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SOLAR POWERED ELECTRIC PROPULSION
PROGRAM SUMMARY REPORT

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JPL Contract No. 951144

APPROVED: _____



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ACKNOWLEDGMENTS

The work to be reported in the following sections has been accomplished through the efforts of a number of people. These people have worked together as a closely coordinated team under the general supervision of the key personnel listed below. Although, in some cases, persons were involved in more than one aspect of the analysis, an attempt has been made to indicate the area of greatest participation.

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1. INTRODUCTION

A program was initiated at Hughes in February 1965 under contract to JPL to determine the feasibility of solar-powered electric-propulsion systems for use in deep space missions of the near future. The overall program, sponsored by the OAR&T office of NASA Headquarters, was divided into a study phase and a hardware phase.

The initial study phase of this program was concentrated on the design of solar-powered electric-propulsion spacecraft capable of performing Mars orbiter missions in the early 1970's utilizing the Saturn IB/Centaur and Atlas/Centaur launch vehicles. A low thrust mission analysis was conducted to determine the payload capability of the electrically propelled spacecraft and subsequently compare it to an all-chemical vehicle, and to establish the optimum design points for the propulsion system. To accomplish these objectives, a continuous thrusting, three dimensional trajectory program was developed which includes considerations of perturbations due to earth, Mars, and Jupiter, as well as effects of solar pressure acting on the solar panel.

Results of this initial phase of the study indicated that solar-electric propulsion systems based on state-of-the-art technology and components could be developed at the present time for unmanned space exploration applications. Furthermore, it was shown that these systems, if properly designed, could be made highly reliable with minimal weight penalties and could meet the constraints imposed by spacecraft integration requirements. Finally, it was concluded that total solar-powered electric-propulsion systems specific weights of 50 pounds per kilowatt or less were achievable and that these systems were potentially more attractive than all-chemical spacecraft, based on both mission performance and cost effectiveness. Results of this study are presented in Reference 1.

The purpose of the hardware phase of the program was to verify experimentally the ion propulsion system performance and designs established during the study effort. Therefore, the hardware verification effort was devoted to developing, integrating, and testing a complete ion engine system that would satisfy the requirements and constraints imposed by a solar-powered, electrically propelled interplanetary spacecraft. The test served as a conclusive demonstration that all components and subsystems required to construct an ion engine system for solar powered spacecraft are

2. SUMMARY

Results of this phase of the solar-powered electric-propulsion spacecraft study have shown a significant improvement in mission capability for unmanned planetary exploration spacecraft by the use of advanced photovoltaic power systems in conjunction with electric propulsion thruster systems. In addition, the results have demonstrated the great versatility of a specific modularized solar-powered electric-propulsion system design, in being adaptable to a wide variety of launch vehicles and several different planetary missions using a specific launch vehicle. The cost as well as technical advantages of the development of one modularized electric propulsion system, as opposed to the need for development of several new all-chemical boosters to accommodate the wide variety of planetary missions now under consideration by NASA, is becoming quite apparent.

A complete ion propulsion system of the type used in all of the spacecraft designs presented herein, employs flight designed components and subsystems, and has been operated closed loop for 500 hours. The power conditioning and control system, as well as the thruster and feed system, were operated in a vacuum chamber which simulated the thermal environment of space. This test served as a conclusive demonstration that all components and subsystems required to construct an ion engine system for solar powered spacecraft are currently available as state-of-the-art technology.

PROPULSION SYSTEM DESIGN

Based on the "optimum" design points, as determined by the mission analysis, two detailed ion engine system designs were derived. The two systems considered were 1) a 9.6 kw engine system for a Mars orbiter mission spacecraft launched by a Titan III M boost vehicle, and 2) a 17 kw engine system for a Jupiter flyby mission spacecraft launched by an Atlas/Centaur boost vehicle. These complete ion engine systems, which consist of thruster arrays, storage and feed systems, and power conditioning and control systems, have been designed to satisfy the various constraints and requirements of the mission, reliability, solar power source, and spacecraft integration. For each mission, designs were derived for systems based on both the oxide cathode and liquid cathode mercury bombardment ion thrusters. Only the oxide cathode engine systems will be reviewed here.

currently available as part of the state-of-the-art technology. Results of this program, which included a 500-hour life test, are reported in Reference 2.

A reliability analysis was made of the thermal-vacuum prototype power conditioning and control system developed during the aforementioned hardware phase of the program and is reported in Reference 3. The reliability analysis of this system (with no modifications from the prototype configuration) results in a total probability of success of 0.85 for a 10,000 hour mission. This analysis has helped define areas of the system where substantial reliability improvements can be made with only minor changes in circuit design and virtually no penalty in system weight. The application of these improvements to a proposed flight model optimized for SERT II requirements has yielded the substantially higher reliability of 0.96 for the complete system for a 10,000 hour mission.

This document is the final summary report of Phase II of the study program that was concentrated on the design of a specific modularized electric propulsion system which would be adaptable to a wide variety of NASA launch vehicles and several different planetary missions using a given launch vehicle. Three detailed electric propulsion spacecraft designs were derived. Two of the spacecraft - one compatible with the Titan III M launch vehicle and one compatible with the Saturn IB/Centaur launch vehicle - were designed to have the capability of accomplishing Mars orbiter/lander missions during the 1971 through 1979 launch opportunities. The third spacecraft - designed to be compatible with the SLV-3C/Centaur launch vehicle - is designed to accomplish a Jupiter flyby and out-of-the-ecliptic missions. Also included is an evaluation of the capabilities of similar spacecraft in accomplishing a Venus orbiter and Mercury flyby missions.

Detailed documentation of the results of this mission analysis and spacecraft design study appear in the following sections.

The valuable assistance of the technical monitors - Jerome P. Mullin of OAR&T and John W. Stearns of JPL - has greatly aided in arriving at a meaningful presentation of the results presented herein, and is gratefully acknowledged.

REFERENCES

1. "Solar Powered Electric Propulsion Spacecraft Study," Final Report, prepared for JPL by Hughes under Contract No. 951144, December 1965.
2. "Development and Test of an Ion Engine System Employing Modular Power Conditioning," Final Report, prepared for JPL by Hughes under Contract No. 951144, September 1966.
3. "Reliability Analyses of Modular Power Conditioning and Control Systems for Ion Engines," Final Report, prepared for JPL by Hughes under Contract No. 951144, October 1966.

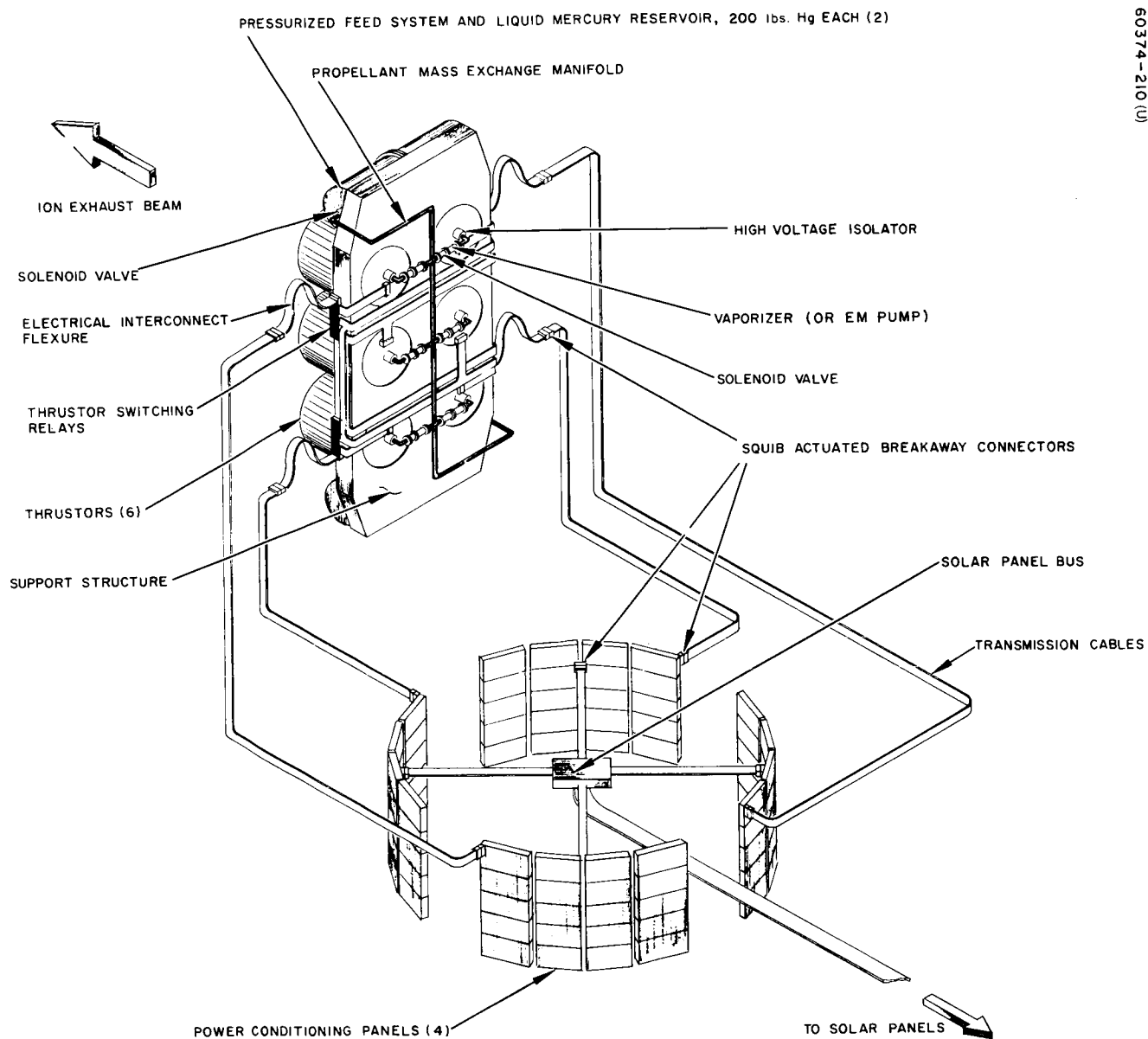


Figure 2-1. Conceptual Drawing of 9.6 Kilowatt Propulsion System

The mission studies provided the design points for the two missions under primary consideration as given in Table 2-1.

TABLE 2-1. PROPULSION SYSTEM DESIGN POINTS

Specifications	Mars Orbiter	Jupiter Flyby
Power, kilowatts	9.6	17
Specific impulse, seconds	4000	3500
Propellant requirement, pounds	394*	865

* Maximum for launch opportunities considered.

Based on reliability and power matching considerations, the total number of thruster modules to be employed in building up to a high power propulsion system was established. It was shown that, for the 9.6 kw Mars orbiter system, four initially operating modules and two in standby were desirable while, for the 17 kw Jupiter flyby system, the optimum combination was six initially operating with two in standby.

Conceptual drawings of complete ion propulsion systems which satisfy the above design constraints as well as spacecraft integration requirements are shown in Figure 2-1 and Figure 2-2. Figure 2-2 shows the integrated 17 kw system. As shown it consists of eight thruster modules. Initially, six of these would be operating and two would be in standby to satisfy the reliability and power matching criteria.

For the 17 kw system the thruster array is based on a 2.6 kw, 25 cm, mercury bombardment engine operating at low current density (e.g., 2 mA/cm²) to ensure long life. Also shown in Figure 2-2 is a conceptual layout of the feed system and propellant tankage. Each of the six reservoirs has a 145 pound liquid mercury capacity and is 28 cm in diameter. A modular concept is employed in the tankage system for three reasons:

- 1) Each tank contains a positive expulsion system which, if a single tank were used, would have to support almost 1000 pounds of liquid mercury under launch conditions.
- 2) Redundancy techniques can again be employed for increased reliability.
- 3) Packaging considerations (especially the desirability of the tankage not protruding beyond the ion thruster exit plane).

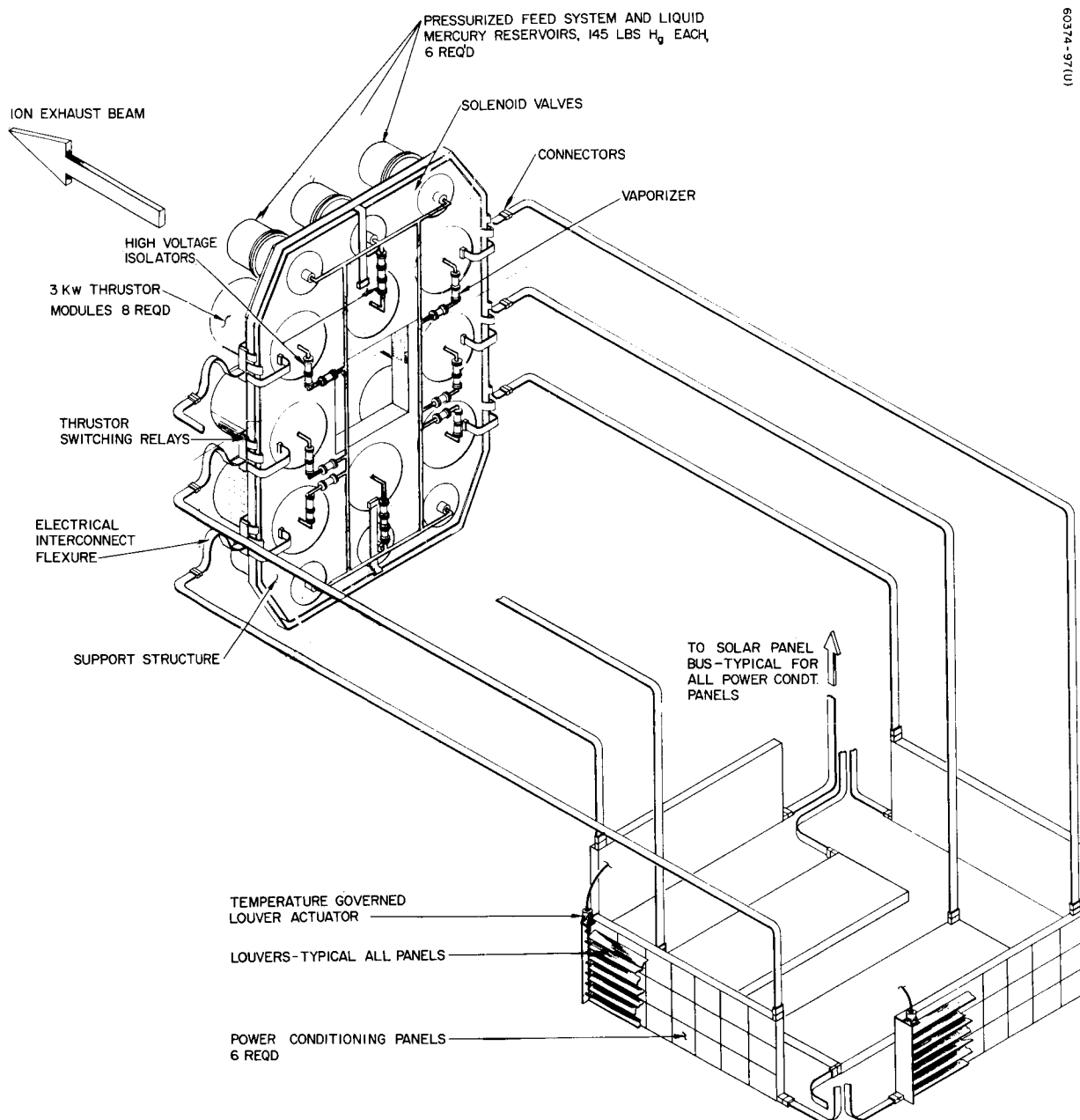


Figure 2-2. Conceptual Drawing of 17 Kilowatt Propulsion System

The rest of the feed system consists of:

- 1) On-off valves to stop propellant flow to nonoperating thrusters.
- 2) Phase separators that convert the liquid mercury to a vapor.
- 3) High voltage isolators which electrically decouple the high voltage thrusters from the feed systems and propellant tanks.
- 4) A propellant distribution system which allows thrusters to be fed from any or all tanks. The overall thruster and tankage array dimensions are 3.5 by 5 feet.

The power conditioning system is made up of six panels of modules, one panel consisting of the power supplies required for a single thruster. Each panel is approximately 28 by 32 inches and is designed in a flat panel for radiation cooling and for attachment to the spacecraft outer skin. Any panel can be switched to a standby thruster should an operating unit fail. Again, extra power conditioning panels could be provided to increase system reliability. However, the reliability of each panel can be made arbitrarily high by increasing the number of power conditioning modules per panel. This reliability advantage among others accrues from the unique power conditioning system design employed. Each power supply is made up of a number of low voltage, low power modules strung in series such that the required thruster voltage and power is obtained. By placing a number of extra standby modules in series with those required, the reliability of the individual supplies can be increased to any desired level.

In general the design considerations discussed above also apply to the 9.6 kw system in Figure 2-1. That system, however, consists of six thrusters, four operating and two standbys, two propellant tanks (based on packaging considerations) and four power conditioning panels. The thruster and tank array is 2.5 by 5.5 feet. Each power conditioning panel is again 28 by 32 inches.

Weight breakdowns for both the 17 kw and 9.6 kw systems are given in Table 2-2. As indicated the specific weights in both cases are well below the 25 lb/kw design goal. Including such items as cabling, plumbing, structure and mechanisms, and redundancy, the estimated specific weights for the 17 kw and 9.6 kw systems are 20 and 21 lb/kw, respectively. It should be noted that the specific weight of the basic propulsion system components such as thrusters, feed system, and power conditioning and controls is on the order of 15 lb/kw. The remainder of the total system weight is due to items that are functions of the mission profile and spacecraft configuration and, therefore, cannot be considered as directly contributing to the engine system figure of merit.

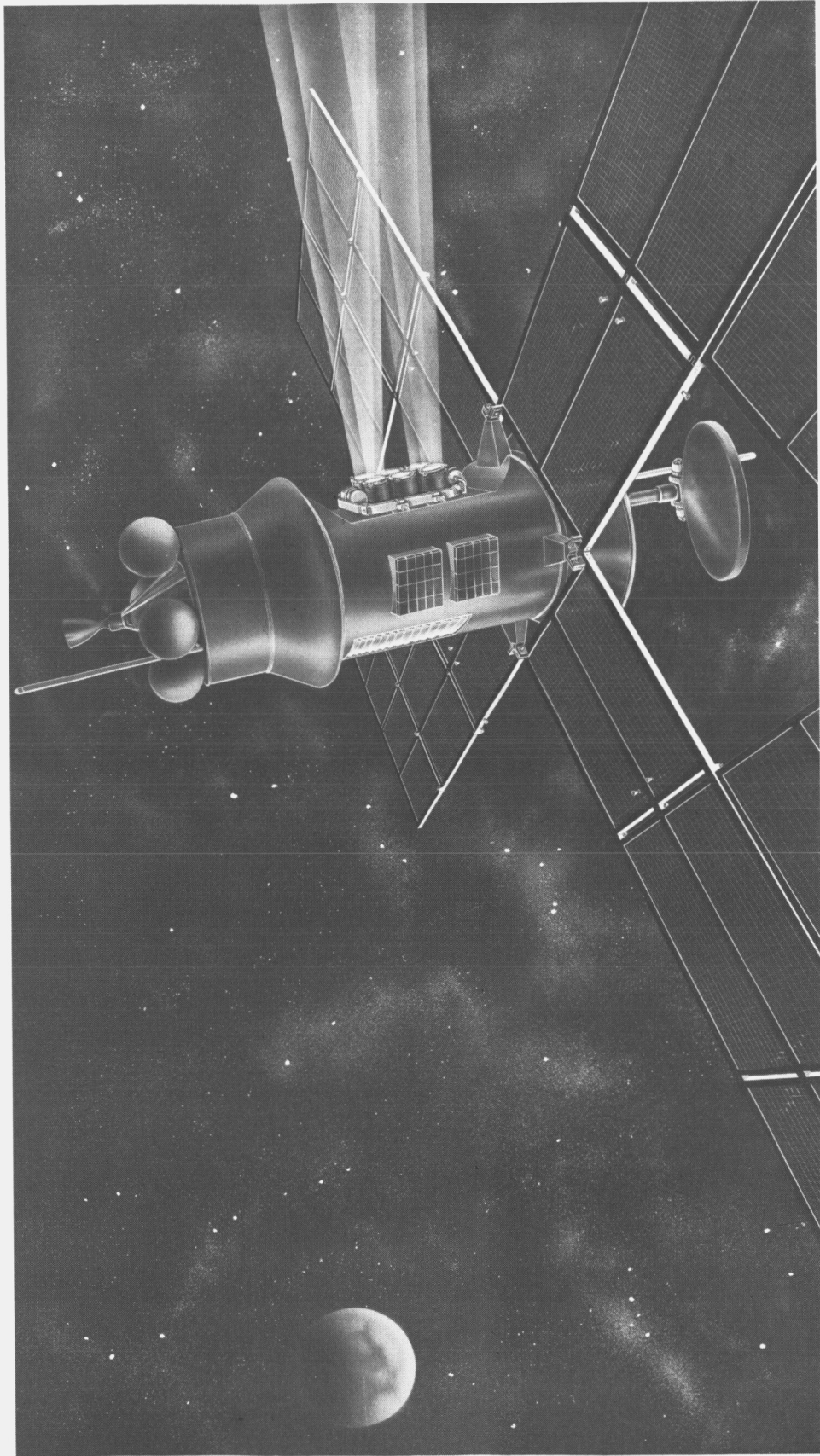


Figure 2-3. Titan III M SEP Spacecraft in Transit to Mars

TABLE 2-2. SYSTEM WEIGHT BREAKDOWN (POUNDS)

Subsystem	17 kilowatts	9.6 kilowatts
Thruster*	80	60
Feed system and plumbing*	24	18
Reservoir	60	24
Power conditioning and controls*	120	72
Cabling	7	6
Structure	40	20
Translator	10	6
Total weight, pounds	341	206
Specific weight, lb/kw	20	21
*Specific weight of basic propulsion subsystems, lb/kw	13	16

SPACECRAFT DESIGN

During the course of this study, three detailed electric propulsion spacecraft designs were derived. Two of the spacecraft, one compatible with the Titan III M launch vehicle and one compatible with the Saturn IB/Centaur launch vehicle, were designed to have the capability of accomplishing Mars orbiter/lander missions during the 1971 through 1979 launch opportunities. The Mars missions are ones in which the spacecraft uses a chemical retro system to attain a given Mars orbit after which the lander capsule is launched from orbit. Details of the mission are given in the following section. The third spacecraft, designed to be compatible with the SLV-3C/Centaur launch vehicle, is configured to accomplish a Jupiter flyby mission with a launch in December of 1973. A fourth spacecraft is presented which is capable of accomplishing an out-of-ecliptic mission; however, this configuration is a slightly modified version of the Jupiter flyby mission spacecraft.

A perspective view of each spacecraft, depicted in the fully deployed configuration during heliocentric transfer, is presented in Figures 2-3 through 2-6 with corresponding weight breakdowns given in Tables 2-3 through 2-7. For comparison purposes, a comparable weight breakdown for a version of the Saturn IB/Centaur spacecraft for a 1973 Mars mission as per the Phase I study ground rules (i. e., lander launched before retro into orbit) is presented in Table 2-5. All of these spacecraft designs utilize the same modular electric propulsion system design, i. e., derivatives of the

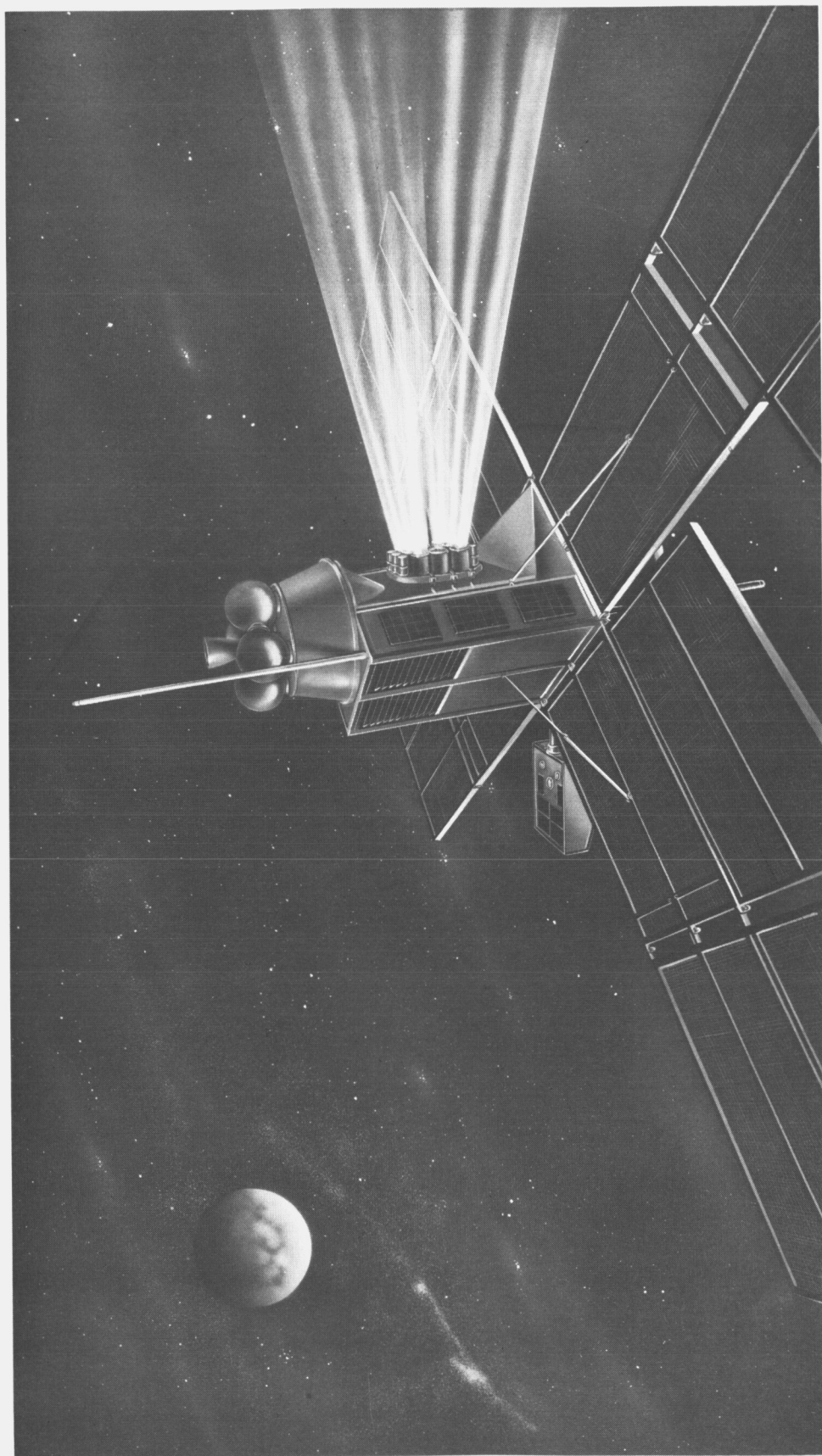


Figure 2-4. Saturn IB/Centaur SEP Spacecraft in Transit to Mars

Boeing developed rectangular folding modular type solar cell array and the Hughes developed modular electric engine system array. The electric propulsion systems for the various spacecraft configurations differ only in the solar panel area and number of engine system modules employed to meet the mission acceleration requirements.

Attitude control for all of these spacecraft is accomplished by means of a resistojet control system which uses nitrogen in a cold gas mode for initial acquisition and planet orbiting phases of the missions, and utilizes nitrogen in the hot gas mode in conjunction with the ion engine array translation for control whenever full solar power is available.

The communications systems for all spacecraft include 50 watts transmitter power and both forward coverage and aft coverage omni-directional antenna to assure continuous communications during the boost phase and initial acquisition. Different sizes of high gain directional antenna (parabolic or planar array) are required for the various configurations because of the varying booster shroud dimensions and packaging limitations as dictated by the solar array area required for the specific mission. In addition, for the Jupiter flyby spacecraft, a medium gain antenna is employed during the intermediate phase of the heliocentric transfer because it requires only coarse pointing accuracy. An exception to this, of course, is the direct out-of-ecliptic mission spacecraft (Figure 2-6). At a constant 1 AU, the telecommunications range for this mission is close enough to permit operation of all links via a hemispherical coverage antenna. (A second antenna to cover the vehicle's shadow side is provided, but would be used only in case of temporary loss of attitude control.)

Because of the large change in energy balance on the spacecraft during the interplanetary missions considered, a portion of the spacecraft outer surface on all configurations is utilized for an active thermal control variable emittance device. It consists of several independent louver panels, each with a sensor actuator control device.

In all arrangements studied the engine cluster is fixed so that its thrust vector is directed parallel to the solar array plane.

The structural arrangements of the Jupiter flyby and the out-of-ecliptic mission spacecraft are essentially identical. The major portion of the spacecraft base is enclosed for thermal control requirements of the electronics and power conditioning. A truss structure extends upward to provide support for the engine cluster and a square framework at the top for the main hinge attachment of the four solar panel assemblies. The truss structure in the vicinity of the engine cluster is constructed to allow longitudinal and lateral translation of the entire engine system.

For both the Mars mission spacecraft, the spacecraft bus (Figures 2-3 and 2-4) is completely enclosed and houses the major subsystems such as the nitrogen gas tanks for attitude control, bus or body-fixed scientific payload, communications electronics, and guidance and control equipment.

