator warm-up phase. Determine if the thermionic converters could withstand these short temperature excursions over the total experiment life.

d. Determine if the control electronics for the thermionic generator could be located adjacent to the generator, thus simplifying the experiment design, reducing the electrical lead weight and minimizing the I^2R losses.

e. Development of an electrical circuit for the thermionic generator that would provide the degree of experiment flexibility outlined in this study.

f. Development of an aluminum solar concentrator to replace the nickel one proposed in this study. The use of a solar concentrator made from aluminum rather than nickel would greatly reduce the magnetic disturbance torques acting on the spacecraft.

g. Determine the extent of radiation induced surface effects on the spacecraft electronics in the proposed modified sun-synchronous orbit.

h. Establish by testing the approximate time required to warm up the thermionic generator under simulated space conditions.

i. Determine the feasibility of developing an attitude control system which employs flux sensing around the thermionic generator aperture as the means of error detection, rather than the more common approach of sensing the sun's position and relying on correct alignment between the sun sensor axis and the solar concentrator axis.

j. Continue the evaluation of potential secondary experiments that could be included on a spacecraft designed to conduct a solar thermionic flight experiment. This effort should be continued to insure selection of the best combination of secondary experiments and to better define the requirements of these experiments.

k. Determine if the size and weight of the solar reflective surfaces experiment could be reduced to make it easier to integrate into the spacecraft design.

l. Conduct a thermal analysis to determine if the spacecraft will require an active thermal control system (as assumed in this study) or if a passive system will be sufficient.