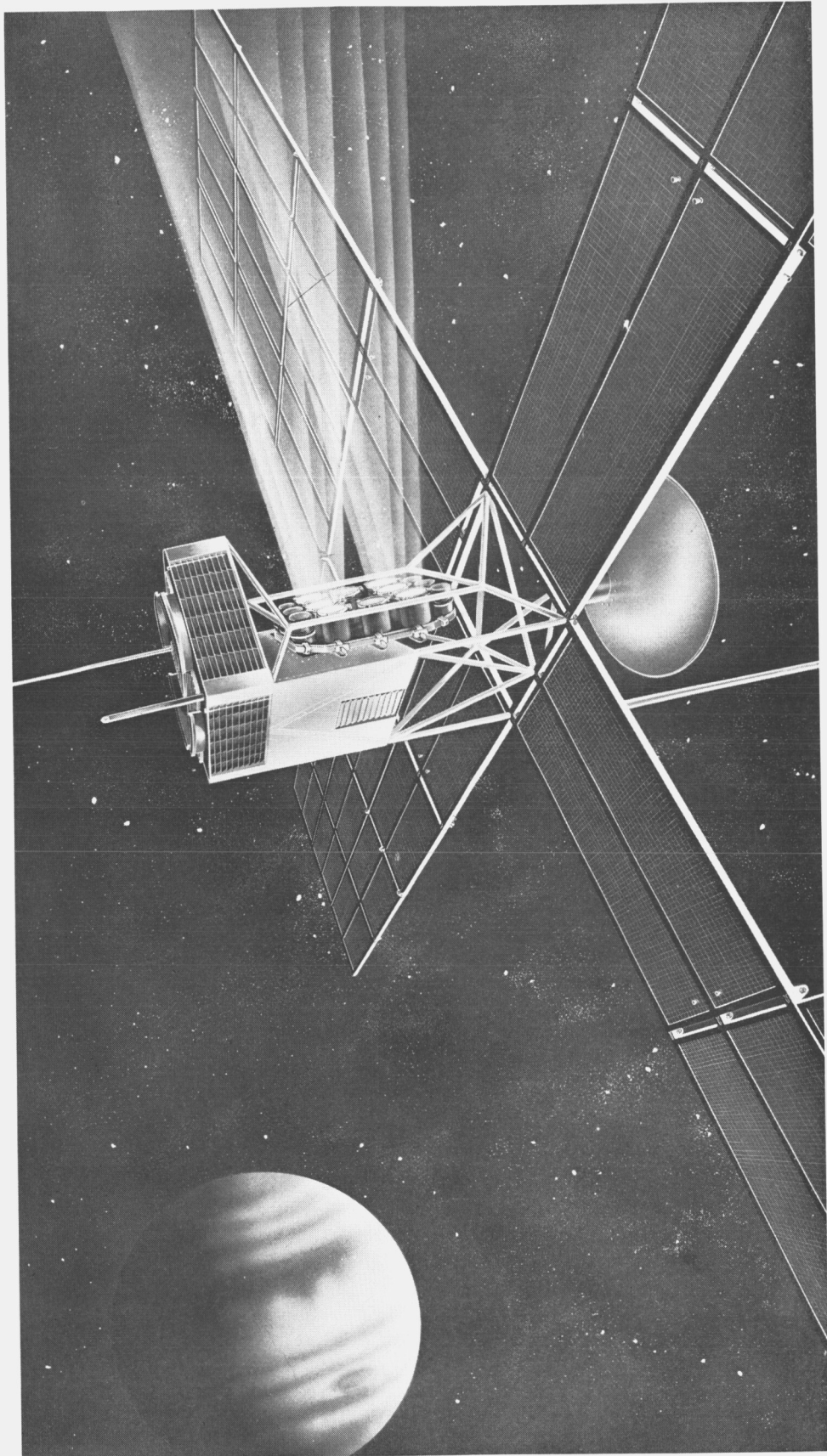


Figure 2-5. Atlas (SLV - 3C)/Centaur SEP Spacecraft in Transit to Jupiter

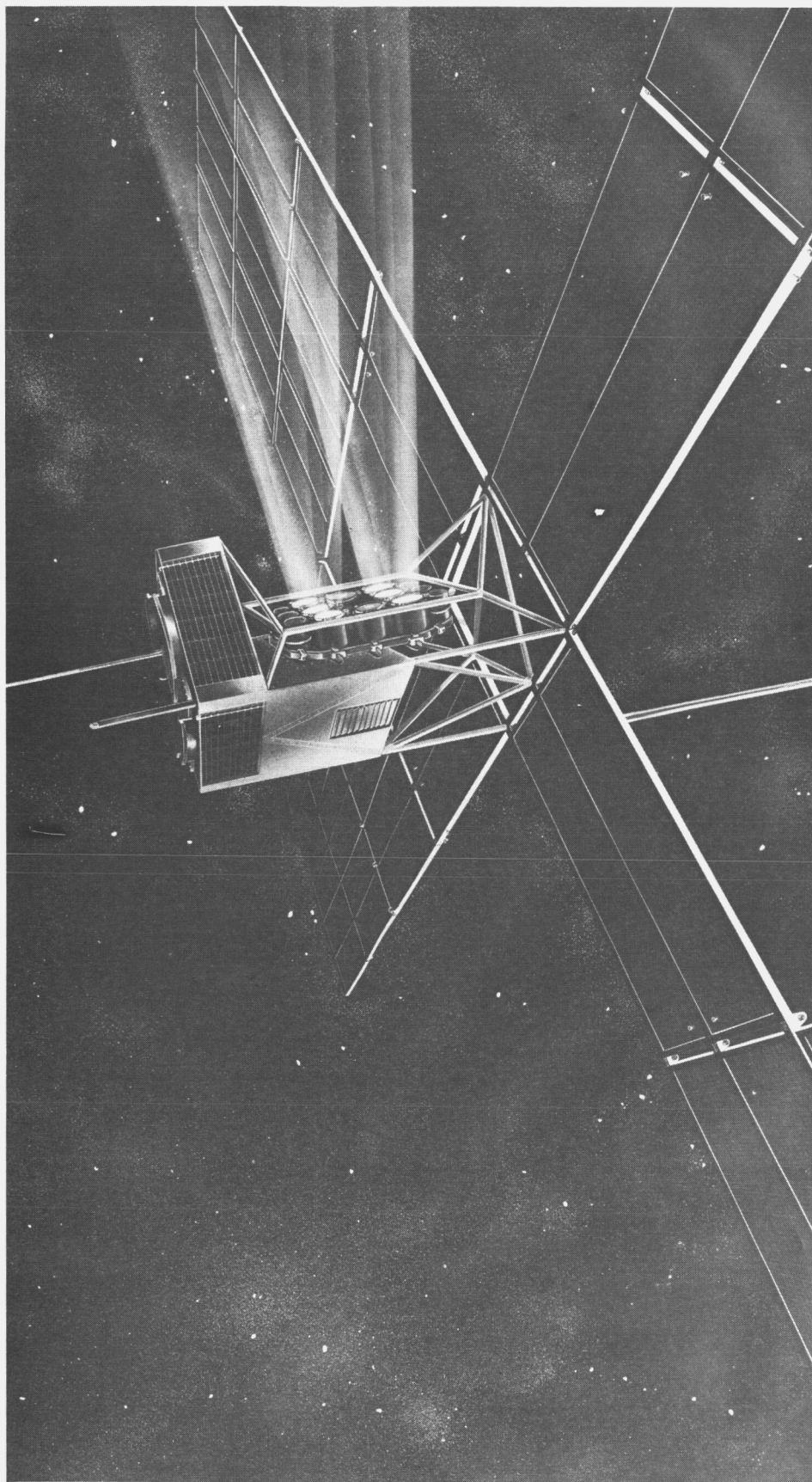


Figure 2-6. Atlas (SLV - 3C)/Centaur SEP Spacecraft in Transit
During Out-of-Ecliptic Mission

TABLE 2-3. TITAN III M SEP SPACECRAFT WEIGHT BREAKDOWN
1973 MARS ORBITER AND LANDER MISSION
(9.6-kw power level)

Items	Weight, pounds	
	Injected	At Approach
Mission payload	1960	1960
Science payload		
Telecommunications	180	
Power	270**	
Solar array (portion jettisoned)	400	---
Electric engine system	250	---
Thrustor system	66	
Propellant tankage	24	
Power conditioning and controls	94	
Structure and mechanisms	34	
Feed system	32	
Propellant, Hg	273	---
Guidance and control	95	90
Structure	455	405
Thermal control	40	40
Electrical harness	60	54
Retro system	1480	1480
Total	5013	4029

* Lander capsule weight is included as part of science payload weight.

** 100 pounds of power associated with the mission payload is that portion of the solar array subsystem retained with the orbiter.

Notes:

- 1) The retro system weight is based on an approach velocity of 1.14 km/sec, propellant I_{sp} = 315 seconds, and mass fraction of 0.90.
- 2) Variation of scientific equipment weight versus selected orbits is as follows:

Orbit	Science Equipment Weight, pounds
1000 km circular	1230
5000 km circular	1510
1000 x 20,000 km elliptical	2230

TABLE 2-4. SATURN IB/CENTAUR SEP SPACECRAFT WEIGHT
BREAKDOWN, 1973 MARS ORBITER AND LANDER MISSION
(14.4-kw power level)

Items	Weight, pounds	
	Injected	At Approach
Mission payload	3130	3130
Science payload		
Telecommunications	2335*	
Power	200	
	595**	
Solar array (portion jettisoned)	445	---
Electric engine system	360	---
Thruster system	95	
Propellant tankage	36	
Power conditioning and controls	135	
Structure and mechanisms	49	
Feed system	45	
Propellant, Hg	410	---
Guidance and control	140	130
Structure	700	500
Thermal control	50	25
Electrical harness	65	58
Retro system	2220	2220
Total	7520	6063

* Lander capsule weight is included as part of science payload weight.

** 275 pounds of power associated with the mission payload is that portion of the solar array subsystem retained with the orbiter.

Notes:

- 1) The retro system weight is based on an approach velocity of 1.14 km/sec, propellant I_{sp} = 315 seconds, and mass fraction of 0.90.
- 2) Variation of scientific equipment weight versus selected orbits is as follows:

Orbit	Science Equipment Weight, pounds
1000 km circular	1915
5000 km circular	2335
1000 x 20,000 km elliptical	3405

TABLE 2-5. SATURN IB/CENTAUR SEP SPACECRAFT WEIGHT
BREAKDOWN, 1973 MARS ORBITER AND LANDER MISSION
LANDER CAPSULE EJECTED PRIOR TO ORBIT
(14.4-kw power level, 5000-km circular orbit)

Items	Weight, pounds	
	Injected	At Approach
Mission payload	1665	1665
Science payload	870	
Telecommunications	200	
Power	595*	
Lander capsule	2300	---
Solar array (portion jettisoned)	445	---
Electric engine system	360	---
Thrustor system	95	
Propellant tankage	36	
Power conditioning and controls	135	
Structure and mechanisms	49	
Feed system	45	
Propellant, Hg	410	---
Guidance and control	140	130
Structure	700	500
Thermal control	50	25
Electrical harness	65	58
Retro system	1385	1385
Total	7520	3763

*275 pounds of power associated with the mission payload is that portion of the solar array subsystem retained with the orbiter.

Notes:

- 1) The retro system weight is based on an approach velocity of 1.14 km/sec, propellant I_{sp} = 315 seconds, and mass fraction of 0.90.
- 2) Variation of scientific equipment weight versus selected orbits is as follows:

Orbit	Science Equipment Weight, pounds (Science payload including 2300 pound lander)
1000 km circular	2885
5000 km circular	3170
1000 x 20,000 km elliptical	3840

TABLE 2-6. ATLAS (SLV-3C)/CENTAUR SEP SPACECRAFT WEIGHT
BREAKDOWN, 1973 JUPITER FLYBY MISSION
(17.0-kw power level)

Items	Injected Weight, pounds
Mission payload	238
Scientific payload	23
Telecommunications	165
Battery power subsystem	50
Solar power subsystem*	(765)
Solar array, 45 lb/kw	765
Electric engine system, 20 lb/kw	340
Thrustor system	80
Propellant tankage	60
Power conditioning and controls	126
Structure and mechanisms	50
Feed system	24
Propellant, Hg	865
Guidance and control	107
Structure (including micro-meteoroid shielding)	205
Thermal control	45
Electrical harness	50
Total	2615

*Entire solar array power available for mission applications at Jupiter flyby.

$C_3 = 1.22 \text{ (km/sec)}^2$, 900 days mission time.

TABLE 2-7. ATLAS (SLV-3C)/CENTAUR SEP SPACECRAFT WEIGHT
BREAKDOWN, 24 DEGREES OUT-OF-ECLIPTIC MISSION
(17.0-kw power level)

Items	Injected Weight, pounds
Mission payload	298
Scientific payload	137
Telecommunications	111
Battery power subsystem	50
Solar power subsystem*	(765)
Solar array, 45 lb/kw	765
Electric engine system, 20 lb/kw	340
Thrustor system	80
Propellant tankage	60
Power conditioning and controls	126
Structure and mechanisms	50
Feed system	24
Propellant, Hg	865
Guidance and control	107
Thermal control	10
Electrical harness	50
Total	2615

* Entire solar array power available for mission applications in out-of-ecliptic final orbit.

Also, on both spacecraft the lander capsule is attached to the forwardmost section of the spacecraft bus and the retro rocket and tankage are mounted such that the thrust vector is directed along the spacecraft longitudinal axis. As with the other spacecraft designs, the electric engine thruster bank is mounted externally to the spacecraft bus to allow longitudinal and lateral translation for control purposes.

MISSION PERFORMANCE

The capability of the foregoing solar-powered electric-propulsion spacecraft in accomplishing the Mars orbiter lander missions during the 1970's launch opportunities is summarized in Table 2-8. Also included for comparison purposes is the mission performance of a Saturn IB/Zero Stage version of this spacecraft, although no configuration is presented.

TABLE 2-8. LAUNCH VEHICLE COMPARISON
MARS ORBITER AND LANDER MISSIONS
5000 x 5000 KILOMETER ORBIT

Trajectory Characteristics				Titan III M 9.6 kw at 1.0 AU		Saturn IB/Centaur 14.4 kw at 1.0 AU		Saturn IB/Zero Stage 41.6 kw at 1.0 AU	
Year	C_3 , $\left(\frac{\text{km}}{\text{sec}}\right)^2$	Approach Velocity, km/sec	Flight Time, days	Injected Weight	Mission* Payload	Injected Weight	Mission* Payload	Injected Weight	Mission* Payload
1971	4.23	1.96	231	5554	1897	8320	3060	23,300	8127
1973	6.93	1.14	293	5013	1960	7520	3130	21,700	8782
1975	5.49	0.90	404	5301	2140	7950	3422	22,500	9392
1977	4.07	1.10	432	5586	2215	8380	3539	23,400	9422
1979	3.07	1.38	407	5786	2217	8680	3534	24,000	9387

* Mission payload includes lander capsule, orbital scientific instrumentation, telecommunications equipment and power generation equipment.

A typical sequence of events for the Mars mission is as follows:

- 1) The SEP spacecraft is to be boosted to the appropriate C_3 energy level by the chemical launch vehicle, the shroud jettisoned, and the spacecraft is then separated from the expended launch vehicle motor.

- 2) Deployment of low-gain antennas.
- 3) Deployment of solar cell array.
- 4) Initial solar acquisition and orientation of spacecraft for heliocentric transfer phase of mission.
- 5) Deployment of high-gain antenna and science platform may occur after proper solar panel deployment and operation. Electric engine startup initiated.
- 6) Transit phase: continuous guidance and control and communications.
- 7) Approach phase: jettison subsystems not required for orbiter operation.
- 8) Attitude control maneuver and retro fire into desired planet orbit.
- 9) Retro system separation, in-orbit attitude control maneuvers, and lander separation.
- 10) Continual orbital and lander data transmission to earth throughout orbiter life.

The performance comparison is based on the "mission payload," defined as the lander capsule, orbital scientific instrumentation, telecommunications equipment, and power generation equipment, that can be placed in a 5000 kilometer circular Martian orbit during the 1970 to 1980 time period. It is evident from these results that the same basic electric propulsion system design concept is compatible with spacecraft designs ranging in Mars mission payload capability from a minimum of 1897 pounds (for a 1971 Titan III M launched spacecraft) to a maximum of 9422 pounds (for a 1977 Saturn IB/Zero Stage launched spacecraft).

In the performance of the Jupiter flyby mission with the previously described spacecraft (Figure 2-3) launched in December 1973 by the SLV-3C/Centaur booster, a 900 day mission time (with thrust termination at 540 days) will permit a mission payload of 238 pounds exclusive of the solar power subsystem. This power system, being no longer required for electric propulsion, can provide over 850 watts of power for science and communications use at Jupiter distance. The initial thrust attitude orientation is approximately opposed to the earth's orbital velocity (as is the hyperbolic excess velocity at departure), thereby permitting the vehicle to remain inside 1 AU for approximately 140 days. Preliminary Hughes studies indicate that this approach has a lower power requirement than a direct transfer trajectory and a resulting performance improvement.

In the performance of the out-of-ecliptic mission with the electric propulsion spacecraft launched by the SLV-3C/Centaur launch vehicle (Figure 2-6), the engines are alternately thrusting normal to the ecliptic plane for 90 days and coasting for 90 days during the flight time required to achieve a given out-of-ecliptic angle, and the spacecraft remains at 1 AU at the same heliocentric longitude as the earth, thus providing a minimum communication distance. A mission payload capability of 298 pounds, exclusive of the solar-power subsystem (which is capable of supplying the full 17 kilowatts of power after thrust termination), is available in a heliocentric orbit inclined 24 degrees to the ecliptic plane 600 days after launch.

MISSION COMPARISON

To further illustrate the advantages of the solar-powered electric-propulsion spacecraft concept, a performance comparison of electric-propulsion spacecraft with all-chemical propulsion spacecraft for the Mars orbiter/lander mission has been made.

The performance comparison for the Mars mission is presented for both the 1973 and 1979 launch opportunities in Table 2-9. For the Mars mission comparison, the performance capability using electric propulsion is based on the Titan III M launch vehicle, with the spacecraft previously defined (see Table 2-3). The required injected weights for the all-chemical spacecraft that would in effect provide the identical mission payload capability for the 1973 and 1979 launch opportunities are shown in Table 2-9 in comparison to the solar-electric spacecraft capabilities.

TABLE 2-9. PERFORMANCE COMPARISON OF ELECTRIC-PROPULSION SPACECRAFT WITH ALL-CHEMICAL SPACECRAFT

	Solar-Powered Electric-Propulsion Spacecraft		Equivalent All-Chemical Spacecraft	
	1973	1979	1973	1979
Required C_3 , (km/sec) ²	6.93	3.07	15.5	14.0
Approach velocity, km/sec	1.14	1.38	3.3	4.5
Mission payload, pounds*	1960	2217	1960	2217
Required injected weight, pounds	5013	5786	7625	18,200
Required launch vehicle	Titan III M	Titan III M	Saturn IB/ Centaur	Saturn IB/ Zero Stage

*For 5000 kilometer circular Martian orbit.

The 1979 all-chemical spacecraft in this comparison is approximately the size currently under study in the Voyager program to be launched by the Saturn V launch vehicle. The 1973 all-chemical spacecraft is approximately that previously studied for the Saturn IB/Centaur launch vehicle, no longer being considered for development.

Within the capabilities of the family of launch vehicles considered here, it would require a Saturn IB/Zero Stage booster, now only in the preliminary study phase, to accomplish these Mars missions with a comparable mission payload to that attainable with a Titan III M launched solar-powered electric-propulsion spacecraft. Estimated launch vehicle costs (including launch operations) for the Saturn IB/Zero Stage booster are approximately four times that for the Titan III M.

AREAS FOR FURTHER INVESTIGATION

During the course of this study, several areas have been uncovered in which interactions between a solar-powered electric-propulsion system and the spacecraft may place rigid constraints upon the total system design. These areas that warrant further investigation are listed below.

- 1) The use of ion thrusters for spacecraft propulsion will introduce into the environment surrounding the spacecraft elements not presently encountered, namely, neutral and ionic mercury or cesium. Since both of these elements are heavy metals it is possible to hypothesize undesirable interactions occurring between these propellant particles and the spacecraft such as sputtering, implantation, and/or coating of sensitive instruments and components. It is recommended that an experimental program be established to identify the propellant-spacecraft interactions and determine the conditions under which they can occur.
- 2) Along with the evaluation of propellant-spacecraft interactions, a definition of the characteristics of the source of expellant is required. An experimental program is recommended in which the distribution and energy of the ions and neutrals leaving a thruster are measured. Furthermore, during the course of this program the relationships between the angular distribution of the ions and neutrals and thruster parameters such as propellant efficiency, neutralizer voltage, accel-decel ratio, etc., should be established.
- 3) A deep-space vehicle must be able to receive weak signals near the range limit of the telecommunications link; it is thus sensitive to the presence of any local noise sources. In particular, for a spacecraft employing ion propulsion, the locally generated noise may interfere with reception of signals, especially at maximum range. The possible degradation of telecom performance by an electric propulsion system should be thoroughly studied.

- 4) For several reasons it is important to understand the nature and effect of variations, both predictable and unpredictable, in thruster parameters such as thrust level and specific impulse. For example, 1) the degree of dispersion of propulsive parameters affects the design of spacecraft guidance and control systems; 2) thrust modulation of peripheral engines in a modularized prime propulsion system is a possible means of spacecraft attitude control; and 3) relatively large variations in single engine thrust level may be a requirement for power matching. Thus, an experimental program should be implemented which determines in detail the range of fluctuations from some desired nominal of thrust and specific impulse during operation of a complete engine system and which studies the effect of relatively large programmed variations of thrust on other engine system operating characteristics.
- 5) Components and subsystems which are employed in modularized ion propulsion systems and which require development effort include insulating valves, electromagnetic pumps, high voltage isolators, large propellant reservoirs, electrical switching systems, transmission cable, and solar panel power interrogator.
- 6) An extensive and detailed reliability analysis of a complete modularized ion propulsion system is recommended. This analysis, which should include the effects of power matching, would result in a definition of optimum thruster module size for various missions.
- 7) An area where relatively little effort has been expended on SEP vehicles is the coupling dynamics between large solar arrays and the spacecraft. Coupling can occur between the solar array and a) the attitude control system, b) the engine positioner, and c) thrusters, when turned on and off. Also static bending can become a problem.

Dynamics of several SEP spacecraft during the boost phase has already been studied by Hughes. However, once a detailed spacecraft design is specified, the vehicle-autopilot dynamic response to variable frequency and amplitude inputs should be studied. All significant panel vibration modes and sensor characteristics should be included. These interactions, which have potentially great effects upon the choice of configuration and of attitude control techniques, have been almost completely neglected in the present study due to the short schedule.

- 8) The necessary guidance tolerances on closest approach distances to Jupiter should be studied. Thrust misalignment on a SEP spacecraft will occur due to cooling of the vehicle as it moves away from the sun, and to change in temperature gradient across the spacecraft. Calculations show that an average of 0.2 degree

thrust direction error in a plane perpendicular to the ecliptic, during the last 1 AU of thrusting, will cause a 50,000 n.mi. miss at Jupiter. In order to detect this before thrust cutoff, the DSIF must detect 0.003 fps velocity error and/or 5 n.mi. radial distance error. This would indicate the possible requirement for on-board instrumentation to supplement DSIF guidance information.

- 9) An efficient, general optimization program for low thrust interplanetary trajectories is a long range goal. The feasibility of this goal has been greatly enhanced by the demonstration that geometric and planetocentric effects can be included as algebraic boundary conditions in a detailed simulation of the heliocentric phase (Appendix C). Additional substantial valid simplifications in the problem appear possible, and may permit an efficient variational solution to the general problem with existing numerical mathematical techniques. The status of both low thrust mission analysis and the above stated long range goal can best be advanced by continuing to promote physical insight into low thrust trajectories, since valid simplifications are a natural by-product of this activity. For this reason it is recommended that effort be expended to demonstrate with more rigor and generality the behavior of the following parameters in best performance low thrust interplanetary trajectories:

- Thrust attitude
- Specific impulse
- C_3 and C_4 orientation
- Thrust profile (i.e., coast periods)

3. MISSION ANALYSIS

The information in this section completely supersedes the mission analysis given in the final report of the first-phase feasibility study (Reference 1). During the early studies, trajectory analysis was developed by Hughes Aircraft Company, for the first time demonstrating the importance of including departure and arrival planet masses in the mission performance analysis. Mission optimization, including the utilization of nonzero hyperbolic excess departure velocity, was not complete at the time of the previous report, and spacecraft design activity was in a state of flux.

In this report an attempt has been made to evaluate mission performance optimization, provide correlation with spacecraft design concept, and correct errors noted in the previous report. However, it should be stressed that none of the errors in the previous work were of sufficient magnitude to affect any of the significant conclusions of that study.

The ground rules for the Mars orbiter mission and the power profile have both been changed. Based on new or uprated launch vehicles, the design of the Mars orbiter spacecraft and propulsion system has been altered and the analysis extended to cover several launch opportunities. Several other missions have also come under preliminary evaluation.

The purpose of the mission analysis studies performed during this contract was to determine the characteristics of best performance low thrust interplanetary trajectories and the best propulsion system design parameters for specific launch vehicles and missions, and to demonstrate the performance capability available with the use of these low thrust propulsion systems and launch vehicles for a number of interplanetary missions. During this investigation a number of interesting and relevant characteristics of low thrust interplanetary trajectories have been determined. Much of the subsequent discussion of the nature of these trajectories relies on physical understanding and insights, supported by computer simulations of a variety of low thrust interplanetary missions. One of the benefits of this approach is that an understanding of the physical nature of the trajectories involved greatly reduces the probability that any significant trajectory parameter will be overlooked in the analysis.

The optimization of any practical problem requires that a mathematical model be selected, the independent parameters identified, and the values of these parameters that provide the best solution determined. Unfortunately,

the limitations inherent in the mathematical model are often ignored after the analysis has been completed, and the results are invalidly interpreted as applicable in circumstances that violate the original model. In an attempt to avoid this kind of misunderstanding, the assumptions and the independent parameters used in this study are defined below.

The performance analysis has been carried out with the aid of a three dimensional trajectory simulation program (Appendix B). The inputs to the trajectory program (the independent parameters) are launch time, C_3 magnitude and orientation, initial power, weights, all propulsion system parameters, and thrust attitude (as a function of time). Performance tradeoffs were investigated by systematic variation of the independent parameters. Best performance trajectories are defined as those for which variations in the independent parameters do not lead to an increase in vehicle payload. It should be noted that maximization of payload is not the only performance criterion of interest on all missions; for example, it may be advantageous to reduce flight time at the expense of a small payload decrement. The best performance case (as defined above and used herein) provides a standard of comparison in tradeoffs of this kind.

DEFINITIONS

- 1) Injected Weight - Total weight of vehicle immediately prior to first use of the low thrust propulsion system.
- 2) Net Approach Weight - Injected weight of vehicle less propulsion system weight (including solar panels) and fuel.
- 3) Net Orbit Weight - Weight in planetary orbit less unexpended weight of retro-propulsion system.
- 4) Hyperbolic Excess Velocity Attitude - Measured from the outward pointing heliocentric radius vector and, in the ecliptic plane, is positive in the direction of planetary velocities. Out-of-the-ecliptic plane attitude is positive towards celestial north.
- 5) Thrust Velocity Attitude - Measured from the outward pointing heliocentric radius vector and, in the instantaneous trajectory plane, is positive in direction of planetary velocities. Out-of-the-trajectory plane attitude is positive towards celestial north.
- 6) Flight Time - Elapsed duration between the times when the vehicle location is 144 earth radii and at the point of closest approach to the target planet. The elapsed time from near earth orbit to 144 earth radii is approximately 4 to 5 days.

PLANETARY DEPARTURE AND ARRIVAL CONSIDERATIONS

As an example, consider the case of a vehicle on an earth escape trajectory, which has perigee just outside the earth's sensible atmosphere. The specific energy of a vehicle in an inverse square gravitational field is given by the formula:

$$E = \frac{V_{\infty}^2}{2} = \frac{V^2}{2} - \frac{\mu}{r} \quad (1)$$

Since energy is a function of the square of the velocity, a change in velocity (such as that produced by a propulsion system) changes energy much more significantly when the velocity is large (i.e., when the kinetic energy is large).

This effect can be seen by differentiating Equation 1

$$\frac{dV_{\infty}}{dV} = \frac{V}{V_{\infty}} \quad (2)$$

and defining dV_{∞}/dV as an effectivity factor that relates the change in asymptotic velocity at "infinity" to a change in velocity at some other point. Rearranging Equation 1 gives

$$V = \sqrt{V_{\infty}^2 + \frac{2\mu}{r}} \quad (3)$$

Therefore,

$$\frac{dV_{\infty}}{dV} = \sqrt{1 + \frac{2\mu/r}{V_{\infty}^2}} \quad (4)$$

Equation 4 shows that the effectivity factor decreases rapidly as the total energy (hyperbolic excess velocity requirement) increases. For example, consider a spacecraft to be launched on an interplanetary trajectory starting from a 100 n.mi. earth parking orbit [i.e., $2\mu/r = 121 \text{ (km/sec)}^2$]. Substituting this value into Equation 4 leads to Figure 3-1. Figure 3-1 shows that, after some initial gain, velocity addition at infinity with a high specific impulse low thrust device will become more advantageous than an equivalent instantaneous velocity change at perigee using a chemical rocket having an associated lower specific impulse.

As a gross approximation (a more detailed analysis is given in Reference 2) this occurs when

$$\frac{V_{\text{perigee}}}{V_{\infty}} = \frac{I_{\text{sp low thrust}}}{I_{\text{sp high thrust}}} \approx 10 \quad (5)$$

or, from Figure 3-1, when $V_{\infty} \approx 1 \text{ km/sec}$.

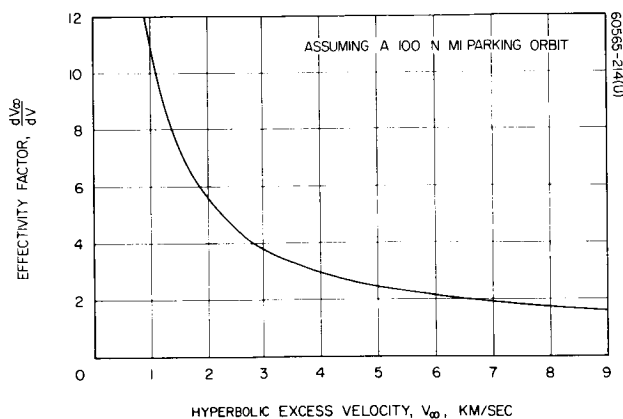


Figure 3-1. Effectivity Factor versus Hyperbolic Excess Velocity

2 km/sec could be expected to provide best performance with the type of solar powered, low thrust ion propulsion system proposed in this study — a conclusion that has been verified by subsequent trajectory studies. The best values of these velocities depend upon the mission requirements, launch opportunity, and vehicle characteristics. Simulation has indicated that the best value of hyperbolic excess velocity from the earth lies between 1 and 2.5 km/sec for most interplanetary missions.

INTERPLANETARY TRAJECTORY GROUPS

For ease of identification and discussion it is convenient to divide interplanetary trajectories into two groups on the basis of fundamental trajectory characteristics. This classification applies not only to low thrust, but also to high thrust (all chemical) trajectories. Recognition of the characteristics of these groups facilitates insight into many of the aspects of low thrust interplanetary trajectories and indicates those parameters whose variation may most logically be explored for the purpose of performance improvement.

Consider high thrust trajectories for any mission for which impulsive velocity additions (chemical velocity additions will be considered to be impulsive in the following discussion) exist only at the initial and (possibly) at the terminal points (e.g., a Hohmann transfer between circular orbits). If a low thrust propulsion system that provides an infinitesimal acceleration is utilized during the heliocentric transfer phase, the changes in trajectory characteristics (launch date, arrival date, C_3 magnitude and direction, etc.) will also be infinitesimal. Successively increasing acceleration (with a fixed thrust attitude time history) produces a family of trajectories having common characteristics. These families (a different family is generated by each thrust attitude profile considered) of trajectories will all be defined as Group I

This general principle of increased sensitivity of energy to velocity changes at high kinetic energy is also true in the general multibody situation, although the formulation is more complex. These same arguments and conclusions apply qualitatively to retro-propulsion at the target planet, and as a result best performance trajectories have nonzero approach velocities at the target planet.

An analysis of the problem with the use of energy relationships (Reference 2) indicated that for earth-Mars trajectories departure and approach velocities of 1 or

trajectories in the subsequent discussion. Group I high thrust trajectories have conventionally been subdivided into the types and classes defined below.

If the orbits of two planets were circular and coplanar, the minimum energy two-impulse interplanetary transfer trajectory would be a Hohmann transfer with a 180 degree heliocentric central angle (see Figure 3-2).

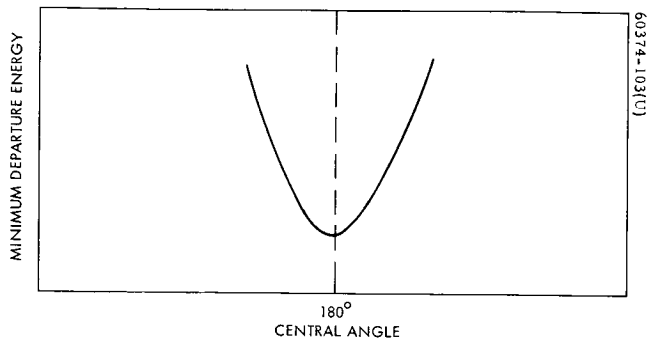


Figure 3-2. Central Angle versus Minimum Departure Energy

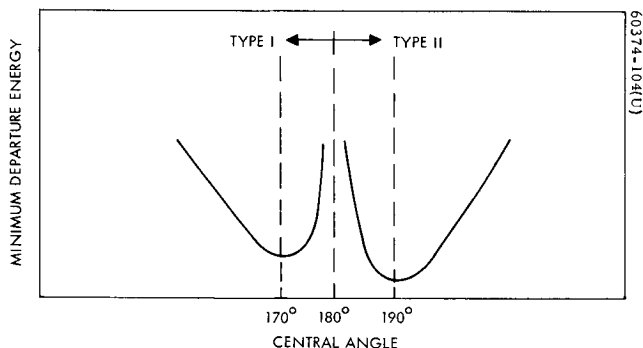


Figure 3-3. Central Angle versus Minimum Departure Energy

Since the planetary orbits are not coplanar, the 180 degree transfers require very high energies because the plane of the transfer trajectory must be nearly perpendicular to the orbital plane of the departure planet, except for the special case where departure and arrival occur at the node of the orbit planes. For the actual planetary geometry, there are two relative minimum values of required escape energy, one on either side of the point where the central angle is 180 degrees (Figure 3-3). The solutions with central angles less than 180 degrees are Type I and the solutions with central angles greater than 180 degrees are Type II. The definition of class is shown in Figure 3-4. For each type of trajectory, there may be two solutions for a given departure energy. The solution with the shortest time of flight (central angle) is Class I, and the longest is Class II. The minimum departure energy for each type is at the juncture of the two classes and represents a trajectory that is tangent (in plane) to the orbit of the target planet. The implications of this tangency condition are discussed later in more detail.

It should be noted at this point that either Type I or Type II high thrust trajectories can be approached as a limit as low thrust acceleration goes to zero. As the low thrust acceleration increases, the distinction between Type I and Type II trajectories vanishes because the use of low thrust propulsion during the trajectory alleviates the geometric difficulty associated with a two impulse (Group I) high thrust trajectory in the vicinity of a 180 degree heliocentric transfer angle and produces a situation similar to that which would exist if the initial orbits were coplanar (although the best transfer

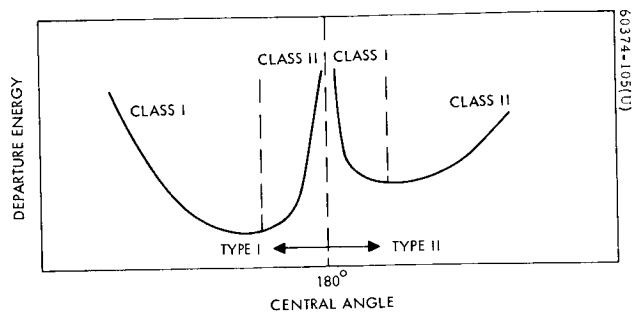


Figure 3-4. Heliocentric Angle versus Departure Energy

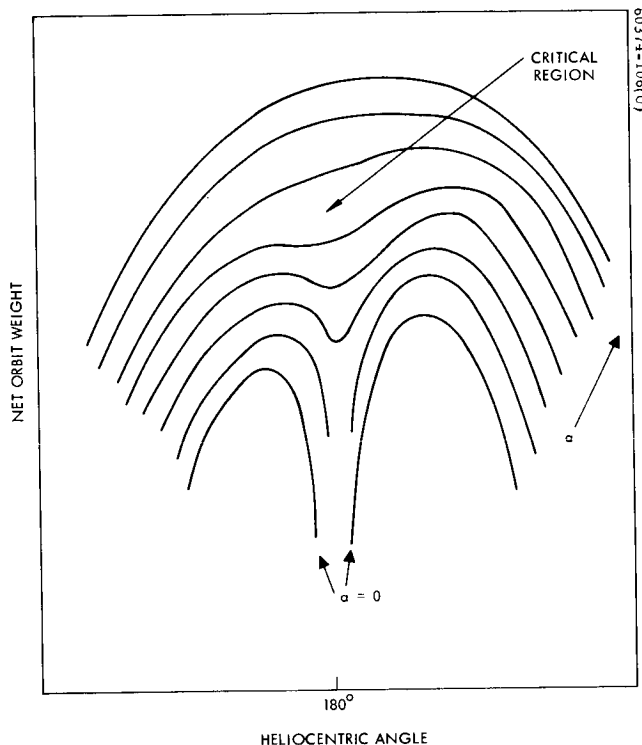


Figure 3-5. Heliocentric Angle versus Net Orbit Weight

angle is not necessarily 180 degrees unless both orbits are circular). This situation is shown schematically in Figure 3-5 for the general planetary orbiter mission.

Those low thrust trajectories that are not Group I will be defined in this report as Group II. Even though the definition is given in this way, all Group II trajectories are related to high thrust trajectories having impulsive velocity additions at intermediate points on the trajectory. The most obvious example of a Group II trajectory is a low thrust spiral. Many small impulsive velocity additions are required to approximate this trajectory with a high thrust vehicle.

All high thrust trajectories are actually special cases within the low thrust trajectory groups, and in a sense the low thrust propulsion system is a midcourse stage which is an adjunct to, but not in conflict with, high thrust propulsion. Neither the definition of the low thrust trajectory groups nor the qualitative characteristics of trajectories therein, are dependent upon the use of solar power as opposed to nuclear or radioisotope power sources. However, it may be noted that while low thrust systems may be defined as those in which the energy is not stored in the fuel, this definition is not as relevant for mission analysis as for propulsion purposes. For mission analysis purposes, the maximum acceleration attainable (or an equivalent variable) is the relevant factor and should comprise

or be included in the definition. A qualification of this type in the definition could be important in the future, but is academic at this time because the maximum attainable acceleration is currently about 10^{-4} g (unless specific

impulse is reduced to an unacceptably low value) and is not likely to be increased substantially (i. e., by more than a factor of five) without an unforeseen breakthrough.

The work done on this contract has demonstrated the fundamental characteristics of best performance Group I trajectories. In contrast, both high and low thrust Group II trajectories are, in general, considerably more complicated, more diverse in their nature, and more difficult to investigate thoroughly; and for these reasons are less well understood. Known characteristics of both groups of trajectories are summarized in the following subsections.

Group I Trajectories

As mentioned previously, all best performance trajectories have significant hyperbolic excess velocities with respect to departure and target planets. In addition, best performance Group I trajectories typically have:

- A heliocentric transfer angle of the order of 180 degrees
- One launch opportunity each synodic period
- Essentially continuous thrusting (no coast periods)
- A preferred (characteristic) flight time
- Near tangency (in-plane) of transfer trajectories to planetary orbits at departure and arrival
- In-plane thrust attitude nearly parallel (or antiparallel) to vehicle velocity
- Asymptotic departure velocity nearly parallel to target planet orbit plane.

The following discussion will describe and qualify these typical characteristics in detail.

One of the characteristics of low thrust propulsion systems is a high weight per pound of thrust. As a result of this the propellant weight that can be expended is a small percentage of total propulsion system weight (e.g., about 30 percent for a Mars orbiter) unless the thrusting time is very long (or unless the specific impulse is too low to be competitive with all-chemical systems), and the velocity added with the low thrust system is correspondingly limited. Under these circumstances the reduction of thrusting time by the use of coast periods degrades this already undesirable situation and results in a performance loss. Coasting arcs may exist on trajectories having accelerations substantially above that required in the best performance case. This relatively high acceleration is used for trajectory shaping and is required when 1) the departure time is considerably removed from that which gives best performance, or 2) when hyperbolic excess velocity at departure or arrival is constrained below that value which gives best performance. (Trajectories with constraints on hyperbolic excess velocity at departure and arrival have previously been investigated extensively, e.g., Reference 5, and the results obtained from these studies illustrate the above statements in that coast periods and relatively large accelerations were required and/or desired.) A terminal coast period will exist on best performance trajectories when approach velocity does not affect payload (e.g., flybys having no deboost at target planet).

For each set of values of mission, vehicle, and propulsion system parameters there is a characteristic (preferred) flight time, and flights having a greater or lesser duration show a performance degradation (see Figure 3-6). A zero-power case is the familiar high thrust trajectory and gives a minimum transfer time. As low thrust acceleration (power) is increased, the hyperbolic excess velocities at the launch and target planets are decreased, leading to increased characteristic flight time. In general, an increase in acceleration on best performance trajectories leads to both earlier departure and later arrival dates. The characteristic flight time will vary for different launch opportunities if either the launch or target planet has a significant orbital eccentricity.

The optimality of Hohmann transfers between coplanar circular orbits is well known, and it has been shown (Reference 3) that in the general case high thrust orbit transfers that require minimum impulsive velocity additions are, for all practical purposes, also tangent-tangent transfers (i.e., transfers in which the in-plane component of impulsive velocity is tangent to the orbital velocity at departure and arrival). This result is intuitively satisfying in that a tangency condition would logically be expected to minimize velocities relative to the departure and target planets. Tangent-tangent heliocentric transfers between planetary orbits are also desirable for low-thrust Group I

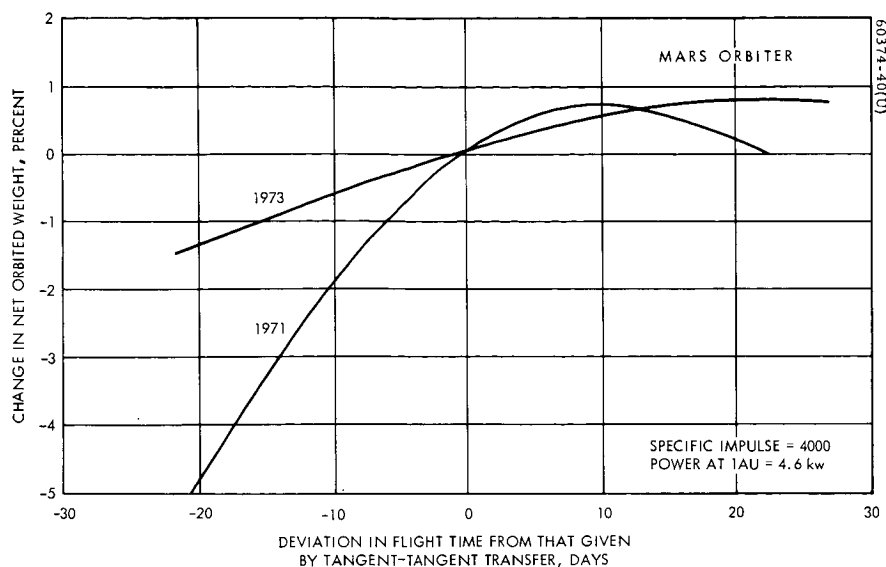


Figure 3-6. Mars Orbiter, Change in Net Orbiter Weight versus Deviation in Flight Time From that Given by Tangent-Tangent Transfer

transfer trajectories because the kinematic problem is similar. Figure 3-6 illustrates this for the 1971 and 1973 Mars orbiters where variation of flight time from that required for tangent arrival does not result in a significant increase in net orbit weight.

It has also been shown (Figures 3-7 and 3-8) that modification of the direction of the in-plane component of hyperbolic excess velocity from the tangent departure condition changes the preferred launch date, but has little effect on performance (launch windows can be obtained with very small performance cost by following this approach). Imposing the constraint that transfer trajectories be of the tangent-tangent type is therefore the practical equivalent of selecting the flight time that provides best performance. This fact has been used to advantage by eliminating flight time as an independent variable in performance tradeoff studies, thereby substantially reducing the time and expense required for trajectory simulation.

Group I trajectories are bounded on one extreme by the zero acceleration case (all chemical system) and on the other extreme (for practical purposes) by that acceleration above which continuous thrust tangent-tangent transfers are no longer possible (since beyond this latter point the increase in approach velocity at the target planet causes a performance loss).

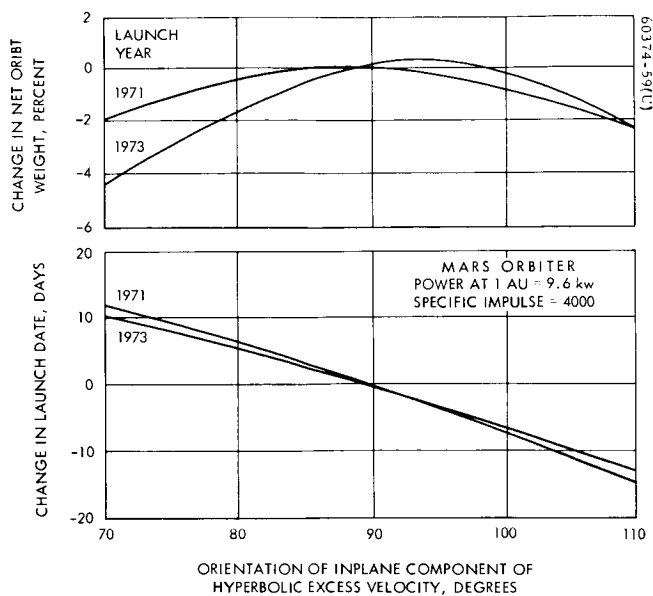


Figure 3-7. Mars Orbiter Change in Net Orbit Weight, and Change in Launch Date versus Orientation of In-plane Component of Hyperbolic Excess Velocity

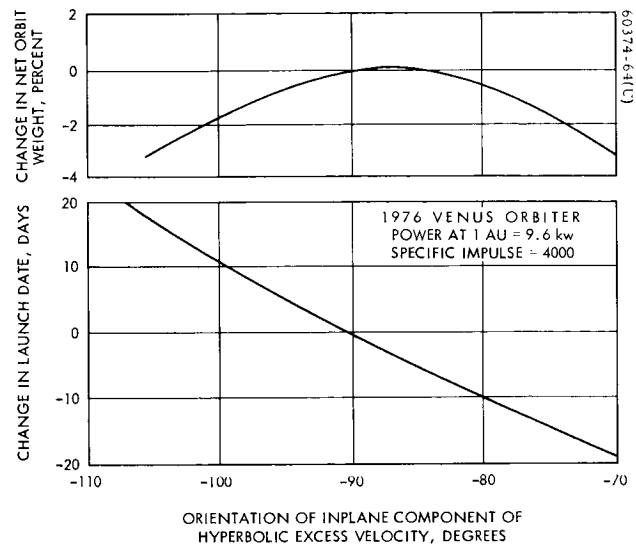


Figure 3-8. 1976 Venus Orbiter Change in Net Orbit Weight, and Change in Launch Date versus Orientation of In-plane Component of Hyperbolic Excess Velocity

For best performance solar powered earth-Mars trajectories this maximum initial acceleration level can be as low as $1.4 \times 10^{-5} g$, although the value is considerably higher for some launch opportunities. It is interesting to note that in nondimensional terms this is only about 2 percent of the local gravitational attraction of the central force field. A best performance Group I trajectory always exists between these two extremes when high and low thrust propulsion systems of the type considered in this study are used. Characteristics of all of the trajectories in this region differ quantitatively, but not qualitatively.

It can be shown that changes in thrust attitude orientation have the greatest effect on trajectory characteristics when done as early in the trajectory as possible. The logical extension of this result leads to the (subsequently verified) conclusion that trajectory shaping is considerably more sensitive to changes in the direction and magnitude of the initial hyperbolic excess velocity than to changes in the low thrust attitude time history. The basic character of Group I trajectories is therefore largely determined by escape conditions from the departure planet rather than by steering with the low thrust propulsion system. The low thrust propulsion system is utilized primarily as an energy changing device (this is why reducing acceleration causes the high thrust trajectory to be approached as a limit), and for this

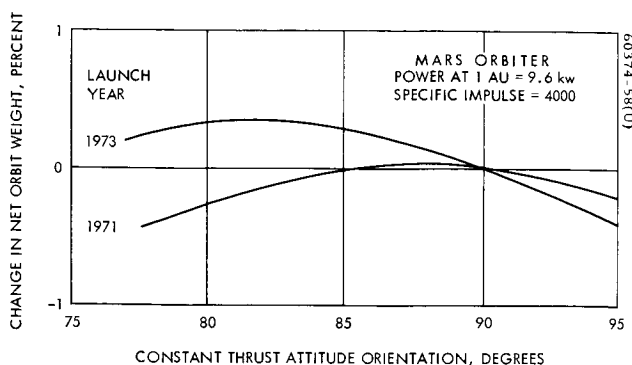


Figure 3-9. Mars Orbiter, Change in Net Orbit Weight versus Constant Thrust Attitude Orientation

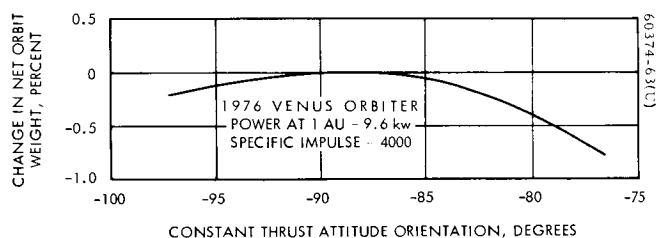


Figure 3-10. 1976 Venus Orbiter, Change in Net Orbit Weight versus Constant Thrust Attitude Orientation

reason the in-plane component of thrust acceleration should be directed approximately along (or counter to) the vehicle velocity. Performance is not sensitive to small changes in thrust attitude time history. For trajectories to the near planets (Mars and Venus), a thrust attitude held constant with respect to the sun-vehicle line will provide essentially identical performance for a range of values of the constant attitude (see Figures 3-9 and 3-10). For these reasons thrust attitude orientation is not, within broad limits, a significant parameter in best performance Group I trajectories; its use as a control parameter for guidance does not result in an important performance loss because the required variation is well within these limits.

The line of nodes of the trajectory and target orbit planes must lie at the terminal point of the transfer trajectory at the time of arrival at this point. This condition may be accomplished by introducing an out-of-ecliptic component of the hyperbolic excess velocity and/or by out-of-plane low thrust attitude control (launch date and flight time are essentially constrained by inplane dynamics). The exact relationship between these quantities that provides best performance in

the general case is not known. This uncertainty has not presented a practical problem on the trajectories that have been investigated because the performance cost associated with the out-of-plane kinematics is relatively small and insensitive in the regions of the best performance trajectories (Figures 3-11 and 3-12).

It will be convenient to define the critical region as the acceleration region in which the second relative maximum performance peak vanishes (see Figure 3-5). The low thrust acceleration level in the critical region depends on flight time, but is primarily a measure of the geometric difficulty caused by the dihedral angle between the launch and target planet orbit planes and the location of the launch planet with respect to the line of nodes of these two planes at the time of launch.

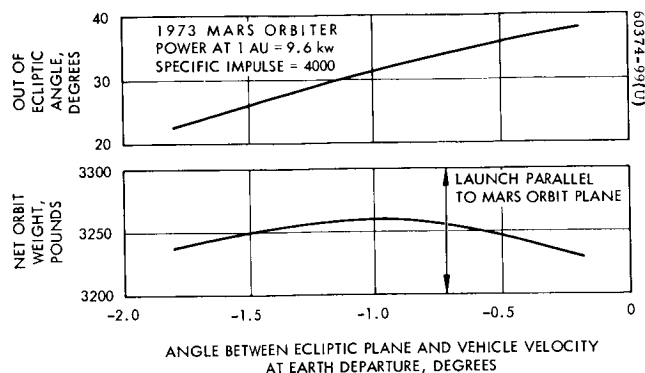


Figure 3-11. 1973 Mars Orbiter, Out-of-Ecliptic Angle and Net Orbit Weight versus Angle Between Ecliptic Plane and Vehicle Velocity at Earth Departure

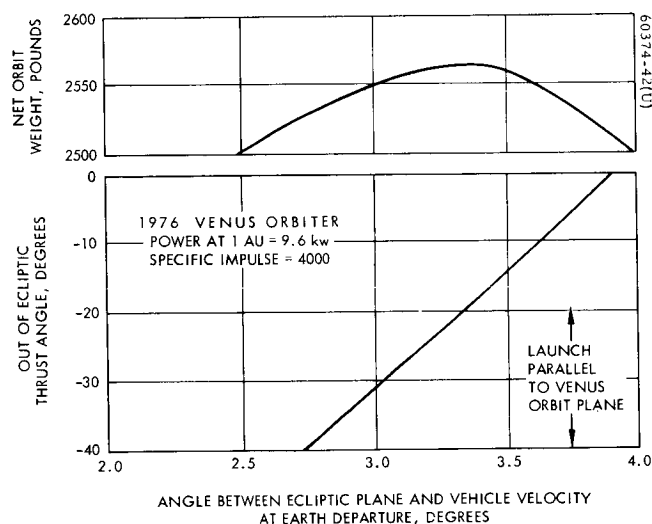


Figure 3-12. 1976 Venus Orbiter, Net Orbit Weight and Out-of-Ecliptic Thrust Angle versus Angle Between Ecliptic and Vehicle Velocity at Earth Departure

Not all the trajectories shown in Figure 3-5 are possible for every interplanetary mission. For example, if the dihedral angle between the initial vehicle trajectory and target planet orbit planes is large, the acceleration level in the critical region may be above that which can be used for tangent-tangent transfers. In this case, both Type I and Type II relative maximum performance, low thrust trajectories will exist (see Figure 3-5).

If the low thrust acceleration on any mission is above the value in the critical region, the low thrust propulsion system can efficiently (with an average out of plane thrust angle less than approximately 30 degrees) reduce the dihedral angle between the initial vehicle trajectory and the target planet orbit plane substantially by the time of arrival at the target planet, leading to a reduction in the out-of-plane component of approach velocity. If the out-of-plane thrust attitude is held constant in this case, it may be shown that selecting an out-of-the-ecliptic hyperbolic excess velocity component that makes the total vehicle velocity at departure nearly parallel to the target orbit plane approximates best performance (see Figures 3-11 and 3-12). This situation exists because the reduction in in-plane thrust acceleration is small when the out-of-plane thrust attitude is rotated out of plane by less than 30 degrees, and the inplane components of the trajectory are therefore not greatly disturbed. If the low thrust acceleration is below the value in the critical region, a substantial performance loss

results when the total vehicle velocity at departure is parallel to the target plane because larger out-of-plane thrust attitudes are required (Figure 3-13). For these cases it is necessary to increase the dihedral angle between the initial vehicle trajectories and target planet orbit planes (with respect to that which would exist for parallel launch) in order to efficiently obtain a nodal arrival condition. The out-of-plane component of approach velocity will be significant for these trajectories.

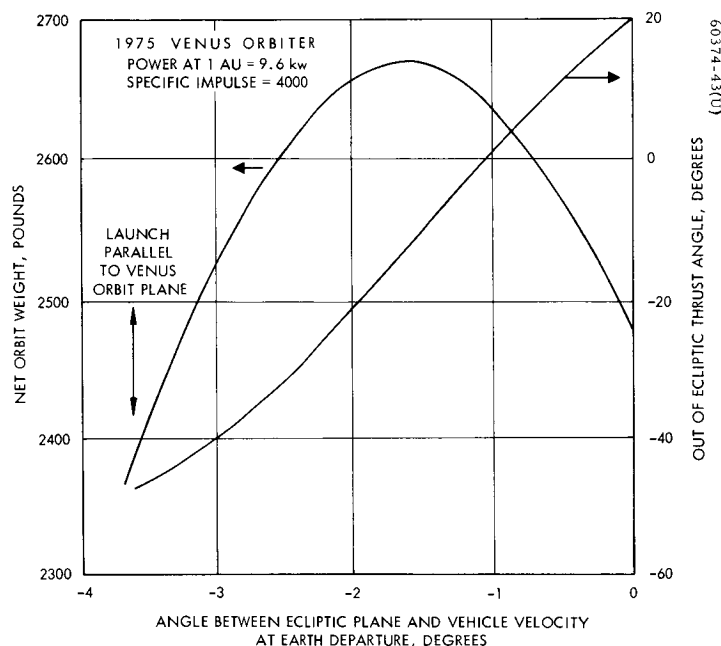


Figure 3-13. 1975 Venus Orbiter, Net Orbit Weight and Out-of-Ecliptic Thrust Angle versus Angle Between Ecliptic Plane and Vehicle Velocity at Earth Departure

Simulation has shown that the acceleration in the critical region is below that of the best performance case for all the Mars launch opportunities. The acceleration in the critical region is substantially above that which exists on best performance transfer trajectories to Venus on four of five launch opportunities because of the greater inclination of Venus to the ecliptic plane (one of the launch opportunities occurs when the earth is near the node between Venus and the ecliptic plane).

It is worth noting that those launch opportunities with unfavorable geometry (i.e., which have high accelerations in the critical region) are also the most inefficient with high thrust propulsion because of this same unfavorable geometry, emphasizing again that the high thrust trajectories are merely special cases of general Group I trajectories.

Group II Trajectories

For Group II trajectories, the heliocentric transfer angle is usually 270 degrees or more, and all trajectories having heliocentric transfer angles in excess of 360 degrees are of the Group II type.

More than one type of Group II trajectory exists for any mission. For example, increasing the included heliocentric transfer angle by approximately 360 degrees will produce another Group II trajectory of a different type; different types of Group II trajectories can also be generated by other changes. Each of the types of Group II trajectory has a best performance case and preferred values of propulsion system parameters. The type of Group II trajectory that provides the best performance is one which has an infinitesimal acceleration and an infinite flight time. For this reason missions utilizing Group II trajectories must contain an explicit or implicit flight time limitation as a mission constraint.

On best performance Group I trajectories, the flight time is a dependent quantity and is a function of vehicle and mission parameters. In general, this is not true for Group II trajectories. A sufficiently large increase in permissible flight time will always result in improved performance for a (nonzero acceleration) Group II trajectory. Partly because of this, coasting arcs often exist on efficient Group II trajectories.

Orientation of the initial hyperbolic excess velocity tangent to earth orbital velocity and low thrust acceleration orientation along (or opposed to) the vehicle velocity appears to provide best performance for some Group II trajectories, but this is probably not true in general.

Group II trajectories exist for all missions for which Group I trajectories are possible. For the same mission, the efficient Group II trajectories have longer mission time and better performance (net weight) characteristics than Group I trajectories. For earth-Mars trajectories the practical disadvantage of the greatly increased flight time for Group II trajectories outweighs the accompanying relatively small performance gain. On the other hand, the very high acceleration and/or hyperbolic excess velocities required on Group I earth-Mercury trajectories cause very poor performance, and a Group II trajectory appears more attractive.

The longer flight times of the Group II trajectories permit considerably larger plane changes to be made with the low thrust propulsion system. Once again this is not required for earth-Mars trajectories, but is particularly advantageous for earth-Mercury trajectories.

Group II trajectories are so complex that a definitive analysis of the general case does not appear likely in the foreseeable future, and a specific analysis for each mission is therefore indicated.

SIMULATION PROCEDURE

Performance analyses have been carried out with the aid of a three dimensional trajectory program that simulates the motion of a point mass subject to forces due to thrust, the sun, earth, Jupiter, target planet, and solar pressure acting on the solar panel (see Appendix B). The computation has been simplified by starting the integration in heliocentric coordinates with the vehicle placed at 144 earth radii. The initial position and velocity of the vehicle are those that would be obtained by boosting from an initial parking orbit of 185 km altitude above the earth's surface. The effects of this approximation have been shown to be relatively insignificant for performance analysis purposes (see Appendix C).

In each synodic period a unique C_3 magnitude and departure date from a launch planet are required for arrival at a given target planet if other trajectory parameters (e.g., thrust attitude orientation, thrusting times, C_3 orientation) and all propulsion system parameters (e.g., specific impulse, efficiency, initial power) are held fixed. The trajectory simulation program searches for the values of C_3 magnitude and launch date that are required to complete the mission with the preselected values of the other parameters. Variation of these other parameters to maximize payload is then conceptually straightforward; the solution is practical to implement because performance is well behaved and insensitive to variations near the best performance case.

All trajectories included herein are three dimensional in both computation and control. For some missions, out-of-ecliptic considerations materially alter the performance that would be obtained from two-dimensional simulation.

TRAJECTORY SIMULATION RESULTS

The input data of Appendix A was used in the trajectory program described in Appendix B to generate performance information for several low thrust interplanetary missions. The Titan III M was used as a launch vehicle for the Mars orbiter, Venus orbiter, and Mercury. The Atlas (SLV-3C)/Centaur was used as a launch vehicle for the Jupiter flyby, out-of-ecliptic probe, and solar probe. Injected weight capabilities for these vehicles are given in Appendix A. During the course of the overall task, complete ion propulsion systems and spacecraft designs for the Mars orbiter and Jupiter flyby missions were to be furnished. Since, because of time limitations, the design effort was initiated prior to final completion of trajectory and mission studies, it was necessary to select the design points for the propulsion system on the basis of preliminary trajectory information. A 17 kw ion propulsion system with the Atlas (SLV-3C)/Centaur and a 9.6 kw system with the Titan III M were selected initially for the Jupiter flyby and Mars orbiter missions, respectively. The more complete and detailed trajectory and mission studies that have subsequently been completed have shown no alternative designs that appear more attractive from a system capability standpoint.

The use of the 9.6 kw and 17 kw propulsion systems on a variety of other interplanetary missions has also been simulated, although detailed spacecraft designs and payload feasibility studies have not been completed for these additional missions. While the 9.6 and 17 kw designs are not necessarily the best choices for these alternative missions, they do, in all cases, provide a performance capability that is attractive in comparison to chemical propulsion.

Mars Orbiter

The design launch vehicle for the Mars orbiter mission is the Titan III M with a 9.6 kw ion propulsion system. A constant attitude, continuous thrusting Group I trajectory has been chosen for the Mars orbiter mission. The trajectory generation procedure used selects tangent-tangent transfers (the practical equivalent of best flight time trajectories), thereby eliminating the need to include flight time in the parameter search.

One of the primary objectives in the Mars orbiter mission studies was the selection of a single propulsion system design that could be used with the same launch vehicle on several successive Mars launch opportunities. For study purposes, the five launch opportunities occurring in the 1970's were selected. The use of a single propulsion system design for these launch opportunities is not only obviously attractive, but also appears very practical because the resulting performance cost is small. Although the other two launch opportunities in the 15 year cycle have not been investigated, there is every reason to believe that they can also be performed with the same vehicle. A compromise power level of 9.6 kw (providing an initial acceleration of about 1.2×10^{-5} g) was selected, and the trajectory studies indicate no obviously superior choice. The following nominal conditions were selected for all launch opportunities in the study:

Thrust attitude = 90 degrees

Hyperbolic excess velocity orientation = 90 degrees

Power at 1 AU = 9.6 kw

Specific impulse = 4000

Thrust at 1 AU = 65 millipounds

Propulsion system specific weight = 75 lb/kw

Titan III M launch vehicle

High and low thrust propulsion characteristics are given in Appendix A. Table 3-1 gives a summary of other nominal conditions (performance losses due to guidance and launch window requirements are not included) for the five launch opportunities. Figures 3-7, and 3-14 through 3-20 show the effects of variation of trajectory parameters.

TABLE 3-1. MARS ORBITER NOMINAL CONDITIONS SUMMARY*

Launch Oppor- tunity	Weight, pounds										$C_3 = \left(\frac{\text{Km}}{\text{Sec}}\right)^2$	Approach Velocity, km/sec	Flight Time, days	Departure Date
	Injec- tion	Net Approach	Fuel	1000 x 2000 km Orbit		5000 x 5000 km Orbit		1000 x 1000 km Orbit						
				Net Orbit	Retro Orbit	Net Orbit	Retro Orbit	Net Orbit	Retro Orbit					
1971	5554	4587	247	3343	1244	2525	2062	2319	2268	4.23	1.96	231	9 May	
1973	5013	4020	273	3255	765	2542	1478	2257	1763	6.93	1.14	293	19 July	
1975	5301	4232	349	3513	719	2747	1485	2429	1803	5.49	0.90	404	17 August	
1977	5586	4475	391	3641	834	2843	1632	2524	1951	4.07	1.10	432	7 Sep- tember	
1979	5786	4672	394	3693	979	2861	1811	2554	2118	3.07	1.38	407	2 Octo- ber	

*Titan III M vehicle - 9.6 kw 4000 I_{sp} engine.

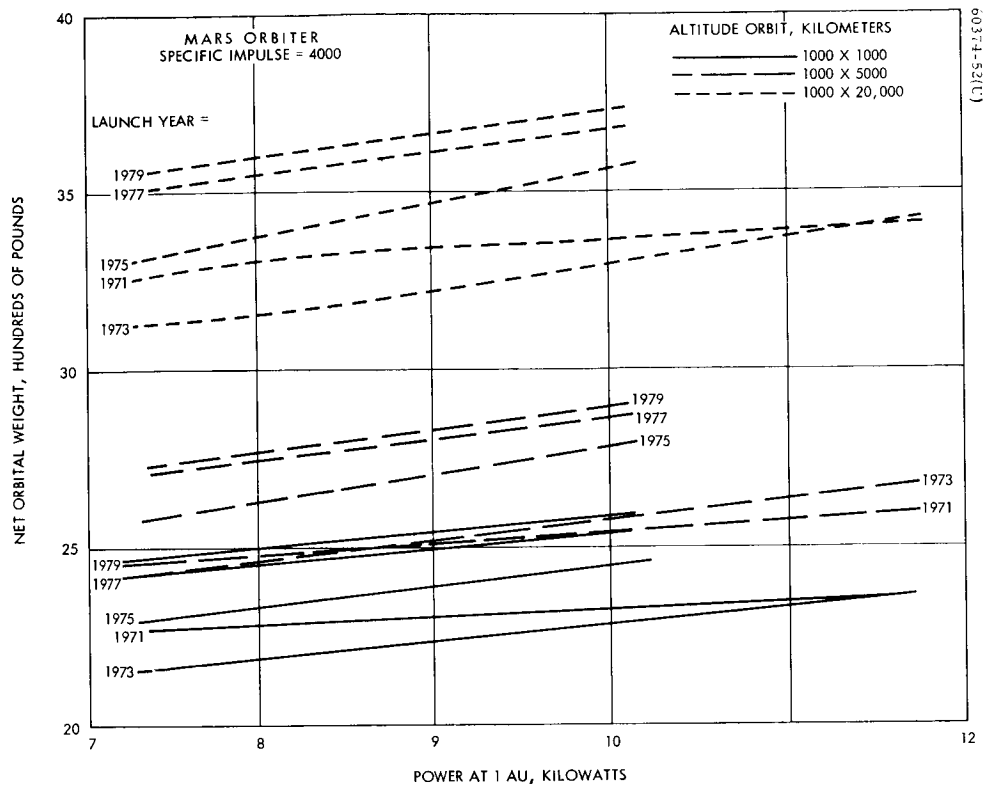


Figure 3-14. Mars Orbiter, Net Orbital Weight versus Power at 1 AU

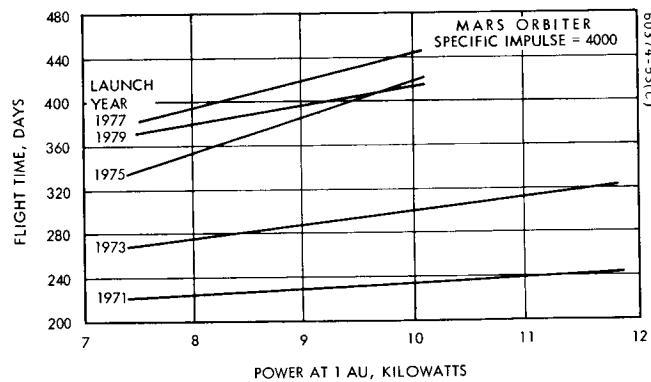


Figure 3-15. Mars Orbiter, Flight Time versus Power at 1 AU

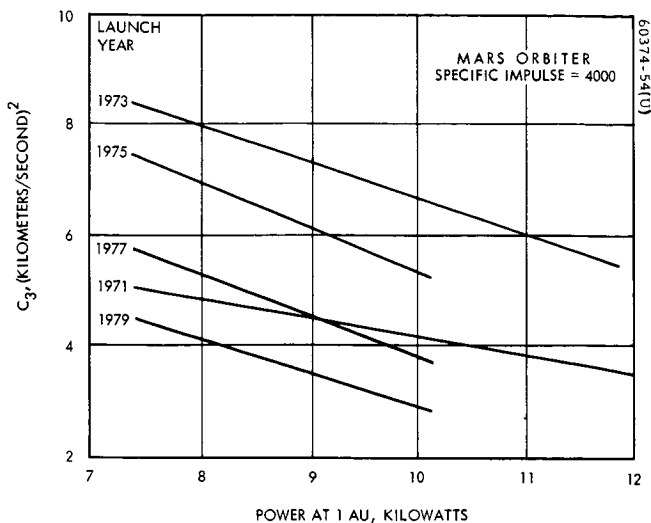


Figure 3-16. Mars Orbiter, C_3 versus Power at 1 AU

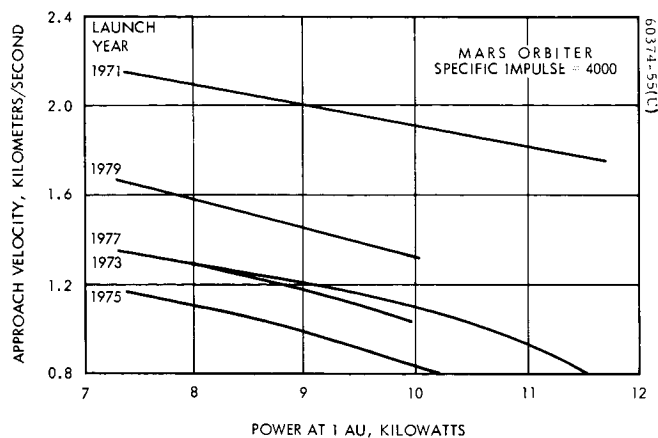


Figure 3-17. Mars Orbiter, Approach Velocity versus Power at 1 AU

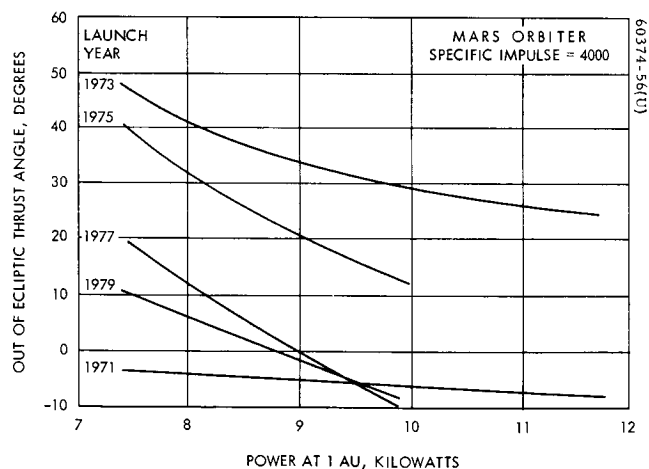


Figure 3-18. Mars Orbiter Out-of-Ecliptic Thrust Angle versus Power at 1 AU

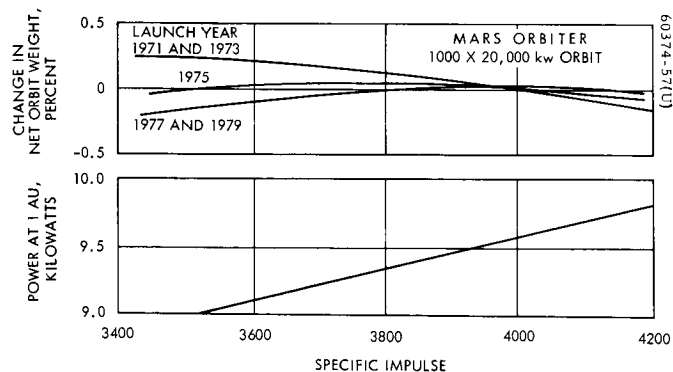


Figure 3-19. Mars Orbiter, Change in Net Orbit Weight and Power at 1 AU versus Specific Impulse

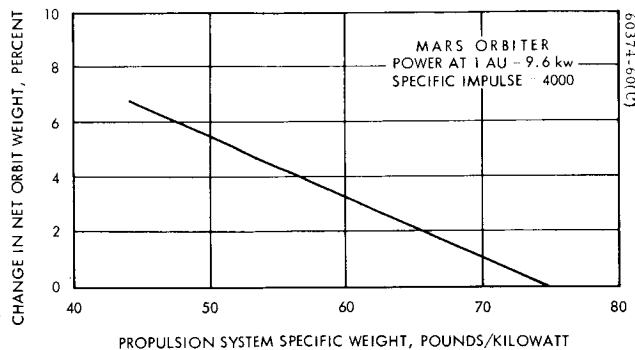


Figure 3-20. Mars Orbiter, Change in Net Orbit Weight versus Propulsion System Specific Weight

The variation of net orbit weight with launch opportunity shown in Table 3-1 is much smaller than that obtained with all chemical systems. This is due to the mitigating effect of low thrust propulsion on the less favorable geometry that exists for, and to the relative advantage of low thrust propulsion with the longer flight time required for, the launch opportunities in the late 1970's.

The increase of net orbit weight with increased power, shown in Figure 3-14, is due primarily to the increase in flight time (Figure 3-15) and the resulting increase in velocity supplied by the low thrust propulsion system. Note that the

payload and flight time for the remaining two launch opportunities in the 15 year cycle may be expected to lie between the values given for 1971 and 1979. Both C_3 and approach velocity (Figures 3-16 and 3-17) increase as power level drops. As discussed previously, these values approach the high thrust solution as a limit (including the direction as well as the magnitude of C_3).

Figure 3-18 shows out-of-the-ecliptic thrust angle as a function of power (assuming injection with the total vehicle velocity parallel to Mars orbit plane). In 1971 the earth is close to a node of Mars orbit and the ecliptic plane at launch. The 1971 transfer trajectory is therefore nearly coplanar with Mars orbit, resulting in very small out-of-plane thrusting requirements. This is not the case with the other launch opportunities (the lack of out-of-plane thrusting requirements at 9 kw in 1977 and 8.8 kw in 1979 indicates an initial nodal intercept, not coplanar conditions) and as a result the required out-of-ecliptic thrust angle increases rapidly as power is decreased. As previously discussed, this leads to the condition where injection with the total vehicle velocity parallel to Mars orbit plane no longer provides best performance. The best out-of-plane C_3 orientation does, of course, approach the all chemical orientation as a limit.

Figure 3-19 shows the percent change in net orbit weight as specific impulse is varied at constant initial acceleration. Although the best value does have some dependence on launch opportunity (and almost certainly upon the final orbit and other vehicle, propulsion system, and mission assumptions as well), the performance differences are insignificant.

Figure 3-9 shows the performance change as different values are selected for the in-plane component of constant thrust attitude orientation. It is clear from this figure that the thrust attitude orientation may be held at any value in the vicinity of 90 degrees without incurring a significant performance penalty.

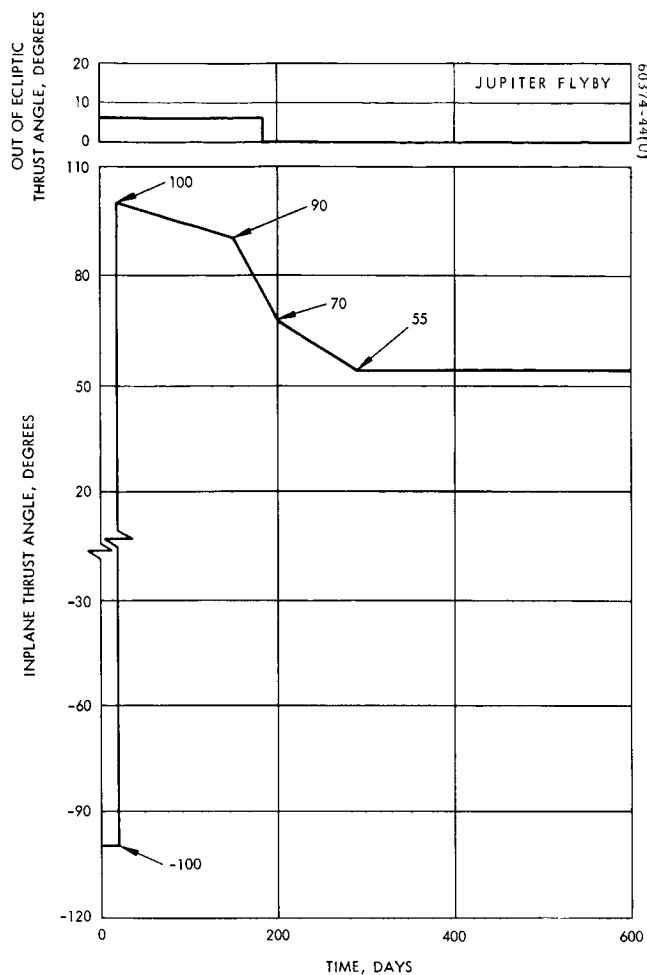


Figure 3-21. Jupiter Flyby, Out-of-Ecliptic Thrust Angle and In-plane Thrust Angle versus Time

provide enough performance advantage to make up the additional weight required (although this may not be true in the general case), and because of design uncertainties and study time limitations, this approach was not considered further. Preliminary investigations also indicated that a thrust attitude oriented along (or counter to) the vehicle velocity appears to provide best performance. Therefore, a thrust attitude orientation that approximates this profile was chosen and is given in Figure 3-21 (the 55 degree limit after 300 days is due to sensor bias limitations on the Canopus tracker).

The initial thrust attitude orientation is approximately opposed to the earth's orbital velocity (as is the hyperbolic excess velocity at departure), thereby permitting the vehicle to remain inside 1 AU for approximately

Figure 3-7 indicates that a launch window of approximately 30 days can be obtained at a cost of 2 to 4 percent in net orbit weight by variation of the orientation of the in-plane component of hyperbolic excess velocity. This performance loss is smaller than for high thrust trajectories by at least a factor of two. Figure 3-20 shows that propulsion system specific weight for the Mars mission is not critical. This insensitivity results from the relatively small percent of total spacecraft weight contributed by the propulsion system.

The effects of the uncertainty in the power curve are discussed in Appendix A.

Jupiter Flyby

The Jupiter flyby design launch vehicle is the Atlas (SLV-3C) / Centaur with the 17 kw low thrust ion propulsion system. The thrust attitude orientation system can be designed for fixed or possibly (at a system weight and reliability penalty) for variable attitude capability. Some flexibility is available with the "fixed" attitude system by varying the bias on the reference sensor signals for the Canopus tracker and sun-seeker. Preliminary investigations indicated that a variable attitude design would not

140 days. Preliminary studies indicated that for a given flight time this Group II trajectory has a lower power requirement than a Group I direct transfer trajectory and a resulting performance improvement, and for this reason Group I trajectories were not further investigated. As indicated previously, a maximum flight time constraint is required for Group II trajectories. A 900 day maximum flight time was selected by Jet Propulsion Laboratories as a ground rule for the study.

The following nominal conditions were selected:

Thrust attitudes as given in Figure 3-21

Hyperbolic excess velocity orientation = -90 degrees

Thrust termination time = 540 days

Flight time = 900 days

Power at 1 AU = 17 kw

Specific impulse = 3500

Thrust at 1 AU = 121 millipounds

Propulsion system specific weight = 65 lb/kw

Atlas (SLV-3C)/Centaur launch vehicle

These inputs give the following weights and trajectory characteristics:

$$C_3 = 1.22 \text{ (km/sec)}^2$$

$$\text{Initial acceleration} = 3.8 \times 10^{-5} \text{ g}$$

Launch Date = 1 December 1973

Approach velocity = 7.086 km/sec

Net approach	645 pounds
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Propulsion system	1105 pounds
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Fuel	865 pounds
------	------------

Injected weight	2615 pounds
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Perturbations were made about the nominal input parameters and indicate in all cases that the values chosen approximate best performance (within the constraint of flight time).

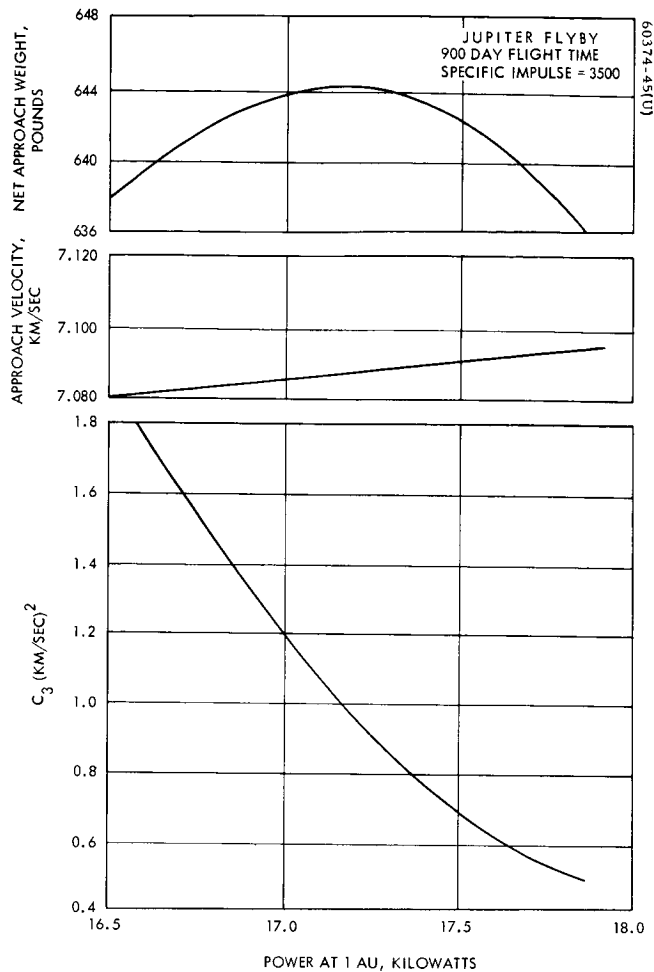


Figure 3-22. Jupiter Flyby, Net Approach Weight, Approach Velocity, and C_3 versus Power at 1 AU

from a propulsion system life or reliability standpoint to do so, the thrust duration could be reduced somewhat without incurring a significant performance penalty.

Figure 3-24 shows the maximum performance available as specific impulse is varied at constant initial acceleration. Although 3500 is approximately the best value, the performance is not sensitive. It should also be noted that the total power requirement does not decrease in direct proportion to specific impulse because of the rapid decrease of engine efficiency at the lower values of specific impulse.

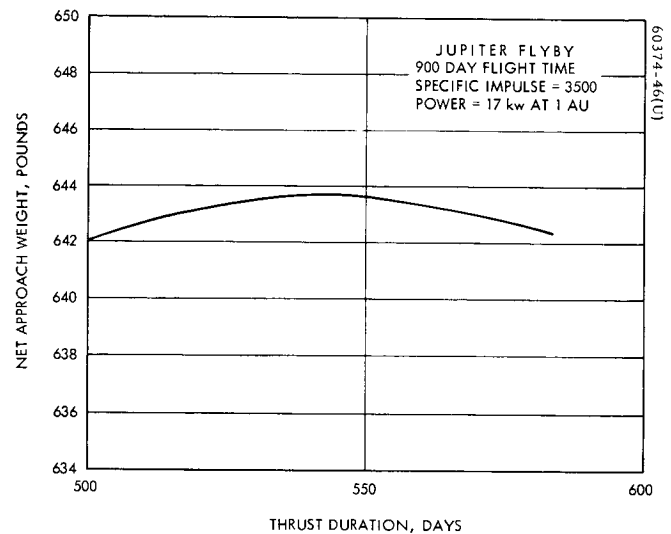


Figure 3-23. Jupiter Flyby, Net Approach Weight versus Thrust Duration

Figure 3-22 indicates that power levels below 17 kw decrease performance due to the reduction in injected weight necessitated by high C_3 . Above 17 kw the increase in injected weight due to a reduction in C_3 is overbalanced by the increased weight of the propulsion system.

Figure 3-23 shows that net approach weight is very insensitive to thrust duration. This would indicate that if it were desirable

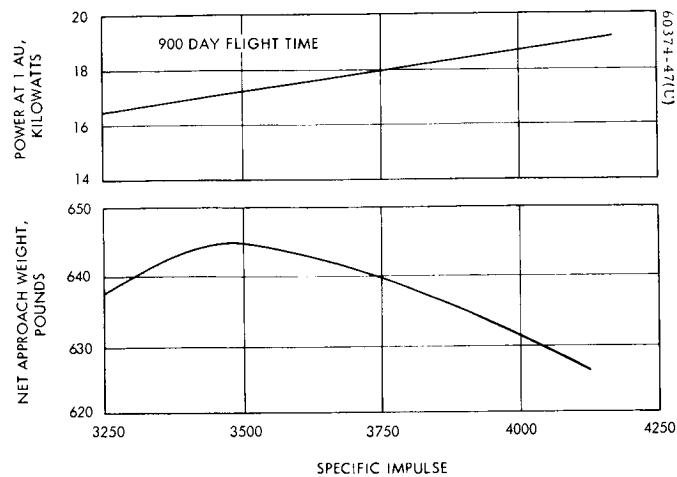


Figure 3-24. Jupiter Flyby, Power at 1 AU and Net Approach Weight versus Specific Impulse

As indicated previously, the initial thrust orientation is approximately counter to total vehicle velocity and at 20 days after earth departure is reoriented to be approximately along the vehicle velocity (Figure 3-21). Figure 3-25 indicates the time of thrust reorientation should be later than the nominal 20 days for best performance and that a launch window in excess of 20 days can be obtained at very small performance penalty by varying the time of thrust reorientation.

Figure 3-26 shows that the hyperbolic excess velocity orientation should be reduced from the nominal value of -90 degrees to about -105 degrees, but that the performance improvement in doing so is very small.

Figure 3-27 shows that performance changes are relatively large for small increases in total flight time. Small changes in the total flight time are much more significant than variations of the other parameters when flight time has been limited to 900 days.

Figure 3-28 indicates the criticality of the propulsion system specific weight. Since 65 lb/kw is probably somewhat conservative, payload capability that is considerably in excess of that which is indicated herein may actually be available. An increase in the flight time would lead to a slightly lower power level and a corresponding reduction in sensitivity to powerplant specific weight.

The effects of the uncertainty in the power curve are discussed in Appendix A.

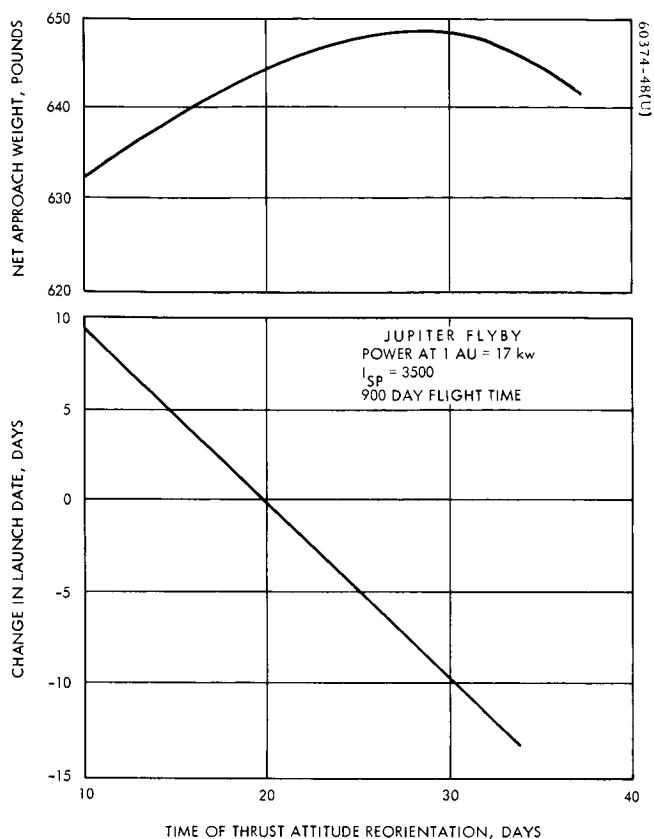


Figure 3-25. Jupiter Flyby, Net Approach Weight and Change in Launch Date versus Time of Thrust Attitude Reorientation

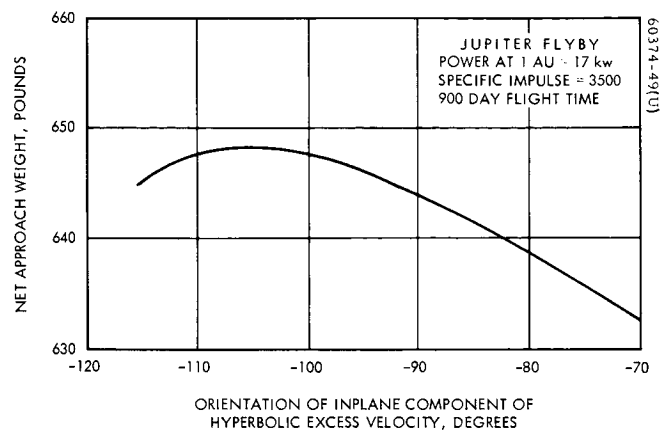


Figure 3-26. Jupiter Flyby, Net Approach Weight versus Orientation of In-Plane Component of Hyperbolic Excess Velocity

Out-of-Ecliptic Probe

The performance capability of the Atlas (SLV-3C)/Centaur launch vehicle with the 17 kw propulsion system design was investigated for an out-of-ecliptic probe mission. In approximate terms the engine is alternately thrusting

normal to the ecliptic plane for 90 days and coasting for 90 days during flight time required to achieve a given out-of-ecliptic angle. With this mode of operation, the vehicle remains at 1 AU at the heliocentric longitude of the earth, providing a minimum communication distance. The magnitude of C_3 was not varied in this analysis.

Net spacecraft weight and flight time as a function of out-of-ecliptic angle are shown in Figure 3-29. It should be noted that a 700 pound net spacecraft weight can be placed 23 degrees to the ecliptic, whereas the same launch vehicle can place a net weight of 700 pounds at an angle of only 10 degrees to the ecliptic without the use of electric propulsion.

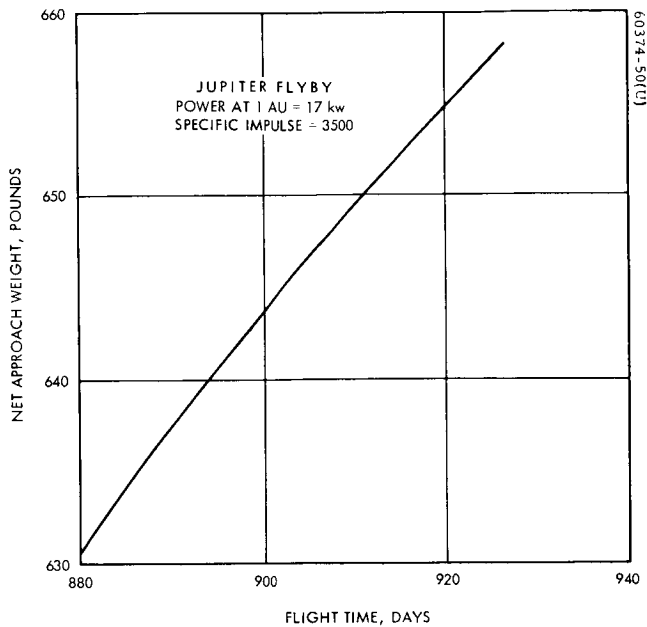


Figure 3-27. Jupiter Flyby,
Net Approach Weight
versus Flight Time

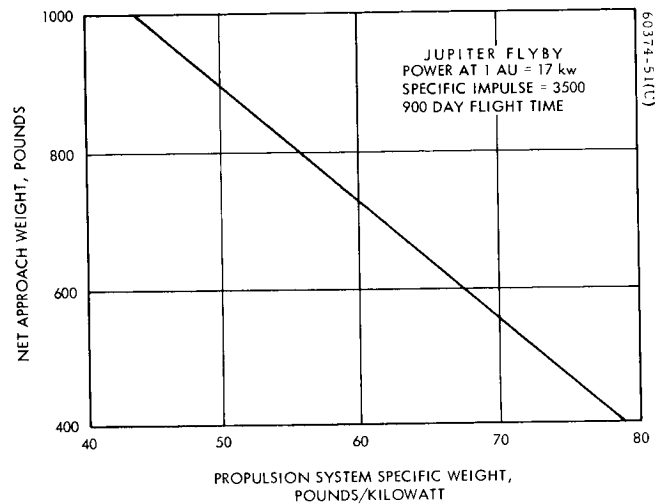


Figure 3-28. Jupiter Flyby,
Net Approach Weight versus
Propulsion System Specific
Weight

This mission is particularly attractive for a low thrust propulsion system test flight and has the ancillary benefit of collection of scientific data that are more difficult to obtain with all chemical probes. The high power is available over each 90 day period through the maximum ecliptic angle for communication and science, since the engine is not thrusting during this period.

Venus Orbiter

The use of the Mars orbiter mission spacecraft and launch vehicle (the Titan III M with a 9.6 kw spacecraft ion propulsion system) for Venus orbiter missions was simulated. Even though no attempt was made to select an optimum low thrust propulsion system size, the performance capability is substantially in excess of that available with the same launch vehicle if low thrust propulsion is not used.

The earth-Venus trajectories are Group I trajectories. Flight times are shorter than for earth-Mars trajectories because of the smaller orbital period of Venus and are less variable between launch opportunities because of the smaller Venus orbit eccentricity. The inclination of the Venus orbit to the ecliptic is, however, nearly twice that of Mars and out-of-plane kinematic effects are therefore considerably more significant.

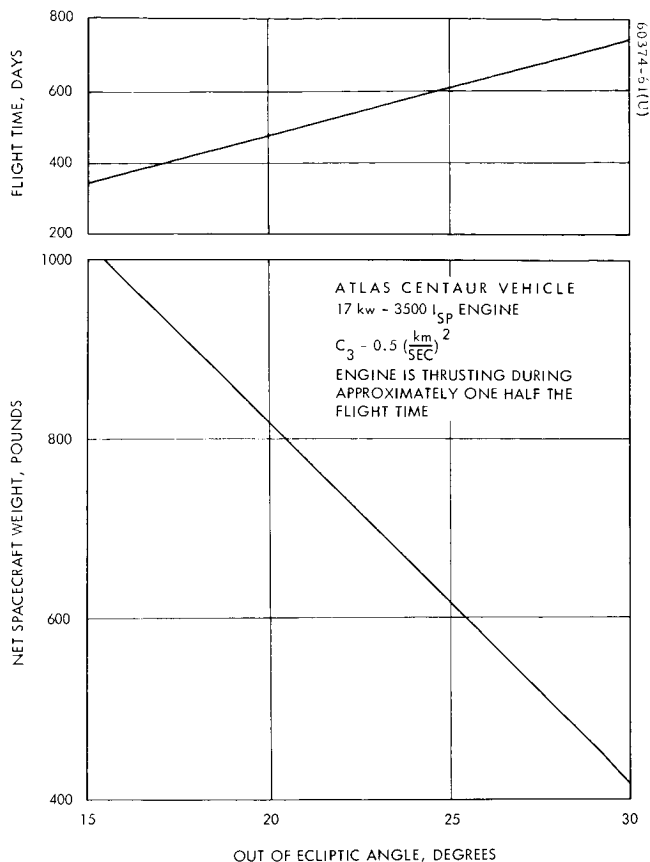


Figure 3-29. Out-of-Ecliptic Probe, Flight Time and Net Spacecraft Weight versus Out-of-Ecliptic Angle

The following nominal conditions were used for all launch opportunities in the study:

Thrust attitude = -90 degrees

Hyperbolic excess velocity orientation = -90 degrees

Power at 1 AU = 9.6 kilowatts

Specific impulse = 4000

Thrust at 1 AU = 65 millipounds

Propulsion system specific weight = 75 lb/kw

Titan III M launch vehicle

High and low thrust propulsion characteristics as given in Appendix A. Table 3-2 gives a summary of other nominal conditions for the three launch opportunities investigated. The weight breakdowns indicate that it would be practical to use a single launch vehicle for more than one launch opportunity.

Figure 3-30 shows the percent change in net orbit weight as specific impulse is varied at constant initial acceleration. As was noted previously this is extremely insensitive.

Figure 3-10 shows the performance change as different values are selected for the constant thrust attitude orientation. It is clear from this figure that the thrust attitude orientation may be held at any value in the vicinity of -90 degrees without incurring a significant performance penalty and that there is no practical advantage to a variable thrust attitude system.

Figure 3-8 indicates that a launch window of approximately 30 days can be obtained at a cost of 2 to 3 percent in net orbit weight. This performance loss is smaller than for high thrust trajectories by at least a factor of two.

The effects of the uncertainty in the power curve are discussed in Appendix A.

TABLE 3-2. VENUS ORBITER NOMINAL CONDITIONS*

Launch Opportunity	Weight, pounds					$C_3 \left(\frac{\text{km}}{\text{sec}} \right)^2$	Approach Velocity, km/sec	Flight Time, days	Launch Date
	Injection	Net Approach	Fuel	Retro	Net Orbit				
1975	5920	4900	300	2230	2670	2.40	2.3	177	12 May
1976	5445	4415	310	1855	2560	4.76	1.6	177	21 December
1978	5225	4195	310	1890	2305	5.86	2.2	180	23 July

*Titan III M launch vehicle 9.6 kw, 4000 I_{sp} engine; 4000 x 20,000 km orbit

Mercury Flyby

A Mercury flyby mission was simulated using the Titan III M launch vehicle and the 17 kw ion propulsion system designed for the Jupiter flyby. The Group II trajectory flown was not optimized in any way; and while the resulting trajectory is very favorable with respect to performance by the same vehicle without the use of electric propulsion, it is still subject to improvement.

It should be noted that the power curve used for the study (Appendix A) is most in doubt within 0.7 AU, and much of the flight time for the Mercury flyby is spent in this region. Although a lower

relative power will decrease the rate of propellant consumption and increase the mission time, the spacecraft velocity capability, and therefore the weight breakdown, will not be changed.

The trajectory conditions selected were the following:

Titan III M launch vehicle

Power at 1 AU = 17 kw

Specific impulse = 3500

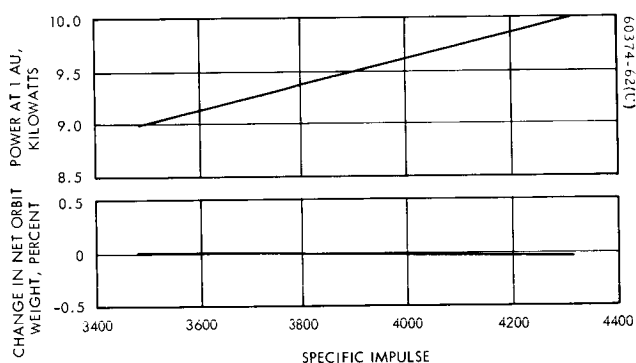


Figure 3-30. 1976 Venus Orbiter, Power at 1 AU and Change in Net Orbit Weight versus Specific Impulse

$$C_3 = 0.5 \text{ (km/sec)}^2$$

Thrust attitude = -90 degrees

Hyperbolic excess velocity orientation = -90 degrees

Propulsion system specific weight = 65 lb/kw

These inputs give the following weights and trajectory characteristics:

Initial acceleration = 1.6×10^{-5} g

Launch date = 5 February 1974

Flight time = 485 days

Approach velocity = 3.8 km/sec

Net approach	3960 pounds
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Propulsion system	1105 pounds
-------------------	-------------

Fuel	1235 pounds
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Injected weight	6300 pounds
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Solar Probe

The advantages of the use of low thrust propulsion for a solar probe have been pointed out by others (e.g., Reference 4). Trajectory simulations utilizing the Atlas (SLV-3C)/Centaur launch vehicle and the 17 kw ion propulsion system design for the Jupiter flyby have qualitatively confirmed the solar probe studies of Reference 4. The basic trajectory profile used has the hyperbolic excess velocity approximately opposed to the earth's orbital velocity, retro-thrust in the vicinity of aphelion, and coast in the vicinity of perihelion. There are different classes of trajectories that are potentially of interest, these classes being distinguished by the number of trajectory perihelions that exist prior to final low thrust termination. Although it appears that the most desirable mission profile will have final thrust termination prior to the second or third perihelion, this decision cannot be made for a practical system without considerably more knowledge of mission constraints than is currently available. This includes not only flight time, but shielding weights required for survivability of the propulsion system as a function of perihelion radius and number of perihelion passages. The uncertainty in the power curve (Appendix A) must also be resolved.

Recapitulation of Trajectory Simulation Results

The interplanetary transfer trajectories considered during this study and their typical characteristics are summarized in Table 3-3. For any mission, the best performance Group II trajectories will lead to higher

TABLE 3.3. SUMMARY OF APPROXIMATE VALUES OF
GENERAL CHARACTERISTICS OF SOLAR POWERED
EARTH-PLANET TRANSFER TRAJECTORIES

Planet	Flight Time, days	Trajectory Group	C_3 , (km/sec) ²	Acceleration at 1 AU $g \times 10^{-5}$	$\frac{\text{Net Approach Weight}}{\text{Injected Weight}^*}$	$\frac{\text{Net Orbit Weight}}{\text{Injected Weight}^*}$
Mars	230 to 430	I	3 to 7	1 to 2	0.6 to 0.7	0.5 to 0.55 (1000 x 20,000 km altitude orbit)
Jupiter	900	II	1 to 1.5	3 to 4	0.25	
Venus	180	I	2 to 6	1 to 1.5	0.6 to 0.75	0.3 to 0.4 (4000 x 20,000 km altitude orbit)
Mercury	500 to 600	II	1 to 1.5	1.5 to 2	0.6	

* Injected weight capability of the launch booster at $C_3 = 0$.

performance but will also have longer flight times (typically hundreds of days longer for Mars and Venus) than Group I trajectories. For any mission, therefore, a tradeoff must be made between the relative importance of flight time and payload weight. The selected trajectory group for each mission is shown in the table.

The flight times show strong variation with launch opportunity for Mars, while the flight time to Venus is nearly constant. These results are due to the relatively high orbital eccentricity of Mars and the nearly circular orbit of Venus. The 900 day Jupiter flight time is that which has been used in the study; shortening this flight time would require a somewhat higher value of initial acceleration. The flight time for the Mercury trajectory is somewhat in doubt because of the uncertainty in the low thrust power available as a function of heliocentric radius. The indicated flight time for the Mercury trajectory is based on very preliminary analysis, and reflects the uncertainty in the available power for this mission.

The inclination to the ecliptic and the eccentricity (variation of heliocentric radius at arrival) of the target planet orbit cause the variation in C_3 requirements for Group I trajectories. The reason for lower C_3 values of Group II trajectories to Jupiter and Mercury is that the longer flight (thrust) time permits an increase in the low thrust propellant-to-propulsion system weight ratio, thereby making use of the low thrust propulsion system more efficient.

Net approach weight and (where applicable) net orbit weight have been nondimensionalized with respect to the injected weight capability of the launch booster at parabolic escape ($C_3 = 0$) rather than at the best values of C_3 for each mission because booster capability is more often described in terms of parabolic escape.

The values shown in Table 3-3 for the Mars and Venus trajectories are determined primarily by the geometry of the target planet orbits and are nearly independent of the launch vehicle and low thrust propulsion system specific weight and power available (as a function of heliocentric radius). In contrast, the Jupiter and Mercury trajectory parameters (with the exception of C_3) are sensitive to system variations. For example, a decrease in mission time will require an increase in the initial acceleration (at 1 AU), and as a result the low thrust propulsion system size must increase and the approach and net orbit weights will be reduced. A reduction in the low thrust power available as a function of heliocentric radius will have this same effect. A change in the specific weight of the low thrust propulsion system will change the approach and net orbit weights, but will not significantly change the other quantities in Table 3-3. The best value of specific impulse for the low thrust propulsion system is not sensitive to the mission or to variation in other parameters, and any value between 3500 and 4000 is suitable for any of the trajectories in the table.

General Conclusions

The following general conclusions have been drawn from the work done under this contract:

- 1) If used properly, low thrust ion propulsion will always increase the payload capability of a given launch vehicle for any interplanetary mission objective.
- 2) Consideration of geocentric and planetocentric effects is vital to mission performance analysis. Furthermore, any low thrust interplanetary trajectory optimization problem can be validly analyzed by inclusion of these effects as algebraic boundary conditions in a detailed simulation of the heliocentric phase.
- 3) At this point in time the potential performance capability available with low thrust ion propulsion is substantially more sensitive to improvements in basic propulsion system parameters (e.g., specific weight and the available power profile) than to modifications in the best performance trajectory characteristics (e.g., thrust attitude time history and C_3 magnitude and direction).

LOW THRUST GUIDANCE AND CONTROL

The 1971 Mars orbiter trajectory has been examined to determine the guidance feasibility and resulting performance cost. The control modes that have been investigated are:

- 1) A change in the acceleration of the vehicle. The specific case simulated was an increase in power utilized at constant specific impulse, but since the low thrust propellant mass is so small

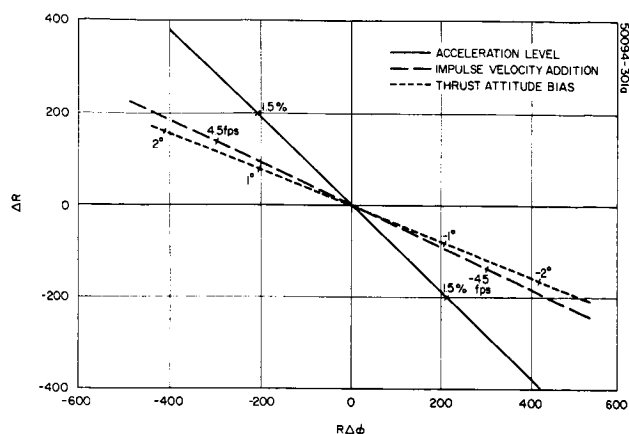
these results differ little from those which would be obtained by changing specific impulse at constant power level. With the efficiency curves for the proposed engine, a 2 percent change in specific impulse at constant power gives a 1 percent change in acceleration level.

- 2) An "impulsive" velocity added by the low thrust system. This may be achieved by incorporating a planned coast period in a nominal trajectory and varying the duration to achieve the impulsive correction. Since 1 day of thrusting corresponds to about 10 m/sec (the exact value depends upon acceleration level), and since corrections larger than this will almost certainly not be required, the approximation to an impulsive change is quite good.
- 3) A small bias added to the programmed thrust attitude time history (which is a constant for the trajectory under consideration).

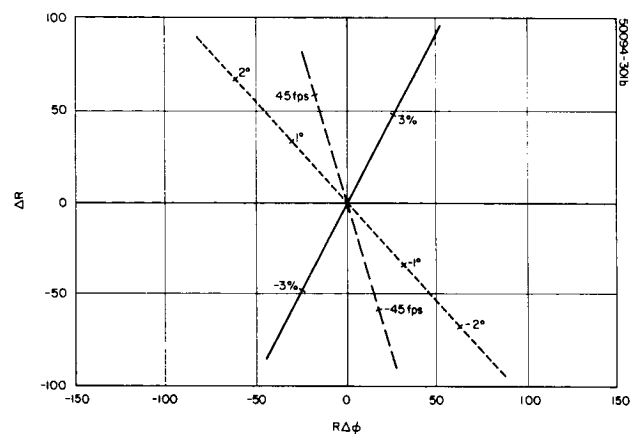
The use of control modes 1 and 2 incurs a performance penalty. No significant performance penalty results from the use of control mode 3 for the small angles which appear sufficient (i.e., less than 3 degrees). These control modes are available without hardware modification. The half cone angle required for the Canopus tracker is only slightly increased over that required by geometric considerations for the present mission (a total half-angle of 25 degrees is presently proposed).

Figure 3-31 shows the changes in radius and tangential displacement that are obtained at the nominal intercept time by the use of the three control modes. The figures illustrate the effect of the control that is implemented (and for the continuous modes subsequently maintained) at four different times during the 250 day flight. The unit of distance is Mars' radius. As indicated in the figures, the relationships are linear to within about one part in 300 in the region shown. The circled numbers refer to the control modes given above. For control mode 3 a positive sign indicates that $\ddot{R}$ becomes more positive.

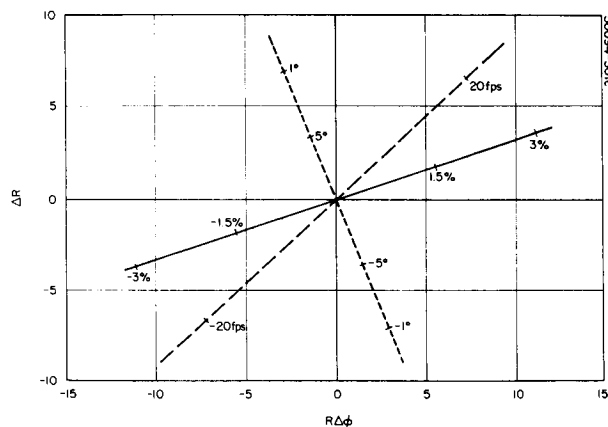
It is clear from these figures that any error in radius or tangential displacement that exists at the nominal intercept time can be corrected by an appropriate combination of two or more of the three control modes. For example, if at 120 days following launch the predicated miss at nominal intercept is $\Delta R = 20$ and $R\Delta\phi = 60$, this can be corrected by increasing thrust attitude (control mode 3) by 1 degree and decreasing acceleration level (control mode 1) by 3.3 percent. The performance costs required to make the correction (or more precisely, the performance degradation in the nominal required to provide the capability to make the correction) depend upon the size of the dispersion that must be corrected.



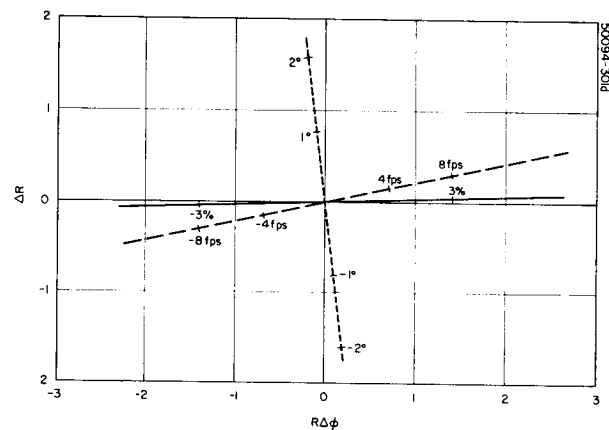
a) $T = 0$ Days



b) $T = 120$ Days



c) $T = 140$ Days



d) $T = 230$ Days

Figure 3-31. Intercept point Displacement (in Martian Radii)
Available from Three Control Modes

It should be noted that the mission may not require that intercept occur at a prescribed time. Intercept can, in general, be more efficiently accomplished by allowing time of intercept to change and the performance indicated in the figures is therefore an upper limit. The more difficult estimation of performance cost for the general case of variable intercept time has not been investigated because the upper limit of the performance penalty is small.

The most significant error by far at the start of the trajectory is that due to dispersions in energy (velocity magnitude). This error can be removed by appropriately timing the initiation of the low thrust engine ignition. A variation of 1 day will probably be sufficient, and from Appendix D the resulting performance cost is found to be about 0.1 percent of net orbit weight. It would also be possible to remove this injection error with a combination of control modes 1 and 3. Since a relatively small acceleration increase would be required, if applied early in flight, the performance cost would be smaller for this approach.

The errors that exist in the trajectory at subsequent times depend on the accuracy of the orbit determination system, the accuracy with which attitude and engine parameters can be controlled, and the control procedure used (e.g., how often updated commands are given). It appears that engine parameters can easily be controlled to a fraction of 1 percent which, from the figures, is very adequate. The low frequency thrust attitude error is probably less than 0.3 degrees, which also appears quite satisfactory. The accuracy which will be obtained from the orbit determination program is not known at this time; however, the undetected existence of velocity errors of several feet per second is extremely improbable, particularly near the end of the flight. It is therefore unlikely that acceleration level changes in excess of 3 percent or attitude commands in the ecliptic plane in excess of 2 degrees will be required for any reasonable control procedure.

Attitude control normal to the ecliptic plane has not been investigated due to the limitations of time and funds. Attitude corrections for guidance purposes are determined by the same parameters that determine in-plane requirements, and will probably be likewise limited to 1 to 2 degrees.

It does not appear that control mode 2 is necessary or desirable (with the possible exception of the timing of original engine ignition) for the trajectory under consideration. However, this is not necessarily true in the general case since the influences of various control modes depend on the trajectory. It is anticipated that the relative performance costs and control authority required will not differ significantly as a function of the trajectory. The following conclusions affecting the design and mission analysis are therefore drawn:

- 1) Guidance errors can be removed with the low thrust propulsion system at a performance cost which corresponds to a loss of less than 1 percent of weight in the Martian orbit.
- 2) The attitude commands required for guidance will not exceed a cone angle of 3 degrees at any time.
- 3) Operation with specific impulse that is off nominal by more than 6 percent will not be required.

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4. William C. Strack, "Solar-Electric Propulsion System Performance for the 0.1 AU Solar Probe Mission, " ALAA Paper No. 66-496, June 1966.
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4. ELECTRIC PROPULSION SYSTEM DESIGN

The propulsion system studies, performed during this program, have as their primary objective an evaluation of the problems associated with the design and development of ion propulsion systems for solar powered interplanetary spacecraft. These studies have consisted of: 1) a study of the effect of power level on the design and operating characteristics of the major propulsion subsystems; 2) an analysis to determine from a weight-reliability standpoint the optimum thruster module size for a high power propulsion system, and to determine the effect of reliability considerations on overall system design; 3) a study of thrust vector displacement in redundant thruster arrays; 4) an analysis of the problem of voltage and power matching between solar panel output and the engine system load and the effect of this matching on the choice of thruster module size; 5) a design layout of each of the major subsystems that would be used to make up a high power (i. e. , 9.6 and 17 kw) propulsion system; 6) an analysis of the thermal, mechanical, and electrical integration problems associated with a modularized ion engine system; and 7) conceptual designs of both 9.6- and 17-kw ion propulsion systems. These analyses and designs have, in each case, been carried out for both the oxide cathode and the liquid mercury cathode, electron-bombardment thruster systems.

Although some portions of the above studies have already been reported in Reference 1, each will, where appropriate, be summarized here. The work performed subsequent to Reference 1 will be presented in detail. In this manner a continuity of the total effort will be maintained.

SUBSYSTEM EVALUATION AND SCALING STUDY

In the initial phase of this program, an evaluation of the state of art of the major propulsion system subsystems (i. e. , thrusters, feed systems, and power conditioning and control systems) was conducted. The thrusters considered were the oxide cathode and liquid cathode Hg electron bombardment engines along with their associated feed systems. After a detailed comparison with conventional type power conditioning systems, a modularized type of power conditioning was chosen as most advantageous for the applications considered in this study. Following these evaluations, scaling studies (or the variation of subsystem characteristics with power level) on applicable subsystems were performed to aid in determining the optimum

engine module size for a high-power ion propulsion system. These parametric studies were also required to define the optimum design parameters for subsystems at any power level and guarantee that the interpolation and/or extrapolation of existing data during the design phase would be done on a realistic basis.

Thruster

Two types of Hg electron-bombardment thrusters were evaluated during this program: oxide cathode and liquid cathode. It will be demonstrated that from a systems standpoint there exists a great degree of interchangeability between these two thrusters, although the oxide cathode engine is probably more representative of the state of the art.

Engine Efficiency

Primary to the evaluation of the ion thruster subsystem is a determination of its power and propellant utilization efficiencies. Much experimental data on thruster efficiency exists and some highlights are shown in Figure 4-1. (The legend indicates the sources of the data.) For the purposes of this study the following data will be chosen as typical of present-day Hg electron bombardment thrusters:

Combined arc and cathode power losses = 600 ev/ion

Propellant utilization efficiency = 85 percent

Figure 4-1 indicates that these assumptions are reasonable. In fact, recent developments indicate that arc and cathode power losses less than 500 ev/ion are readily attainable even at the low specific impulses (e. g. , 3500 seconds). The thruster power efficiency can now be determined by adding to the arc and cathode losses the other losses such as those incurred by the neutralizer, feed system, and accelerator electrode. The total engine efficiency η_e is given by

$$\eta_e = \eta_u \eta_p \quad (1)$$

$$\eta_e = \eta_u \frac{I_{sp}^2 / 100 \eta_u^2}{I_{sp}^2 / 100 \eta_u^2 + P_L} \quad (2)$$

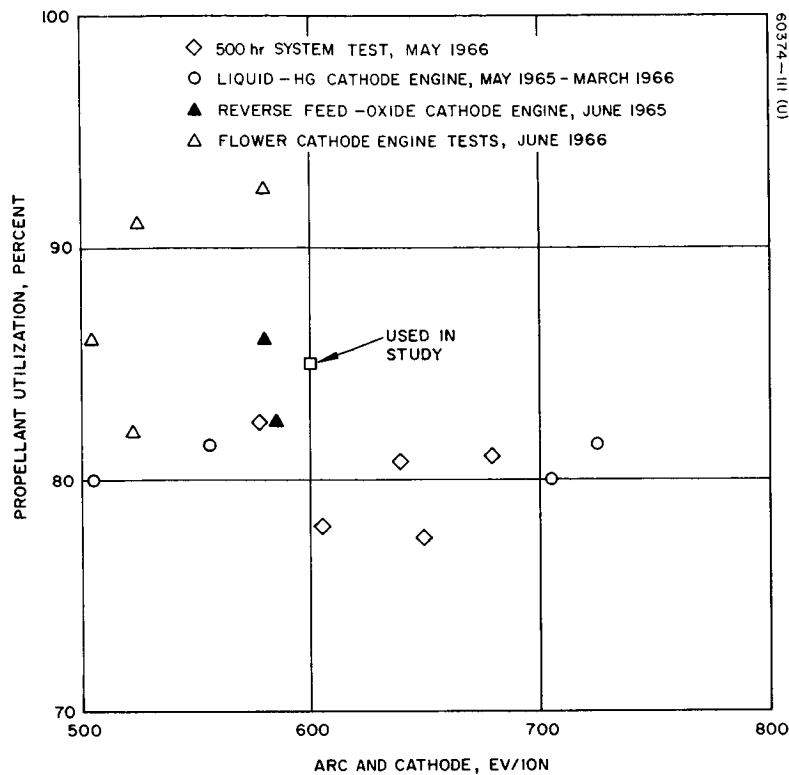


Figure 4-1. Propellant Utilization Efficiency versus Total Power Loss in Discharge Chamber

where

η_u = propellant utilization efficiency

η_p = power efficiency

I_{sp} = effective specific impulse, seconds

P_L = total losses, ev/ion

Equation 2 is plotted in Figure 4-2 for various values of P_L and η_u . For present day engines neutralizer, feed system, and accelerator electrode power losses amount to an additional 150 ev/ion (see Figure 4-2), producing a total power loss of 750 ev/ion. The dashed curve in Figure 4-2 thus represents what will be considered here state-of-the-art performance and it is used as an input to the trajectory studies previously discussed. (However, to obtain total propulsion system efficiency the data in Figure 4-2 must still be multiplied by the power conditioning efficiency to be discussed later.)

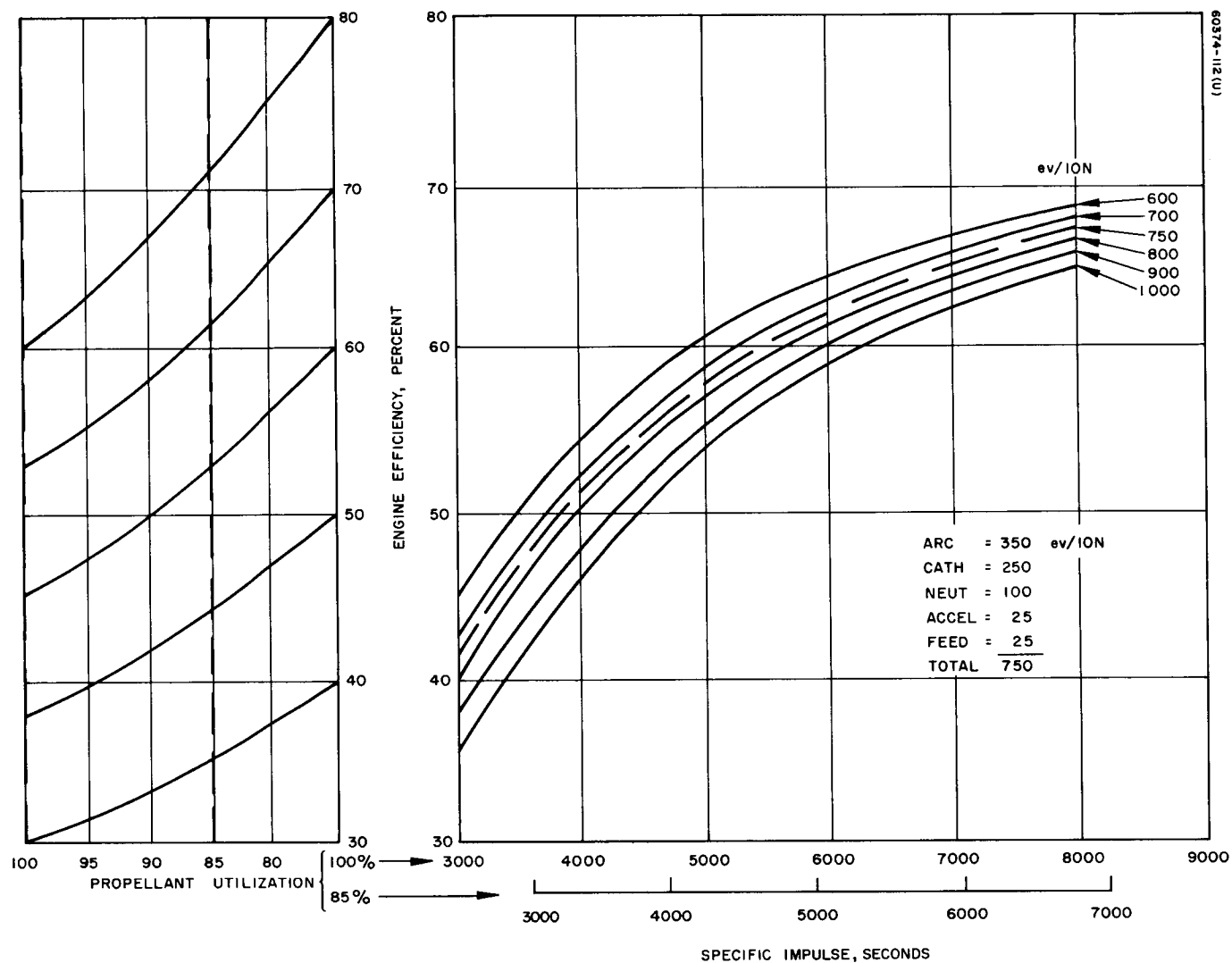


Figure 4-2. Mercury Bombardment Engine Efficiency

For the values chosen above, Figure 4-2 yields:

$$I_{sp} = 4000 \text{ seconds} ; \eta_e = 64 \text{ percent}$$

$$I_{sp} = 3500 \text{ seconds} ; \eta_e = 59 \text{ percent}$$

which, as shown in the Mission Analysis Section, are the selected effective specific impulses for the Mars and Jupiter missions, respectively.

Oxide Cathode Engine

To establish the relation between power level and engine design parameters, the mercury bombardment engine was scaled over a range of power levels. There are, in general, two factors that will influence the thruster scaling study — the perveance of the optical system and the thruster life requirements. By consideration of these two factors it is possible to determine, as a function of power level, an optimum thruster configuration that provides a lifetime of at least 1 year at I_{sp} levels of 3500 and 4000 seconds.

In Reference 1 the approach to the thruster scaling study was for the most part analytical. The purpose for reviewing these studies here is to provide wherever possible some measure of experimental verification of the earlier work.

Perveance. A relationship was derived in Reference 2 that gives the current of a single accelerator electrode aperture (i. e. , one of the several hundred holes that make up the ion optical structure) as a function of hole diameter (w) and electrode spacing (d) (see Figure 4-3). (This relationship was experimentally confirmed when modifications were made to the ion optical system during the Hardware Verification Phase of this contract.) For ion optics in which the screen electrode is countersunk on the plasma side (making the electrode thickness effectively zero), the current through a single aperture (I_H) is given by

$$I_H = 4.4 \times 10^{-9} \left(\frac{1}{0.336 + \frac{d}{w}} \right)^2 V_T^{3/2} = P_H V_T^{3/2} \text{ amps} \quad (3)$$

where

V_T = the total accelerating voltage ($|V_A| + V_B$), volts

V_A = the negative voltage applied to the accelerator electrode, volts

V_B = the beam voltage, volts

P_H = the perveance of a single aperture, A/volts^{3/2}

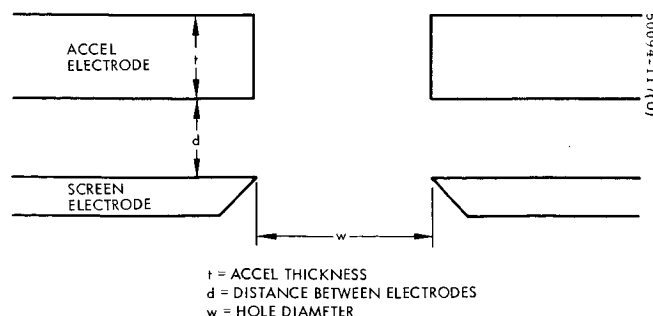


Figure 4-3. Definition of Electrode Geometrical Parameters

The total current from a given thruster (I_B) is

$$I_B = \bar{P}_H N V_T^{3/2} \quad (4)$$

where

$\bar{P}_H$ = average perveance of all apertures

N = number of holes in accelerator electrode

Due to variations in plasma density as a function of screen electrode radius, $\bar{P}_H$ is typically equal to $1/2 P_H$ (References 4, 5, and 6). Hence, Equation 4 becomes

$$I_B = 1/2 P_H N V_T^{3/2} \quad (5)$$

The The number of holes (N) of a thruster electrode is given by

$$N = \gamma \left(\frac{D}{w} \right)^2 \quad (6)$$

where

γ = ratio of open electrode area to total electrode area

D = active engine diameter, cm

w = hole diameter, cm

Substituting for P_H from Equation 3 and for N from Equation 6, Equation 5 becomes

$$I_B = 1/2(4.4 \times 10^{-9}) \left(\frac{1}{0.336 + \frac{d}{w}} \right)^2 \left(\frac{\gamma D^2}{w^2} \right) V_T^{3/2} \text{ amps} \quad (7)$$

Solving for I_B/D^2

$$\frac{I_B}{D^2} = 2.2 \times 10^{-9} \left(\frac{1}{0.336 w + d} \right) \gamma V_T^{3/2} \text{ A/cm}^2 \quad (8)$$

Considerable data is available in the literature to indicate the current range over which thrusters of 10, 15, 20, and 50 cm diameters have been operated. These data points indicate values of I_B/D^2 of 1.5×10^{-3} A/cm² are typical and that some isolated values greater than 2×10^{-3} have been obtained. Substitution into Equation 8 of the values of w , d , γ , and V_T associated with these thrusters gives calculated values of I_B/D^2 of approximately 2×10^{-3} A/cm². Thus, this relationship (i.e., Equation 8) roughly approximates the maximum envelope of the experimental data.

Electrode Life. The accelerator electrode in any ion thruster will ultimately fail mechanically from sputtering damage due to ion impingement. These ions are either fast ions in the beam that directly impinge the accelerator or charge exchange ions, formed in the first few centimeters of the beam, which are electrostatically attracted to the accelerator. Assuming the direct impingement can be reduced to essentially zero, the electrode lifetime (T) will be, to a first order, inversely proportional to the charge exchange current density to the accelerator (j_a) and directly proportional to the accelerator thickness (t), that is

$$T \propto \frac{t}{j_a} \quad (9)$$

The charge exchange current density to the accelerator is directly proportional to the product of the beam current density and the neutral efflux that is given by

$$[(1 - \eta_u)I_B],$$

that is

$$j_a \propto I_B^2 \frac{(1 - \eta_u)}{D^2} \quad (10)$$

The required accelerator thickness is then limited by

$$t \geq \frac{K T I_B^2 (1 - \eta_u)}{D^2} \quad (11)$$

where the constant K may be estimated analytically, but in general must be determined experimentally. Using the values of Reference 7 as typical,*

*The calculation of K here is presented to illustrate a typical case. Values for other systems may vary at least a factor of two due to uncertainties in estimating T from experiment.

$$T \geq 10^4 \text{ hours (estimate)}$$

$$D = 20 \text{ cm}$$

$$\eta_u = 0.8$$

$$I_B = 0.6 \text{ amperes}$$

$$t = 0.15 \text{ cm}$$

then (assuming $T = 10^4$ hours)

$$K = 8 \times 10^{-2} \frac{\text{cm}^3}{\text{A}^2 \text{ hour}}$$

Finally, Equation 11 can be written for the general case as

$$t = 8 \times 10^{-2} \frac{T I_B^2 (1 - \eta_u)}{D^2} \text{ cm} \quad (12)$$

Thus, given I_B , T , and η_u , the accelerator electrode thickness may be calculated for a thruster of a given diameter.

Electrode Design. One additional consideration that must be discussed prior to developing a relationship between thruster weight and power is a definition of the number and size of the individual apertures in the ion optical system. Again, it is necessary to rely on relationships between the variables that have been established experimentally as currently representing the state of the art, but which may change as new designs are developed. The following relationships will be assumed typical:

- 1) From ion optical studies:

$$w \geq 3t$$

$$3d \geq w$$

- 2) To minimize danger of voltage breakdown between the electrodes:

$$\frac{V_T}{d} \leq 10^5 \text{ V/cm}$$

- 3) For mechanical integrity of the electrode structure the center-to-center distance between individual apertures (ℓ) must be

$$\ell \geq 1.25w$$

- 4) To assure proper beam collimation the accel-decel ratio

$$V_T/V_B \leq 4$$

Assuming the equalities to hold in the above relationships, the minimum number of holes ($N_{\min}$) required to produce a total beam current at some specific impulse (i. e., V_B) can be determined using Equations 3 and 5. For example, by substituting $d/w = 1/3$ into Equation 3, the maximum perveance of a single aperture is given as

$$(P_H)_{\max} \cong 10^{-8} \text{ A/volt}^{3/2}$$

Solving Equation 5 for N , and substituting $(P_H)_{\max}$ for P_H and $4V_B$ for V_T , gives

$$N_{\min} = 2.5 \times 10^7 \left(\frac{I_B}{V_B^{3/2}} \right) \quad (13)$$

Furthermore, the maximum number of holes ($N_{\max}$) for a given thruster diameter, beam current, and required lifetime can be determined from Equations 6 and 12. For example, for given values of D , I_B , and T , the accelerator electrode thickness, t , can be found. Assuming then that $w = 3t$, the minimum value of w is defined. Moreover, it can be shown that for a hexagonal closed pack array of circular holes that

$$\gamma = 0.9(w^2/\ell^2) \quad (14)$$

Assuming that $\ell = 1.25w$, the maximum value of γ is defined. Substituting $w_{\min}$ and $\gamma_{\max}$ into Equation 6 then gives $N_{\max}$.

The number of holes may be chosen between the two limits given by Equations 13 and 6. For fabrication purposes it is desirable to have relatively few holes. However, the minimum possible number is not usually chosen since this implies operation at the maximum possible acceleration-deceleration ratio. The higher acceleration-deceleration ratios are to be avoided where possible because of the deleterious effects on the focusing of the emerged beam. Typically thrusters are built with $N = 3/4 N_{\max}$.

Engine Size and Weight. A general expression for engine diameter as a function of input power may be derived from Equation 8 by again assuming values for thruster parameters (i. e., w , d , γ , and V_T) typical of operating engines. The power input to an ion engine is given by

$$P_{\text{input}} = \frac{I_B V_B}{\eta_p} \quad (15)$$

Substituting $(2 \times 10^{-3} \text{ A/cm}^2 \times D^2)$ for I_B as determined previously from Equation 8 for typical thrusters and assuming $\eta_p = 0.7$ for an I_{sp} of 3500 seconds (i. e., $V_B = 1700$ volts), Equation 15 becomes

$$P_{\text{input}} = 4.8 D^2 \text{ watts} \quad (16)$$

This relationship is plotted in Figure 4-4.

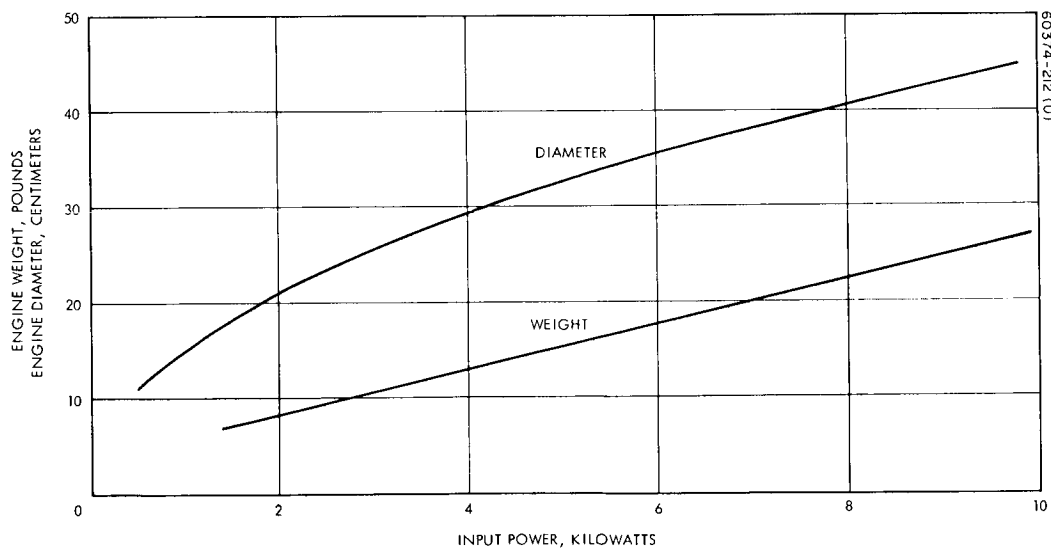


Figure 4-4. Engine Size versus Engine Power

A summary of the power versus weight data for oxide cathode thrusters that are currently available in the literature is presented in Figure 4-5. Based on the previous analyses that provide thruster diameter as a function of power and approximate electrode design specifications, a weight estimate was made for specific flight-type thruster designs. These weight estimates were then used to modify the original curve as shown in Figure 4-5. These

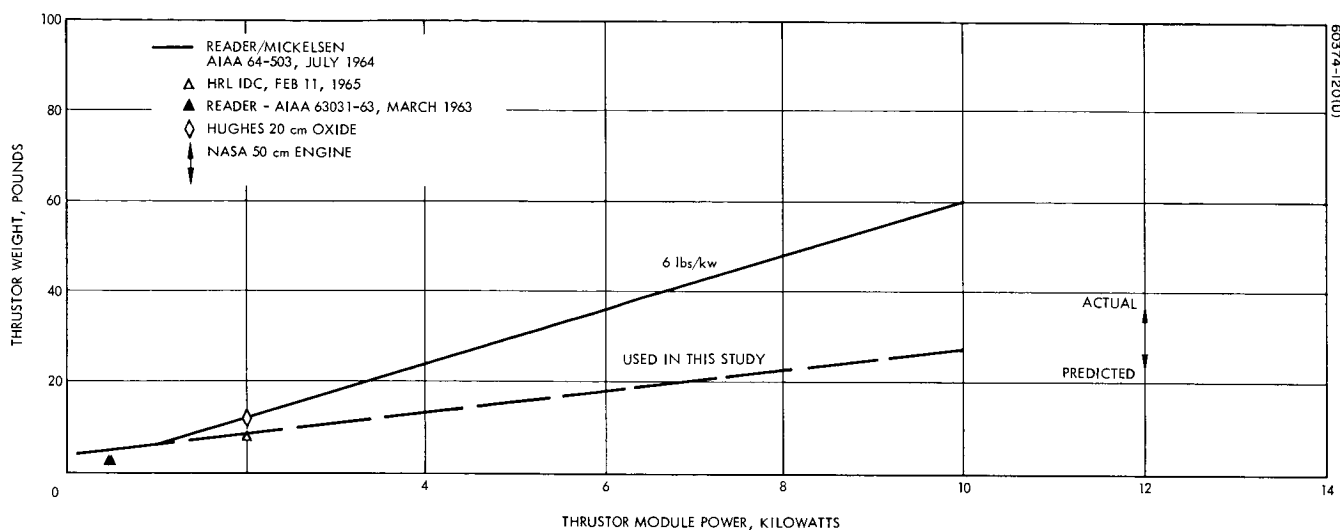


Figure 4-5. Engine Weight versus Engine Power (Oxide Cathode Engine, Life = 10,000 hours; Specific Impulse = 3500-4000 seconds)

data are also presented in Figure 4-4 to provide direct comparison of engine size and predicted flight weight as well as for later use in the Subsystem Layout Section.

Liquid Mercury Cathode Engine

The second type of mercury bombardment thruster considered in this study is one which employs a liquid mercury cathode (LMC) in the discharge chamber. In Reference 1 thruster designs utilizing a low temperature (130°C) LMC were studied. These designs incorporated a lightweight aluminum radiator external to the main thruster unit. Because of the required low temperature of the heat rejection radiator (80°C), these thrusters were shown to have an undesirable thermal constraint when applied to interplanetary missions. For this reason, a program to develop a high-temperature (250° to 300°C) liquid mercury cathode was initiated by NASA. To date, liquid mercury cathodes have been operated successfully at temperatures up to 300°C . However, operation of these cathodes in a thruster is still in the preliminary stages. These engines are therefore to be considered developmental at this time.

A design study was conducted during this program to determine the thruster configuration modifications that would be necessary, assuming the availability of the high temperature liquid mercury cathode. Included in this study was a determination of the relation between power level and thruster design parameters. The details of the analysis are given in Appendix E. The chief difference between the oxide and liquid mercury cathodes that affect the scaling studies is the heat rejection requirement. In fact, it can be shown that the same engine diameter versus engine power relationship derived for the oxide cathode (Figure 4-4) may be used for the liquid Hg cathode. However, because of the added shell weight (see Appendix E), the engine weight versus power relationship will differ. This effect is illustrated in Figure 4-6. It is seen that the percent of weight increase is much less for small engines than for large. It should be noted that the engine shell weight versus power level relationship developed in Appendix E is based on the assumption that a single cathode (located on the axis of symmetry) is employed in the thruster. It is likely, however, that at the higher power levels (e. g. , engine diameters > 30 cm) a multiple cathode arrangement would be required to maintain a relatively uniform plasma density in the discharge chamber. Such an arrangement would shorten the average thermal conduction path from the cathode to the thruster shell, thereby reducing the overall engine weight at the higher power levels.

Feed System and Propellant Reservoir. Two types of feed systems must be considered: 1) an Hg vapor feed system for the oxide cathode thruster, and 2) a liquid Hg feed system for the liquid cathode thruster. Although the specific components employed are quite different, the two systems are functionally identical. Each system consists basically of a means of controlling propellant flow and of electrically isolating the thruster from the feed system and reservoirs. For the vapor feed system both flow control

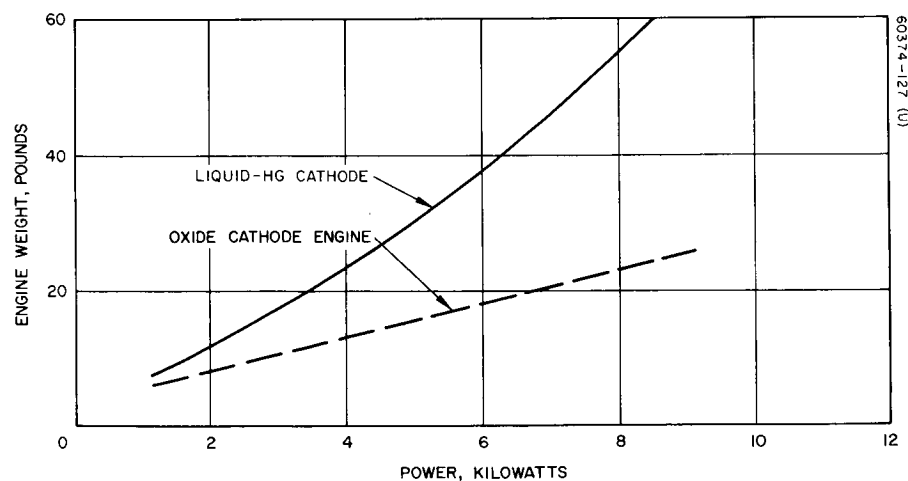


Figure 4-6. Engine Weight versus Engine Power (Liquid-Mercury Cathode Engine, Life = 10,000 hours; Specific Impulse = 3500-4000 seconds)

and conversion from liquid mercury to mercury vapor are accomplished by a vaporizer which is basically a fine mesh screen that will hold back liquid mercury by surface tension forces up to pressures of approximately 1 atmosphere. Mercury is vaporized from the surface exposed between the screen wires and the evaporation rate (hence the propellant flow rate) is controlled by the vaporizer temperature. By this means precise, continuous control is easily achieved.

An electrical isolator is located between the vaporizer and the thruster. This unit permits the feed system to operate at spacecraft electrical ground while the thruster is at high voltage. Basically the isolator is a section of insulating tubing whose size and temperature are adjusted so that the mercury vapor pressure will be too low to sustain a discharge along the tube at operating voltage.

In the LMC system similar functions to the above are handled by different components. Here liquid mercury must be delivered under pressure to the cathode. Flow is directly proportional to the pressure and is adjusted by an electromagnetic pump which produces force by the interaction of a current and a magnetic field. In the proposed designs a permanent magnet and a variable current would be used to control the pressure. To obtain voltage isolation in this case gas bubbles are injected into the insulating section of the feed line to break the electrical continuity. They are permitted to escape into the vacuum by a downstream porous section of line which will not pass liquid mercury. A mechanism is provided to assure that at least one bubble will be in the insulating section of the feed line at all times.

To determine the relation between engine module power level and feed system design parameters, it is first necessary to establish what feed system components are affected. Each thruster module requires the following feed system elements: 1) vaporizer, 2) isolator, 3) valve, and 4) plumbing. The propellant reservoir is not included in the above list because the total amount of propellant required is a constant for a given application, and the way in which the propellant is stored (number and size of tanks used) is independent of the number of thruster modules chosen. For example, one tank could feed several engines, or several tanks could feed one engine. The propellant storage system for the oxide cathode engine is identical to that for the liquid mercury cathode engine.

The requirement that each thruster module have a complete duplication of the four components listed above arises directly from the operating characteristics of the feed system module. Flow control is produced by either the vaporizer or EM pump. Since each thruster requires separate flow control, one device could not properly supply control for more than one thruster. Duplication of high voltage isolators and valves is required to allow thruster modules to be electrically isolated and separately switched in and out of a central propellant storage system (consisting of one or many tanks).

The basic feed system module is shown in Figure 4-7. The thrustor module and a propellant storage tank are included for completeness. The feed system component maximum weights are given in Table 4-1. These weights are independent of the power level of the engine module over the range of interest. Therefore, the feed system weight per engine module (excluding the storage system) is a constant. The total feed system weight (again, excluding the storage system) is, of course, a direct function of the number of engine modules because of the one-to-one relationship between engine modules and feed system components.

In general there are two basic types of propellant reservoirs that can be considered available for mercury bombardment engine systems. In the type used as an example in this report the mercury is contained in a flexible "Bellofram" type bladder and kept under constant pressure by a piston driven either by vapor pressure or a "negator" constant force spring. In the second type of system the piston is replaced by a volume of gas which transmits its pressure to the mercury through the flexible bladder. In this latter type of

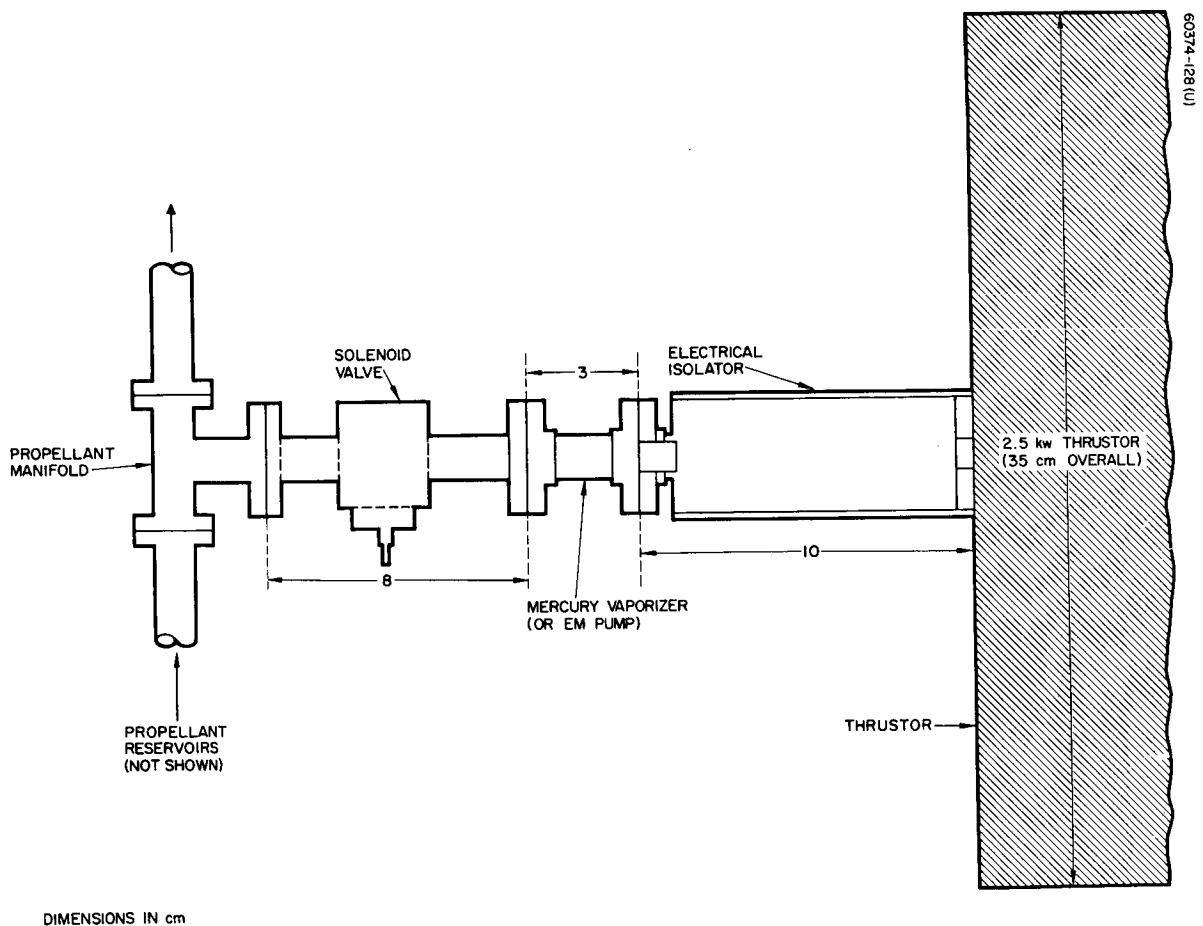


Figure 4-7. Basic Feed System Module

TABLE 4-1. FEED SYSTEM COMPONENT WEIGHTS

Component	Oxide Cathode Engine	Liquid Hg Cathode Engine
Flow control	0.5	1.0*
Isolator	0.2	0.3
Solenoid valve	0.5	0.2
Plumbing**	<u>~1.5</u>	<u>~1.5</u>
Total	2.7	3.0

* Design goal spacecraft.

** Depends on spacecraft configuration.

system the mercury pressure decreases as the mercury is expelled from the reservoir since $PV = \text{constant}$ for the gas. Any small effects that this variation in pressure may have on the vaporizer performance may be compensated for by adjusting the vaporizer temperature.

A detailed design analysis on the vapor pressure-piston type reservoir was presented in Reference 1. As a result of this analysis, reservoir sizes and weights were given as a function of liquid Hg storage capacity. Although these data will not be repeated here, they were used in the Subsystem Layout Section of this report.

Power Conditioning and Controls. An ion engine power conditioning system for a solar powered, electric propulsion system must perform several functions. First, the system must be able to convert solar panel output voltage to usable ion engine voltages. Second, the system must be able to control and regulate the ion engine operating voltages and currents. Third, the system must provide a power and impedance match between the solar cell source and the ion engine loads. Further, this must be accomplished with power conditioning circuitry and components that will provide low specific weight, high power efficiency, long operating life, and high system reliability. Using these design criteria, two power conditioning design approaches were considered and evaluated. The first, or conventional, method consisted of a single high power level circuit for each of the required power blocks; the second method was to build up the various power supplies by adding the outputs from low power modules. When applied to an ion engine power conditioning, the modular approach has many significant advantages over the more conventional methods. These advantages, which led to the selection of the modular approach for this system, are described in Reference 1.

Engine size (power level) has a significant effect on the specific weight of the associated power conditioning. As indicated in Figure 4-8, the specific weight varies from 12.0 to 6.3 lb/kw over the range of 1.2 to 6.0 kw input power to the engine. The rate of decrease in specific weight is greatest at the low end of the engine power range; the specific weight approaches a limit of 6 lb/kw above the 6 kw level. At 2.5 kw power conditioning specific weight is 8 lb/kw.

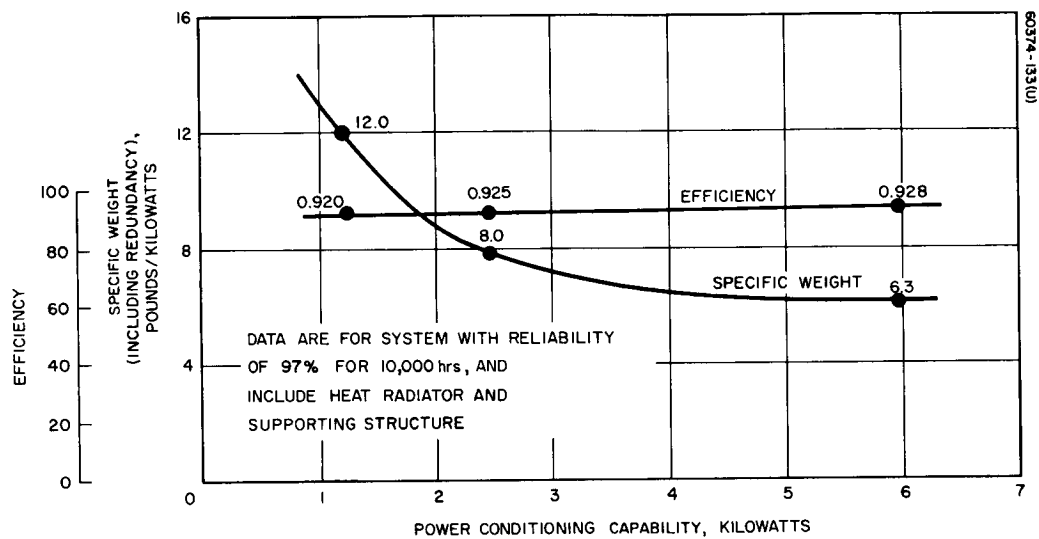


Figure 4-8. Power Conditioning and Control System Specific Weight versus Power

Table 4-2 is a breakdown of system weights for the engine sizes considered and illustrates the sources of the reduction in specific weight with increasing engine size. The most significant factors are the decreasing weight penalty of redundancy and the decrease in percent of system weight contributed by the low-power supplies and control circuitry. The decreasing relative weight of the circuitry required for redundancy is due to the use of 100 percent redundancy at low power levels and fractional redundancy at high power levels, with system reliability held constant. The modular concept of power conditioning lends itself readily to the use of fractional redundancy. The benefit of this capability is most evident in the low weight required for redundancy in the beam supply, as shown in Table 4-2. It is interesting to note that the redundancy weight is 44 percent of the operating weight in the 1.2 kw system, as opposed to 22.6 percent for the 6 kw system. A similarly interesting comparison is that of the weight of the beam supply, the principal power increment. The beam supply operating weight is 42 percent of the total operating weight in the 1.2 kw system and 56.5 percent in the 6 kw system, illustrating the decreasing relative weight contribution of the low-power increments and control circuitry at the higher power levels. It should be emphasized that system weights used include the weight of the power conditioning structure supporting the system modules, and also includes the weight for radiation cooling, implicit in the thermal self-sufficiency of each module.

TABLE 4-2. POWER CONDITIONING AND CONTROL SYSTEM WEIGHTS
 $I_{sp} = 3500$ seconds

Function	1.2 Kilowatts Output					2.5 Kilowatts Output					6.0 Kilowatts Output				
	Output Power, watts	Operating Weight, pounds	Redundant Weight, pounds	Total Weight, pounds	Specific Weight, pounds/kilowatts	Output Power, watts	Operating Weight, pounds	Redundant Weight, pounds	Total Weight, pounds	Specific Weight, pounds/kilowatts	Output Power, watts	Operating Weight, pounds	Redundant Weight, pounds	Total Weight, pounds	Specific Weight, pounds/kilowatts
Beam	830	4.2	1.8	6.0	6.85	1750	6.02	2.76	8.78	4.87	4200	17.5	3.68	21.18	4.8
Arc	165	1.3	0.8	2.1	21.00	350	3.0	1.00	4.00	11.50	875	5.0	1.00	6.00	6.5
Cathode Heater	120	0.6	0.6	1.2	12.00	250	1.2	0.7	1.90	7.60	600	1.4	0.6	2.00	4.6
Accelerator	12	0.8	0.5	1.3	26.00	25	0.4	0.3	0.70	14.00	60	0.8	0.8	1.60	13.3
Neutralizer	48	0.2	0.1	0.3	6.00	100	0.2	0.1	0.30	3.00	240	0.3	0.2	0.50	5.0
Feed	25	0.2	0.1	0.3	12.00	25	0.2	0.1	0.30	12.00	25	0.2	0.1	0.30	12.0
Control	--	0.5	0.5	1.0	0.83	--	0.6	0.6	1.20	0.46	--	0.7	0.5	1.20	0.2
Frame	--	2.2	--	2.2	1.83	--	3.0	--	3.00	1.10	--	5.0	--	5.00	0.8
Total	1200	10.0	4.4	14.4	12.0	2500	14.62	5.56	20.18	8.0	6000	30.9	6.98	37.78	6.3

Analysis of system efficiency versus engine size shows only a small improvement as engine size increases. This increase in system efficiency occurs since as the engine power increases the less efficient feed system supplies become a smaller part of the total system. The feed system supplies are less efficient than the beam supply because of the continuous control capability that is required. As shown in Figure 4-8, the total system efficiency varies from 92 percent at 1.2 kw to 92.8 percent at 6 kw.

The specific weights and efficiencies shown in Figure 4-8 represent substantial improvements relative to prior art in the field of ion engine power conditioning and controls. These improvements have been achieved through the use of high frequency inverters, operating at frequencies between 8 and 10 kHz compared to the previous 1 to 2 kHz.

The use of high frequency inversion has resulted in substantial reduction in size and weight of magnetic components such as transformers and reactors. At the same time, the use of ferrite cores (an unusual application at these power levels) has permitted operation of transformers at lower core losses than possible at these high frequencies with other materials such as silicon steel or nickel-iron. The low core losses and low copper losses, directly resulting from the small size, have resulted in transformers with 98 percent efficiency at 10 kHz.

The use of high speed power transistors (switching in 0.25 micro-second) has also made switching at 10 kHz possible with low losses in transistors, characteristically 98 percent switching efficiency.

The use of ferrite core transformers and high speed power transistors has been made possible by modularization of the power system, adding the outputs of 200 to 250 watt inverters to obtain kilowatt total outputs. This modularization technique is necessary in view of the maximum current limitation of available high speed transistors, and the maximum size of available ferrite cores, and also avoids the inefficiency of equalization techniques required when using parallel transistors.

More recently high speed transistors have become available with higher current rating than those available at the time the systems described in Table 4-2 were designed. However, reliability of these transistors is still not known. Moreover, even when their reliability is established these systems will still require modularization as defined by available ferrite cores and for the higher engine power levels.

RELIABILITY ANALYSIS

An important question to be answered before a system design can begin is the total number of modules to be employed in building up to a high power electric propulsion system. There are several important considerations which affect the answer to this question; the foremost of which are 1) system reliability and weight tradeoffs, and 2) power matching. A

combination of these two factors is used to choose the module size for the final propulsion system designs. The system weight-reliability tradeoffs are discussed here. The power matching studies are presented in the following section.

As is well known, the establishment of component reliability figures at high confidence levels requires costly and time consuming test programs. It is possible, however, to build up system reliability through redundancy techniques even though component reliability is either somewhat low or possibly not established. Redundancy, whether series, parallel, or standby, will increase propulsion system weight. It is desirable, therefore, to determine the method by which the requisite reliability can be obtained with a minimum addition to system weight.

The reliability theory for modularized ion engine systems is based on an extension of the classical concepts currently used in reliability engineering. As discussed in References 1 and 8 the only applicable technique for increasing the reliability of a thruster system is standby redundancy. It was shown that the reliability of a system can be increased to any desired level if enough standbys are employed. However, in order to minimize the weight penalty, a computer program was developed which determines the combination of operating and standby modules such that the redundant engine system weight is a minimum for a given desired system reliability. Once the number is found, the optimum module size (from a weight-reliability standpoint) for an engine system of a given power level is defined. As discussed here the reliability analysis and computer program described in Reference 1 have been modified to include the more general type of solar-electric spacecraft where thrusters may be switched on as well as off during the course of an interplanetary mission.

Standby Redundancy with Switching

It is shown in detail in the Power Matching Studies Section that, due to variations in available power and output voltage of the solar panel, engine and power conditioning modules will be switched during the course of the mission. In each case the modules and their respective subsystems will be designed such that disconnected modules can be reinstated. From a reliability point of view the shut down modules can then be considered as standbys.

Since, in the general case, solar-electric spacecraft can in their heliocentric trajectories travel in toward and then away from the sun, the reliability analysis of a standby redundant modularized system must be extended to include missions in which modules are turned both on and off at different times. As will be shown the intervals of time where modules are turned on and those where modules are turned off can be analyzed separately and then fitted together at the proper points in the proper sequence. An approach different from that taken in Reference 1 and more suitable to numerical computation has been used and will be discussed here. The results of a model example of this theory are shown in Figure 4-9 and Table 4-3.

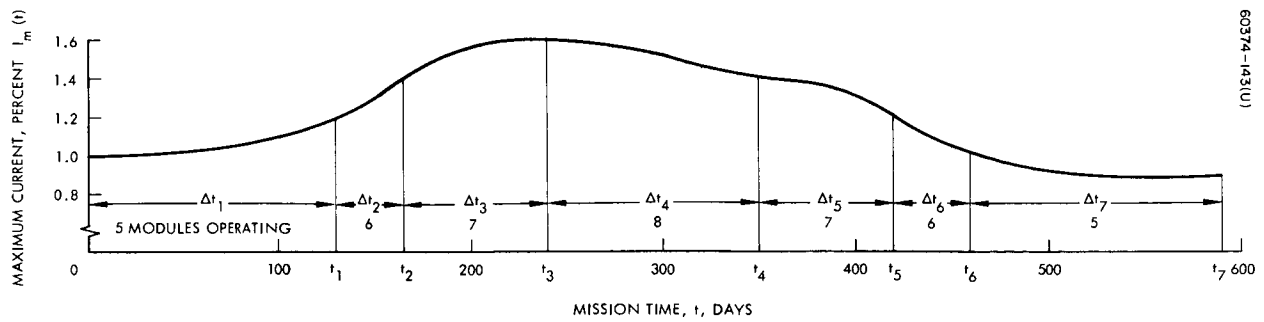


Figure 4-9. System Reliability Calculation, Model Example

TABLE 4-3. SYSTEM RELIABILITY CALCULATION – MODEL EXAMPLE

	t_1	t_2	t_3	t_4	t_5	t_6	t_7
P_0	0.67706	0.86589	0.74527	0.58878	0.74527	0.86589	0.67706
P_1	0.26405	0.12469	0.21911	0.31114	0.21911	0.12469	0.26405
P_2	0.05149	0.00898	0.03221	0.08222	0.03221	0.00898	0.05149
P_3					0.00314	0.00043	0.00669
P_4						0.00001	0.00065
P_5							0.00005
r_0	0.67706	0.58626	0.43692	0.25769	0.19205	0.16629	0.11259
r_1	0.26405	0.31306	0.36177	0.34943	0.31688	0.29833	0.34590
r_2	0.05149	0.08359	0.14977	0.23691	0.26142	0.26759	0.26847
r_3					0.06398	0.09093	0.14870
r_4						0.01034	0.04690
r_5							0.00940
R	0.99260	0.98272	0.94846	0.84403	0.83433	0.83348	0.83196

Adding Modules

Suppose that there are m_0 operating and n_0 standby modules at $t = 0$ and that at times t_μ , $\mu = 1, 2, \dots, v$ an additional module is turned on. The reliability $R(t_\mu)$ of the system in $(0, t_\mu)$ can be calculated as follows.

By definition let

$P_n(t_\mu)$ = probability that exactly n failures occur in $(t_{\mu-1}, t_\mu)$

$r_n(t_\mu)$ = probability that exactly n failures occur in $(0, t_\mu)$.

It is clear from their definitions that $r_1(t_\mu) = P_1(t_\mu)$. For other values of n , $r_n(t_\mu)$ may be calculated recursively using the relation

$$r_n(t_\mu) = \sum_{\alpha=0}^n r_\alpha(t_{\mu-1}) P_{n-\alpha}(t_\mu). \quad (17)$$

The system reliability $R(t_\mu)$ may then be found from

$$R(t_\mu) = \sum_{n=0}^{m_0} r_n(t_\mu) \quad (18)$$

Turning Off Modules

Suppose that there were m_0 operating and n_0 standby modules at $t = 0$ and that v modules have been turned on, the last at time t_v . Now it is desired that modules be turned off, one at each of the times $t_{v+1}, t_{v+2}, \dots, t_{v+k-1}$, and that the reliability $R(t_k)$ at the end of this mission be known. Using the previously described method the probabilities $r_n(t_{v+1})$ can be determined for $n = 0, 1, \dots, m_0$. Remembering that one operating module becomes a standby at time t_{v+1} , it is seen that the probabilities $r_n(t_{v+2})$ are given by

$$r_n(t_{v+2}) = \sum_{\alpha=0}^n P_\alpha(t_{v+2}) r_{n-\alpha}(t_{v+1}); \quad n = 0, 1, \dots, m_0 \quad (19)$$

$$r_{m_0+1}(t_{v+2}) = r_0(t_{v+1}) P_{m_0+1}(t_{v+2}) \quad (20)$$

With a similar recursion for the other r_n 's, the mission reliability may then be calculated from

$$R(t_k) = \sum_{n=0}^{n_o+k-1} r_n(t_k) \quad (21)$$

The statistics of the exact number of failures $P_n(t_\mu)$ are taken to be the usual Poisson Law

$$P_n(t_\mu) = \frac{e^{-\lambda_\mu \Delta t_\mu} (\lambda_\mu \Delta t_\mu)^n}{n!} \quad (22)$$

where

$$\Delta t_\mu = t_\mu - t_{\mu-1} \quad (23)$$

and the failure rate λ_μ is given by

$$\lambda_\mu = (\lambda_p + m_o \lambda_c) \frac{m_o + \mu - 1}{m_o} \quad \mu = 1, \dots, \nu \quad (24)$$

$$\lambda_\mu = (\lambda_p + m_o \lambda_c) \frac{m_o + 2\nu - \mu - 1}{m_o} \quad \mu = \nu + 1, \dots, \nu + k \quad (25)$$

where

λ_p = failure rate of those items dependent on engine size

λ_c = failure rate of those items independent of engine size

Weight-Reliability Optimization

The weight-reliability optimization study for the engine system can now be done for variable power missions.

Before proceeding with the optimization study, formulas for the engine switching times, t_μ , and the weight penalty function, G_{m_o, n_o} , will be developed.

Switching Times

It is shown in the Power Matching Section that the engine switching times t_{μ} are determined from the solar panel maximum current curve, $I_m(t)$. If I_o is the available current at $t = 0$, then the switching times occur when the curve $I_m(t)$ changes from I_o by increments of I_o/m_o . This procedure is illustrated in Figure 4-10. The times t_1 and t_2 occur when the current increases to $I_o + I_o/m_o$ and $I_o + 2I_o/m_o$, respectively. The last turn-on time t_v is determined from the condition that the maximum current $I(t_{\max})$ is less than $I_o + (v + 1) I_o/m_o$. After $t = t_{\max}$, modules are switched off each time the current falls an amount I_o/m_o .

Percent Weight Increase

The weight, w , of a thruster array consisting of m_o initial modules and n_o standbys is

$$w(m_o) = [w_p + (m_o + v) w_c] \left(1 + \frac{n_o}{m_o + v}\right) \quad (26)$$

where

w_p = weight of those items dependent on engine size

w_c = weight of those items independent of engine size

v = number modules turned on during mission.

Thus, the weight penalty incurred to increase system reliability to some desired level by employing a modularized system with redundancy is given by the quantity G_{m_o, n_o} , where

$$G_{m_o, n_o} = \frac{W_{m_o, n_o} - W_{1, 0}}{W_{1, 0}} \quad (27)$$

With C defined by $C = W_c/W_p$, the weight penalty can be written as

$$G_{m_o, n_o} = \left(m_o + n_o + v - \frac{n_o}{m_o + v} - 1\right) \frac{C}{C + 1} + \frac{n_o}{m_o + v} \quad (28)$$

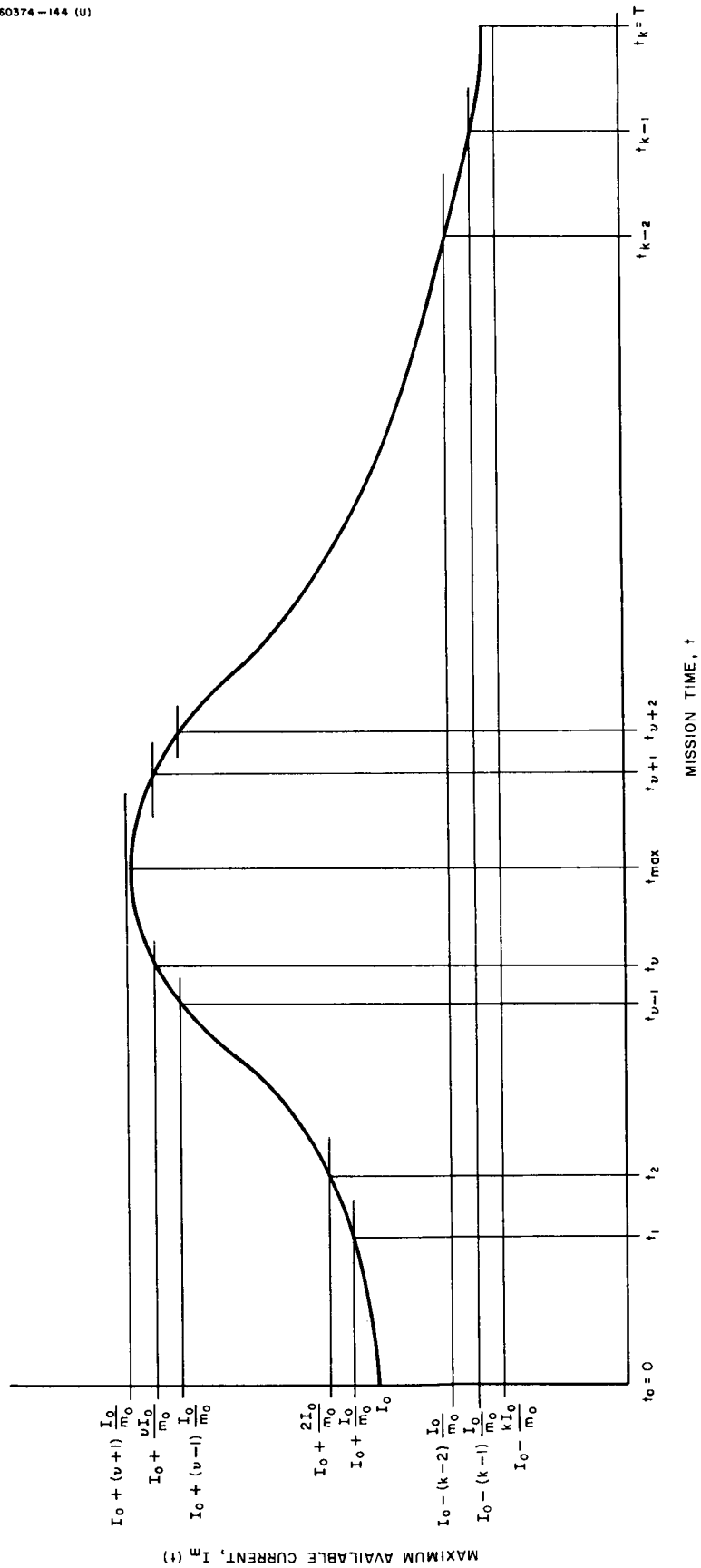


Figure 4-10. Representative Curve of Maximum Available Current $I_m(t)$ versus Time

Optimum Module Size

The procedure for determining the optimum engine module size based on a weight-reliability criterion, may be summarized in the following manner. A desired system reliability, $\bar{R}$ and a range for m_o are chosen. For each value of m_o in its range, the set of times t_{μ} is found and the smallest value of n_o is determined such that

$$R_{m_o, n_o}(t_o, t_1, \dots, t_k) \geq \bar{R} \quad (29)$$

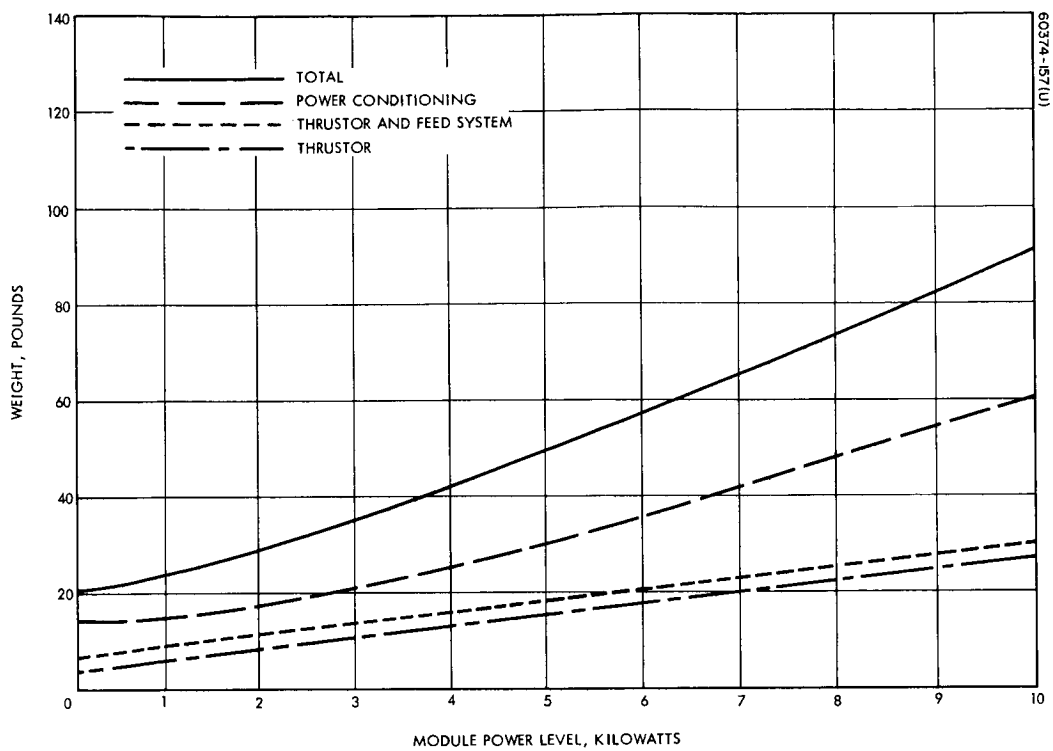
From the set of operating and standby module pairs (m_o, n_o) generated, the one is chosen which makes the percent increase in system weight, G , a minimum.

The weight optimization procedure was carried out by computer for all combinations of the values of $R_{1,0}$, λ_c/λ_p , C and $\bar{R}$ shown in Table 4-4. The Jupiter mission and the 1971, 1973, 1975, 1977, 1979 Mars missions were considered. A representative portion of the results of this study is presented in Appendix F.

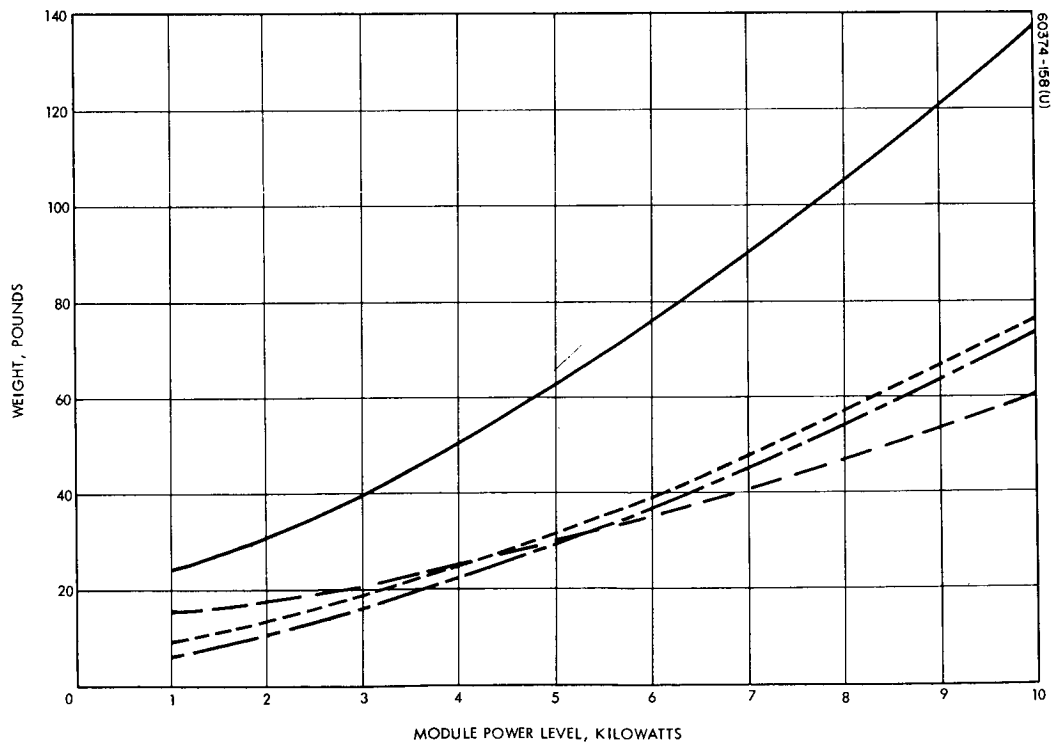
TABLE 4-4. PARAMETERS USED IN WEIGHT OPTIMIZATION STUDY

$R_{1,0}$	λ_c/λ_p	C	$\bar{R}$
0.5		0.07	
0.6	1.0	0.05	0.95
0.7	4.0	0.03	0.97
0.8	10.0	0.01	0.99
0.9		0.003	

In order to determine the optimum module size from a weight-reliability standpoint for a given propulsion system (e.g., 9.6 and 17 kw), it is now necessary to define values for the parameters λ_c/λ_p , $R_{1,0}$, and C . The ratio λ_c/λ_p is rather difficult to establish at this time since virtually no information is available on the failure rates of ion thrusters. However, since a ratio, rather than absolute values, is involved a reasonable estimate can be made. For the systems presently under consideration, a ratio of four was assumed. Although this assumption was based on a careful consideration of the components involved and their possible failure modes, it is only a best



(a) OXIDE CATHODE ENGINE SYSTEM



(b) LIQUID CATHODE ENGINE SYSTEM

Figure 4-11. Module Subsystems Weight versus Power

guess at this time. (However, it should be noted that the optimum module size is relatively insensitive to the value of λ_c/λ_p - e.g., see Appendix F.) The parameter $R_{1,0}$ is the most difficult to establish since it assumes a knowledge of the failure rate of the thrust device. Since this information is not available, a relatively conservative value of 0.8 was chosen for the reliability of an unmodularized system. The magnitude of C for present day Hg bombardment engines can be found by considering the scaling studies presented previously.

The scaling studies which have resulted in functional relationships between module size (power level) and module weight are summarized for the oxide cathode and liquid cathode engine systems in Figure 4-11. Using these data, along with the assumptions that the overall system reliability must be greater than or equal to 0.99, that $\lambda_c/\lambda_p = 4$, and that $R_{1,0} = 0.8$, the optimum module size for the four specific systems under consideration can be found.

The graphical technique employed in determining the optimum module size is demonstrated in Figure 4-12 for each of the model systems. In these graphs the weight of the 17 and 9.6 kw systems are given for various basic system building blocks. Included also is the weight penalty for various numbers of redundant modules. Since the reliability analysis provides the number of redundant modules required as a function of the number of initial operating modules (i.e., m_0), a single curve can be drawn which represents the total weight of a particular system which meets, in all cases, the overall system reliability requirement. The optimum module size (from a weight-reliability standpoint) is then determined by that value of m_0 which minimizes the total system weight. For example, in Figure 4-12a the heavy curve indicates the various possible system designs for a modularized 17 kw, oxide cathode engine system whose reliabilities are equal to or greater than 0.99. The number of initial operating modules which provide the lightest overall system is 2, yielding an optimum module size of 8.5 kw. A similar inspection of Figures 4-12b to 4-12d gives the results of the weight-reliability optimization for the three remaining systems as shown in Table 4-5.

It is important to note that, since the reliability of a modularized power conditioning system can be built up internally, the power conditioning subsystem is not included in the added increments of weight when standbys are employed (Figure 4-12). In addition, the reservoir weight is excluded. These design considerations are discussed in detail in later sections.

Note that the optimization presented here is in terms of weight-reliability criteria and must be re-evaluated in terms of power matching considerations -- discussed in the following section.

The results provided in Table 4-5 must now be considered from a power matching standpoint. Only then can the module size be chosen for the final system design.

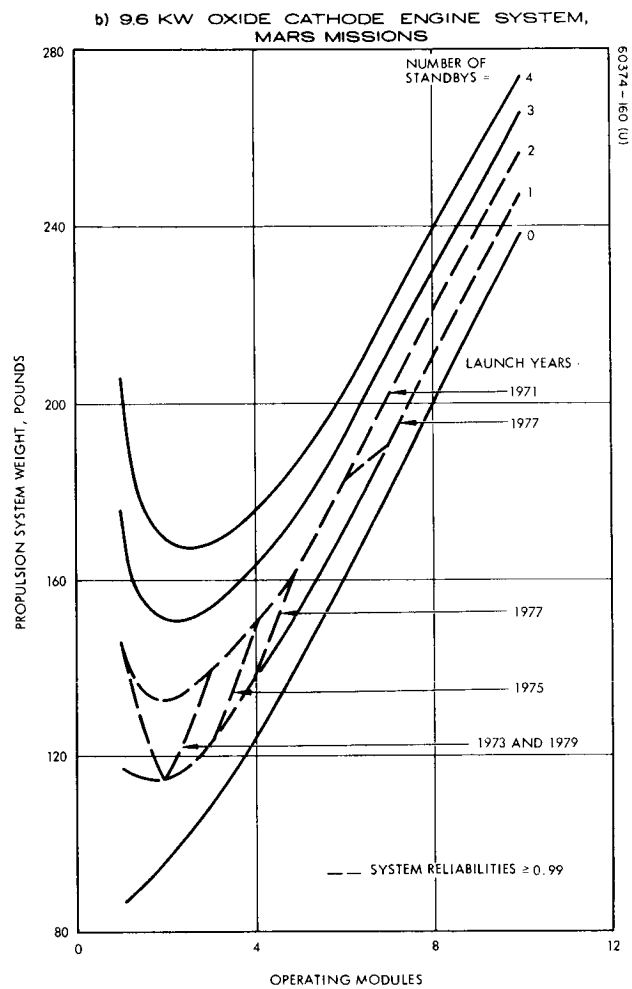
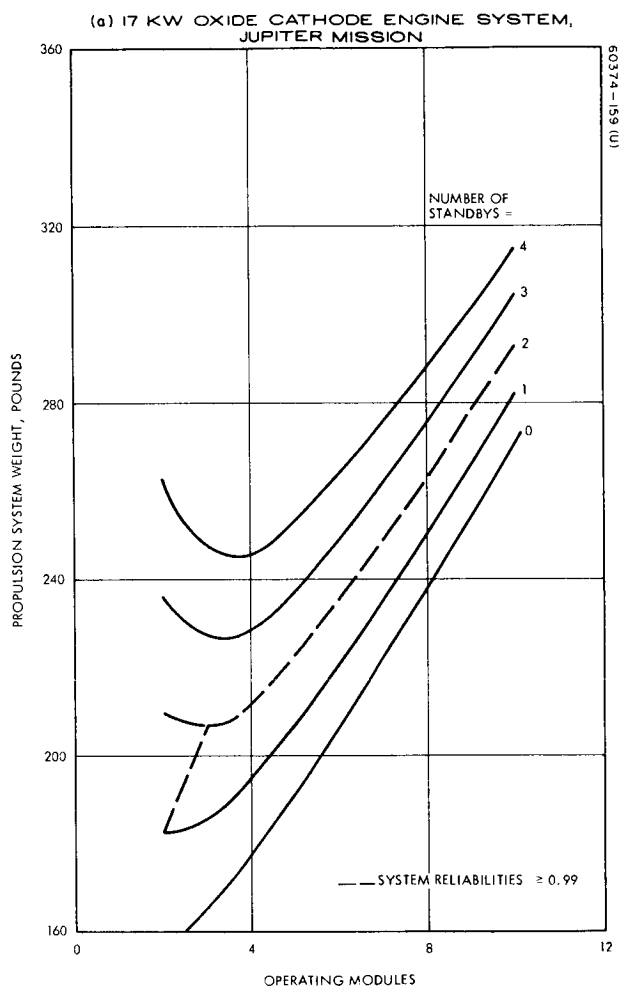


Figure 4-12. Module Size Optimization

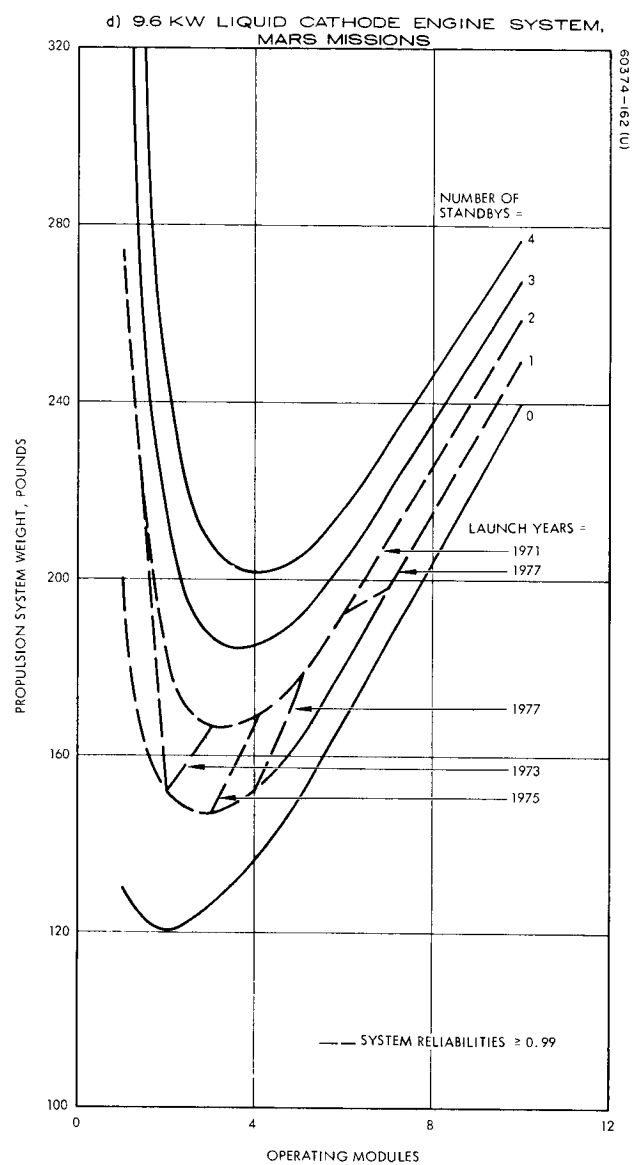
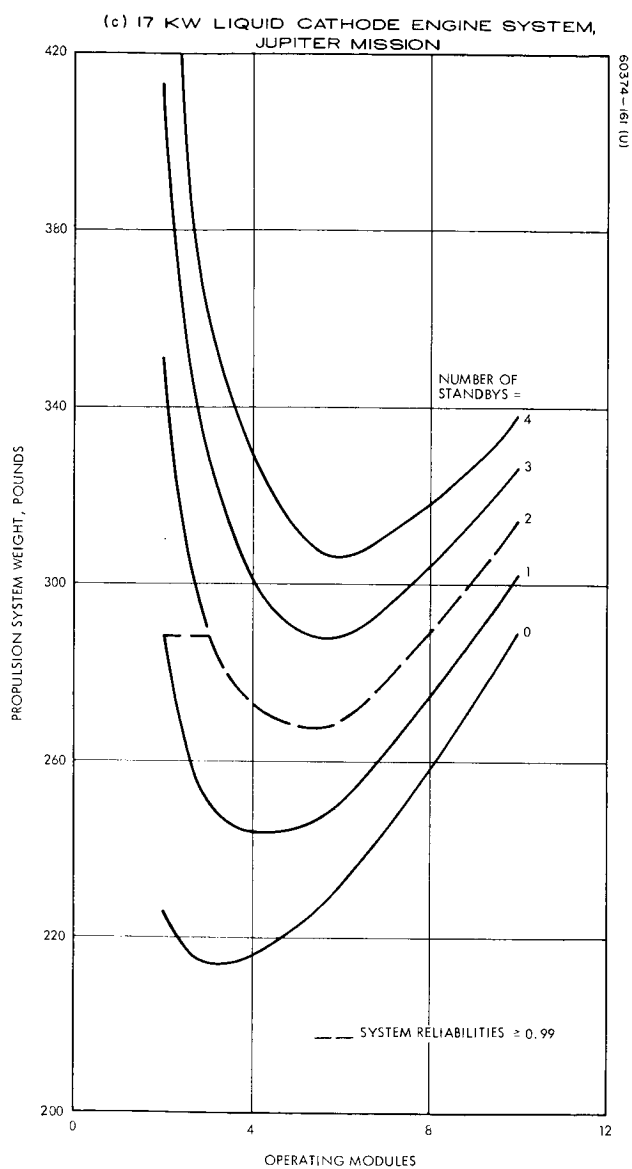


Figure 4-12 (continued). Module Size Optimization

TABLE 4-5 WEIGHT-RELIABILITY OPTIMIZATION

Engine System	System Power Level, kilowatts	Optimum Number of Modules m_o	Optimum Module Size kilowatts**
Oxide-cathode (Jupiter)	17	2	8.5
Oxide-cathode (Mars)	9.6	2	4.8
Liquid-cathode (Jupiter)	17	6	2.8
Liquid-cathode (Mars)	9.6	2 or 3*	4.8 or 3.2

*Depends on launch year.

**Includes power conditioning efficiency.

Thrust Vector Displacement

In the previous sections the procedure for determining the optimum number of operating and standby engines on the basis of a weight-reliability criterion was presented. It was shown that for high power ion propulsion systems, modularization of the thrust system was desirable. A problem which accompanies the modularization of a thruster system is the shift in the location of the center of thrust should engines be shut down due to a decrease in available power or should an engine fail and a standby unit be substituted. Since the accommodation of this thrust vector shift necessitates a mechanical means for moving the complete array, the required motion must be determined and minimized.

An analytical approach to determining the location of the standby modules such that the maximum thrust vector displacement necessary during the mission is minimized is indicated in Reference 1. It is also shown that there is a tradeoff between system reliability and maximum thrust vector displacement. These studies will not be repeated here.

POWER MATCHING STUDIES

A major problem in designing a solar electric propulsion system is the utilization of the maximum available power from the solar panels. In order to guarantee this utilization, the thruster and power conditioning systems must be designed and programmed to provide the proper load at all times to the constantly varying power and voltage outputs of the solar array.

The electrical properties of a solar cell array used to power an electrically propelled spacecraft can be given in terms of a terminal V-I characteristic which is nonlinear (e.g. see Reference 1) and varies continuously with distance from the sun. In order to provide a set of V-I characteristics for the various missions under consideration, a computer program was developed based on the analysis presented in Appendix G. This program provided a family of V-I characteristics as summarized in Figure G-1.

As an example of the system problems associated with power matching consider a hypothetical interplanetary mission described by a set of V-I characteristics such as shown in Figure G-1.

Since the load is specified to be an electric thruster array, some of the power conditioning objectives can be specified. First, the voltage and current out of the solar array must be transformed to new values corresponding to the operating points of electric thrusters. Second, the load resistance that the engines present to the source must be either adjusted or transformed continuously into the value of the source resistance at the maximum power point, $R_{S_0}(t)$, if maximum power transfer is to be achieved. The advantages of a power conditioning system that could keep the engines at their optimum operating points and simultaneously present the correct resistance $[R_{S_0}(t)]$ to the source at each instant of time are obvious.

It was shown in Reference 1 that the problems of voltage and power matching could be solved by proper use of modularized dc inverters in the power conditioning system and a modularized thruster array. It was concluded that unless large numbers of thruster modules are employed power mismatches become intolerable. However, if the thrusters are capable of operating with a variable power input, this problem can be alleviated. Although, in general, ion engines do have this capability, the total power variation (by means of a flow rate variation) to an individual thruster must be held to some reasonable limit. It was shown that the larger the number of initially operating modules the lesser will be the variation in input power per thruster. Thus, from a power matching standpoint large numbers of modules are desirable. However, reliability-weight considerations have shown that employing numbers of modules greater than some "optimum" can result in severe system weight penalties. It is necessary, therefore, that a compromise between reliability-weight and power matching considerations be made when choosing the module size to be employed for a given application.

Consider, for example, the 1977 Mars and the 1973 Jupiter missions. It will be assumed that the current used by the ion engine modules can be varied during the mission so that maximum power is always transferred from the solar array. Under these conditions, the current to the engine system during the missions can be found from their respective trajectories and the power versus distance curve in Figure G-4. These current versus time curves are shown in Figures 4-13 and 4-14. If the engine system consisted of one large engine, its current would have to be varied in accordance with these curves for maximum power transfer to be maintained. The question now arises as to what effect the number of modules initially operating m_0

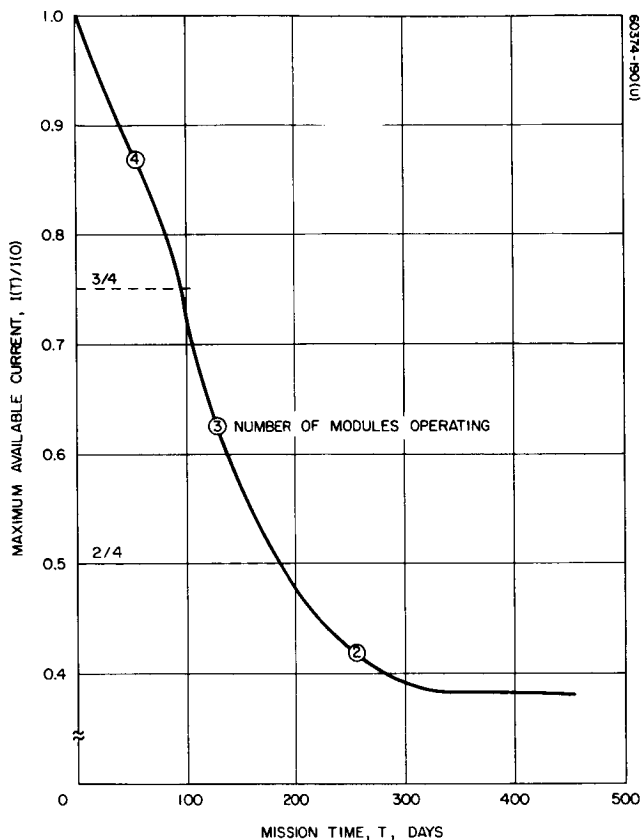


Figure 4-13. Normalized Maximum Available Current versus Time (1977 Mars Mission)

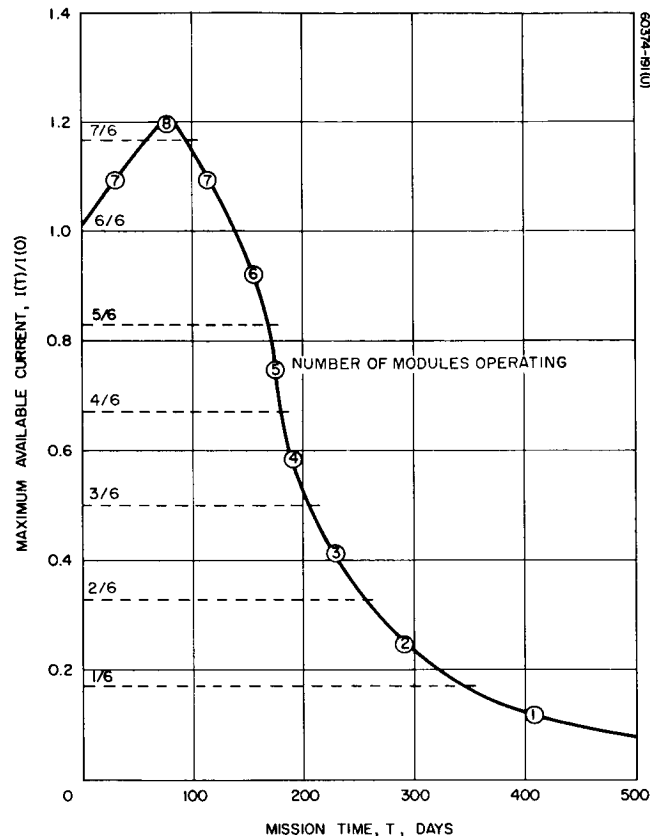


Figure 4-14. Normalized Maximum Available Current versus Time (1973 Jupiter Mission)

has on the maximum current variation required of an individual module. In Figure 4-15 normalized module current is shown for $m_0 = 2$ and 4 during the 1977 Mars mission and for $m_0 = 6$ during the Jupiter mission. From these curves it is seen that the maximum current deviation of a module will occur either during the last or next to last time interval. For example in Figure 4-14 two modules operate during the time when the total current falls from $2/6$ to $1/6$ of its original value, then one module is turned off and the single module operates until the end of the mission. Keeping in mind that each module is rated at $1/6$ of the total current, it is seen that in the next to last interval the two operating modules went from full capacity ($1/6$ each) to half capacity ($1/12$ each) and thus had a 50 percent power deviation. In the last time interval the single current engine goes from full capacity ($1/6$) to 0.47 capacity $0.075 / (1/6)$ at the end of the mission. This latter variation represents a current deviation of 53 percent. Notice that if the total current were zero at the end of the mission, the current deviation would be 100 percent, no matter what value of m_0 is used. It can also be seen that to improve the

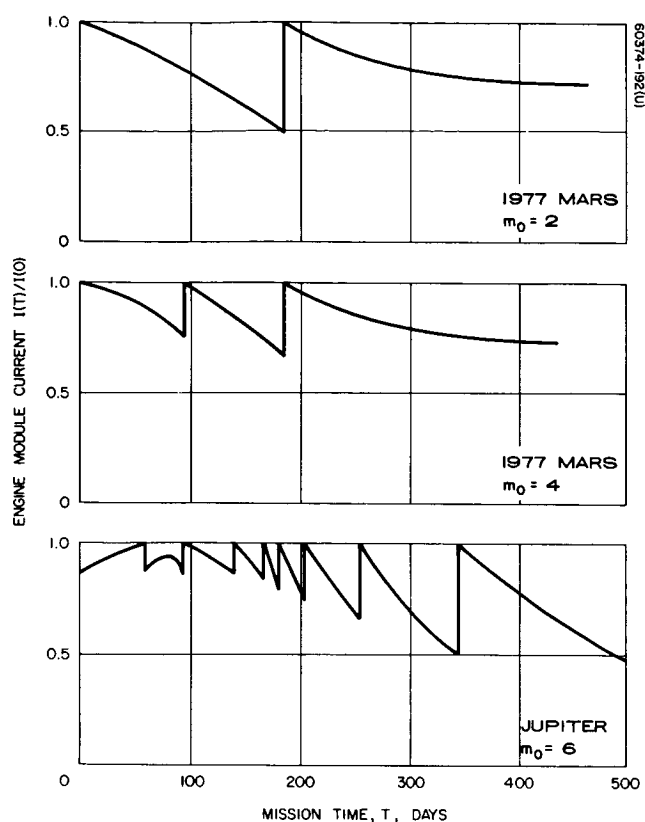


Figure 4-15. Normalized Single Engine Module Current versus Time

current regulation m_0 must be large enough so that the current at the end of the mission $I(t)$ is greater than $I(0)/2m_0$.

The maximum percent current changes are given in Table 4-6 for modules corresponding to $m_0 = 2, 4$, and 6 operating during the 1971, 3, 5, 7, 9 Mars mission. These data indicate the advantage of the higher number of initially operating thrusters, m_0 , in terms of the beam current range over which an individual module must operate to accomplish power matching. On the other hand the weight-reliability optimization (Table 4-5) showed that an m_0 equal to 2 (3 in some cases) was desirable. Although, at this time, the penalty (weight and/or performance) associated with operating an ion engine over a wide power range is quantitatively unknown, it is in general desirable to minimize the required power variation. For this reason a compromise choice of m_0 equal to 4 will be made for the propulsion systems for the Mars missions considered in this report.

TABLE 4-6. MAXIMUM CURRENT DEVIATION IN ANY THRUSTOR MODULE FOR MARS MISSIONS (PERCENT)

Launch Year	$m_0 = 2$	$m_0 = 4$	$m_0 = 6$
1971	50	33	25
1973	50	33	30
1975	50	33	26
1977	50	35	24
1979	50	33	25

The effect of m_0 on the required operating power range of an individual thruster for the Jupiter mission is shown in Figure 4-16. In this case the maximum thruster power deviation occurs in the last switching interval for cases where $m_0 \leq 6$ and the next-to-last interval when $m_0 > 6$. The heavy line in Figure 4-16 then represents the maximum engine current variation versus m_0 . (This curve is valid only for $m_0 < 14$. However, this range covers the region of interest since higher values of m_0 result in prohibitive weight penalties as determined by reliability considerations — e.g. see Figure 4-12.) Since the curve is flat beyond $m_0 = 6$, there is no advantage to a larger number of modules. However, there is reason for having no less than six thrusters initially operating since then the power matching penalty becomes quite severe. Therefore, based on these power matching considerations, a m_0 of six will be chosen for the propulsion system to be designed for the Jupiter mission described in this report. For the oxide cathode engine system, this choice which is off-optimum from a weight-reliability standpoint results in a weight penalty of 52 pounds (~ 3 lb/kw) as shown in Figure 4-12. However, it should again be pointed out that this weight penalty must be viewed along with the possible weight and/or performance penalties that would be incurred should an engine have to operate over an 85 percent power range (i.e. if $m_0 = 2$, the optimum from reliability considerations).

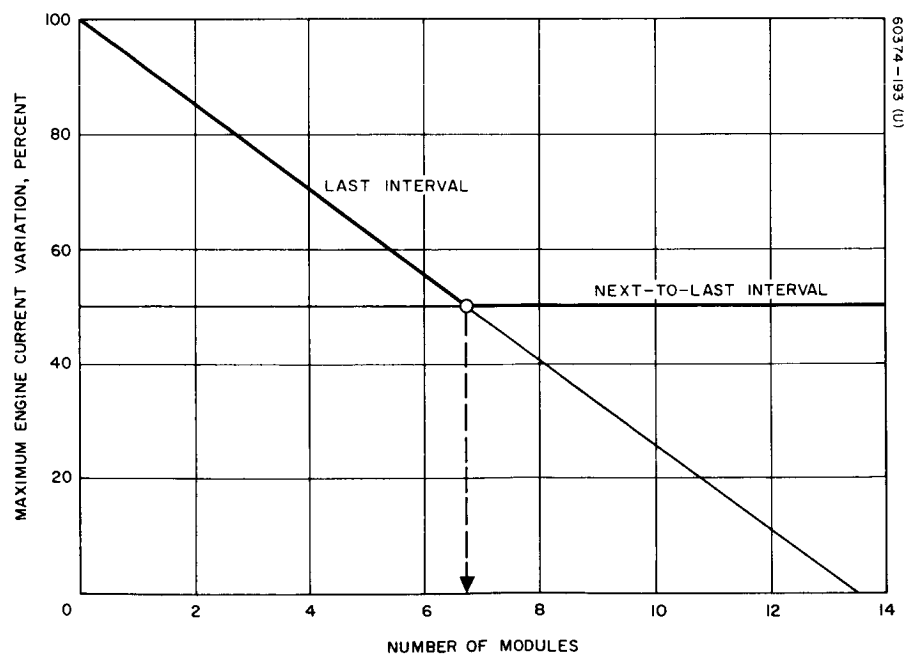


Figure 4-16. Engine Current Variation for Jupiter Mission as a Function of Number of Initially Operating Modules

SUBSYSTEM LAYOUTS

Based on the mission requirements and booster vehicle capabilities, four ion engine systems were chosen for detailed consideration: 17 and 9.6 kw systems employing oxide cathode thrusters, and 17 and 9.6 kw systems utilizing liquid cathode thrusters. A compromise between reliability-weight and power matching considerations, as well as the obvious desirability of limiting thruster design and development programs then led to a choice of a 25 cm engine module as the building block for all four systems. Thus, the 17 kw (Jupiter) system utilizes six operating thruster modules each with an associated power conditioning panel and feed system, plus two standby thrusters. The 9.6 kw (Mars) system requires four operating plus two standby modules. By means of the scaling study results, a preliminary layout of each of the major propulsion system subsystems can now be established.

Thrusters

This discussion summarizes the key design parameters leading to the design of the 25 cm oxide cathode thruster module. As previously shown, an engine size is defined by specification of the thruster module power level (Figure 4-4). Since the thruster module power level has been established at 2.6 kw (maximum), an engine diameter of 25 cm has been chosen. At maximum power this corresponds to approximately 1A of beam current or 20 millipounds of thrust at 3500 sec/Isp.

Design parameters associated with the thruster optics are electrode spacing, electrode thickness, hole size, and hole distribution. It has been shown under "Subsystem Evaluation and Scaling Study" that the electrode spacing and hole size may be chosen within a given range specified by Equations 3 and 6. For the example design, d and w are chosen as 3 mm and 6 mm, respectively. In addition, Equation 12 shows that the associated minimum, accel electrode thickness to assure 1 year operation is 1.9 mm. The screen electrode may be somewhat thinner and has been chosen to be 1.0 mm. The hole distribution (distance between centers) is defined by the ratio of open area to total electrode area (γ). It may be shown that for $\gamma = 0.6$, the above specifications lead to distance of 7.5 mm between centers.

Electrolytic tank studies have shown that the diameter of the accelerator hole should be 0.75 that of the screen aperture. This ratio allows the electrode a greater margin on life as limited by charge exchange erosion. The optics will not be affected if the accel holes are reduced to 4.5 mm, since the ratio of accel thickness to hole size is still under one-half ($1.5/4.5 = 0.33$). The above design specifications for the 25 cm thruster module are summarized in Table 4-7.

The arc chamber diameter is already defined (equal to the engine effective diameter). The arc chamber depth is specified as 20 cm. In conjunction with the reverse feed propellant distribution concept (being developed by Hughes Research Laboratories) these arc chamber dimensions

TABLE 4-7. THRUSTOR DESIGN SPECIFICATIONS

Active thruster module diameter (D)	25 cm
Accel electrode — molybdenum	
Hole size	4.5 mm
Distance between centers	7.5 mm
Thickness	1.9 mm
Screen electrode — molybdenum	
Hole size — countersunk	6.0 mm
Distance between centers	7.5 mm
Thickness	1.0 mm
Electrode spacing	3.0 mm

should minimize the beam profile nonuniformity. The 25 cm oxide cathode engine uses a cathode of the Hughes Research Laboratories flower configuration. The magnetic field of the 25 cm engine will be derived from permanent magnets equally spaced around the periphery of the engine. A detailed design drawing of the 25 cm thruster module appears in Figure 4-17.

As previously discussed, the optimum module size for both the 17 and 10 kw liquid cathode thruster systems is also 25 cm. Furthermore, the major design difference between the liquid and oxide cathode engines is the aluminum shell requirements of the former. The electrode spacings, hole size, anode diameter, etc., are as summarized in Table 4-7. A design layout of a liquid cathode thruster is shown in Appendix E.

Feed System and Propellant Reservoir

Figure 4-7 shows the layout for the feed system module of an oxide or liquid-Hg cathode thruster. The component specifications of the feed system module are independent of thruster module power level (within the range of 1 to 10 kw). Figure 4-7 also indicates a typical thruster feed system arrangement. A second configuration where propellant is introduced from the front rather than the rear of the thruster is shown in Figure 4-17.

The feed system module components are seen to be a valve, vaporizer (or EM pump), and electrical isolator. The dimensions of these units are indicated on the layout drawing. (The components need not be strung out in a line as shown in the figure. They may be turned about in any configuration compatible with spacecraft constraints.)

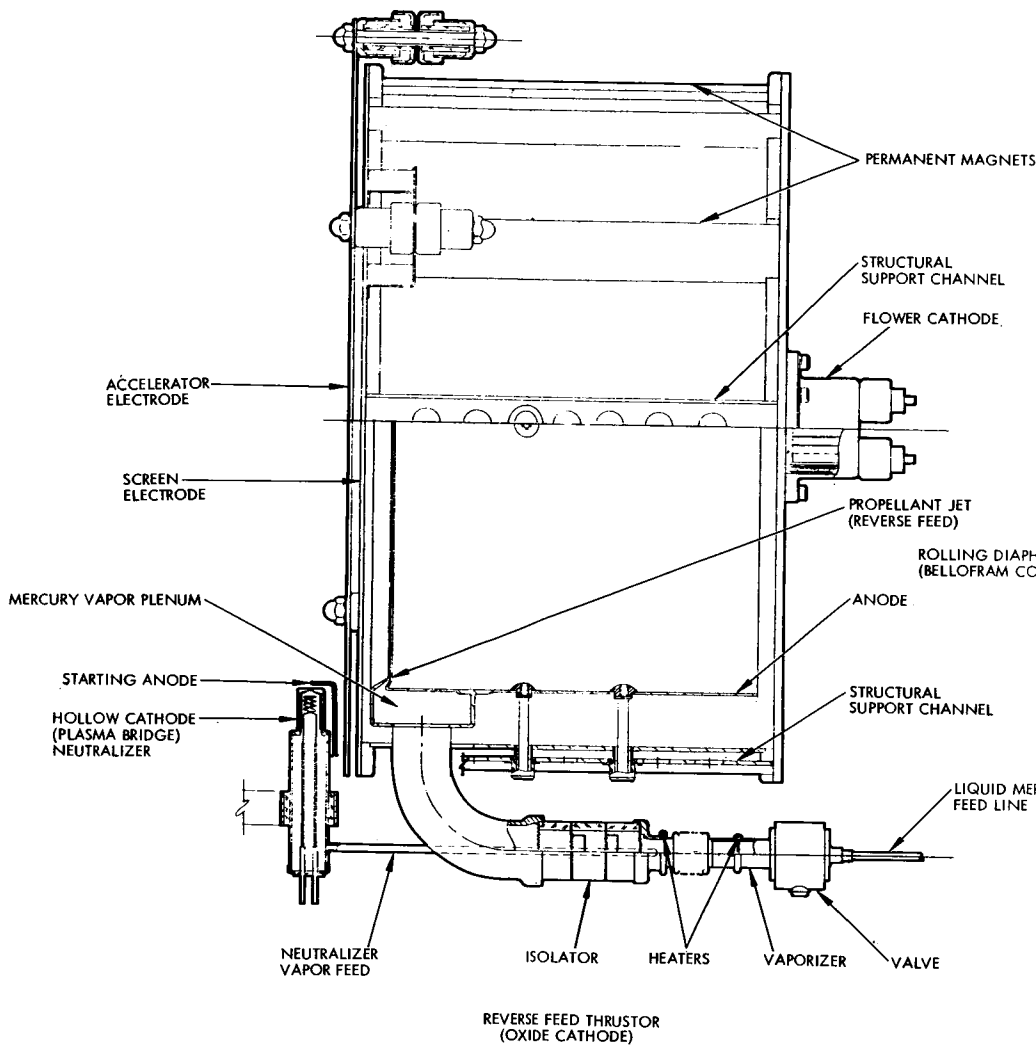
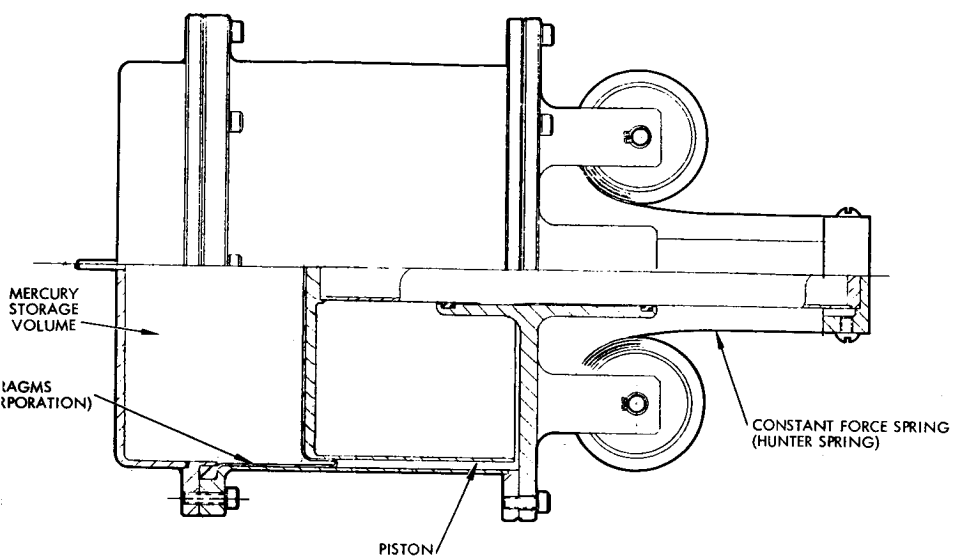


Figure 4-17. Detailed Design of "Reverse Feed" Thrustor Module System - Mercury Bombardment Ion Engine (Oxide Cathode)

60374-213(U)



CURY

2000A - HR Hg STORAGE RESERVOIR

and Feed
e)

4-38

The mercury propellant reservoir is discussed separately from the rest of the feed system since, in both the oxide and liquid-mercury cathode systems, there need not be a single propellant tank associated with each engine module.

The total mercury propellant requirements for the Mars missions (9.6 kw) are (as given in the Mission Analysis Section):

<u>Launch Year</u>	<u>Propellant Mass, pounds</u>
1971	247
1973	273
1975	349
1977	391
1979	394

The propellant requirement for the Jupiter mission (17 kw) is 865 pounds. Based on reliability, spacecraft packaging, and reservoir state-of-the-art considerations, the number of tanks utilized for the Jupiter and Mars missions are six and two, respectively. Since the need for a propellant reserve is still under question, no additional propellant is provided in the designs discussed in this report.

The propellant reservoir layout is shown in Figure 4-18. The dimensions (diameter and height) and weights for reservoirs of both propulsion systems are tabulated in Table 4-8. As indicated above, the same tankage is used for all of the Mars launch years. The single reservoir size was chosen to satisfy the requirement for a spacecraft design that could be considered for Mars missions during the 1970-to-1980 period.

Power Conditioning and Controls

Analysis of the Jupiter and Mars missions indicates engine power requirements (output of power conditioning) of 2.6 and 2.2 kw, respectively, and 2.8 and 2.4 kw input to power conditioning, respectively. The power conditioning and control systems for both engine sizes would be identical in size and number of modules. The 2.6 kw system would be slightly heavier than the 2.2 kw system due to a small increase in transformer weight. Other differences in the two systems (e.g. 1700 volts beam voltage for Jupiter mission and 2200 volts beam voltage for Mars mission) do not influence the basic design of the power supplies. The voltage and power levels used in the description of the power supplies are for the Jupiter mission. However, the techniques and functions described are equally applicable to both the 2.6 kw and the 2.2 kw systems.

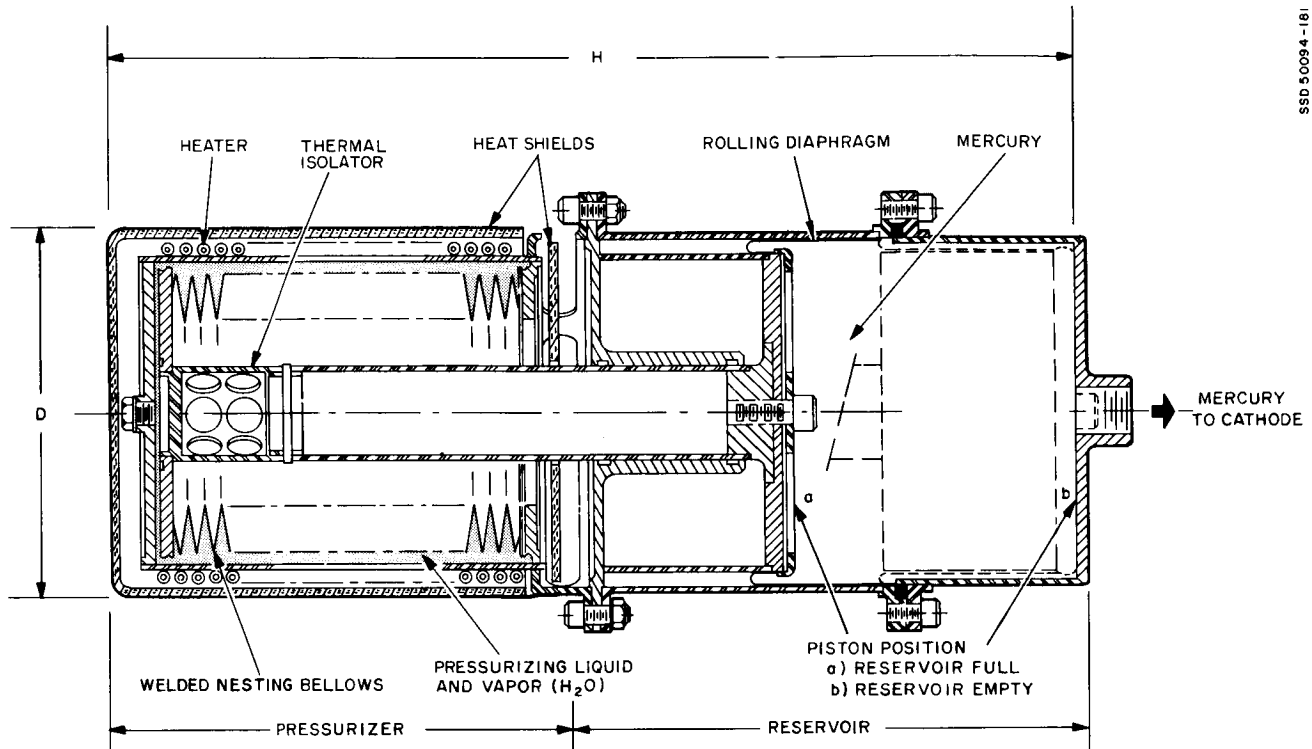


Figure 4-18. Vapor Pressurized Mercury Reservoir for Both Oxide Cathode and Liquid Mercury Cathode Engine Systems

TABLE 4-8. RESERVOIR DESIGN SPECIFICATIONS

Mission	Total Power, kilowatts	Total Propellant, pounds	Number of Reservoirs	Reservoir Capacity, pounds	Dimensions		Empty Reservoirs Weight, pounds
					Diameter, centimeters	Height, centimeters	
Jupiter	17	865	6	145	28	25	10
Mars							
1971	9.6	247	2	200	32	25	12
1973	9.6	273	2	200	32	25	12
1975	9.6	349	2	200	32	25	12
1977	9.6	391	2	200	32	25	12
1979	9.6	394	2	200	32	25	12

Detailed breakdowns of the power conditioning and control systems for the Jupiter mission 2.6 kw oxide cathode engine system and 2.6 kw liquid cathode engine system are presented in Tables 4-9 and 4-10. These tables list the number and types of power conditioning supplies required for each system along with their power, voltage, and current rating. The total number and type of inverter modules, both operating and standby, for each supply, and an indication of the pertinent control functions are given. Also included is a specification of salient system parameters such as overall efficiency, specific weight, and dimensions.

Figure 4-19 shows the circuit block diagrams of the two power conditioning systems. As indicated, dc voltage adding techniques are used in the main beam and accelerator supply subsystems, whereas ac voltage adding techniques are employed in the other supplies. The block diagrams show the incremental and linear control systems which automatically start up the engine and maintain its operating point. This system also provides automatic engine restart and shutdown control capability. The micrologic control system is employed primarily to turn on and turn off the inverter modules to maintain the power conditioning supplies at their rated output voltages. The linear control system, on the other hand, controls the engine heater systems and maintains the ion engine operating current at programmed or set levels.

A detailed description of the function and operation of each of the power supplies and control units depicted in Figure 4-19 has been provided in Reference 1 and will not be repeated here. Furthermore, a power conditioning and control system of the type discussed in this report has been developed and tested under the Hardware Verification Phase of this program and has been described in Reference 3. It is appropriate, however, to point out certain details in the power conditioning and control systems designed during this study to indicate differences with previous work.

Power requirements of the 2.6 kw, oxide cathode, and liquid cathode engines are shown in Tables 4-9 and 4-10. From the requirements it is apparent that a 250 watt inverter module lends itself to efficient use in the beam, arc, and cathode heater supplies which represent the major power demand. Such an inverter module would result in seven operating modules plus two standby for the beam supply, two operating modules plus one standby for the arc supply, and one operating module plus one standby for the cathode heater supply. (Each of these supplies requires one other operating module to perform auxiliary functions such as filtering, etc.)

A 250 watt module is realizable at high efficiency and low weight, with inversion at 20 kHz, using the Solitron MHT7805 transistor. This transistor, rated at 10 amperes collector, 325 volts V_{ce0} , has switching speeds of the order of 0.1 microsecond or better, and results in low switching losses at 20 kHz inversion. The low V_{ce} saturation of 0.5 volt at 5 amperes is also ideally suited to the inverter application.

TABLE 4-9. POWER CONDITIONING SYSTEM CHARACTERISTICS FOR 2.6-KW OUTPUT
OXIDE CATHODE ENGINE (JUPITER MISSION)

Function	Supply Rating		Operating Modules	Standby Modules	Total Modules	Total Weight, pounds	Average Power Efficiency, percent	Total Loss, watts	Transistor Type	Inverter Type, oscillation	Module Output, volts	Control Range, percent	Primary Controlling Parameter
	Power output, watts	Voltage, volts	Current, Amperes										
Beam	1800	1700	1.06	8	2	10	8.78	94	130	MH77805	Self ¹	243 dc	Total voltage
Accelerator	26	3300	0.08	1	0	1	0.7	94	2	MH77805	Self	3300 dc	None
Arc discharge	347	36	9.7	3	1	4	4.0	89	53	MH77805	Driven ²	15 ac	Arc voltage
Cathode heater	260	5	52	2	1	3	1.9	93	20	MH77805	Driven	5 ac	Cathode current
Neutralizer heater ³	100	40	2.5	1	0	1		93	8	MH77805	Chop	40 ac	Neutralizer emission current
Feed system heaters	26	40	0.65	1			0.5	95	2	MH77805	Chop	40 ac	Beam current ⁴
Control				1		1	0.7						
Frame							3.0						
Totals	2560			16	6	20	19.50		205				

Specific weight = 7.0 lb/kw in. Overall power efficiency 92.5 percent. Power conditioning panel area = 895 in².
= 7.8 lb/kw out.

¹ Self-oscillating inverter operating frequency ranges between 16 and 24 kHz, depending on solar cell bus voltage.

² Driven oscillators receive drive power from accelerator module

³ Neutralizer heater supply V, I requirements based on indirectly heated cathode

⁴ I_B is automatically programmed to match solar cell power characteristics

TABLE 4-10. POWER CONDITIONING SYSTEM CHARACTERISTICS FOR 2.6-KW OUTPUT
LIQUID HG CATHODE ENGINE (JUPITER MISSION)

Function	Supply Rating			Operating Modules	Standby Modules	Total Modules	Total Weight, pounds	Average Power Efficiency, percent	Total Loss, watts	Transistor Type	Inverter Type, oscillation	Module Output, volts	Control Range, percent	Primary Controlling Parameter
	Power output, watts	Voltage, volts	Current, Amperes											
Beam	1800	1700	1.06	8	2	10	8.78	94	130	MH77805	Self ¹	243 dc	Incremental	Total voltage
Accelerator	26	3300	0.08	1	0	1	0.7	94	2	MH77805	Self	3300 dc	None	None
Arc discharge	607	30	20.2	4	1	5	5.0	89	76	MH77805	Driven ²	12 ac	0 - 100	Arc voltage
Neutralizer heater ³	100	40	2.5	1	0	1	0.5	93	8	MH77805	Chop	40 ac	0 - 100	Neutralizer emission current
Feed system heater	26	40	0.65					95	2	MH77805	Chop	40 ac	0 - 100	Beam current ⁴
Control				1		1	0.7							
Frame							3.0							
Totals	2560			15	3	18	18.08		218					

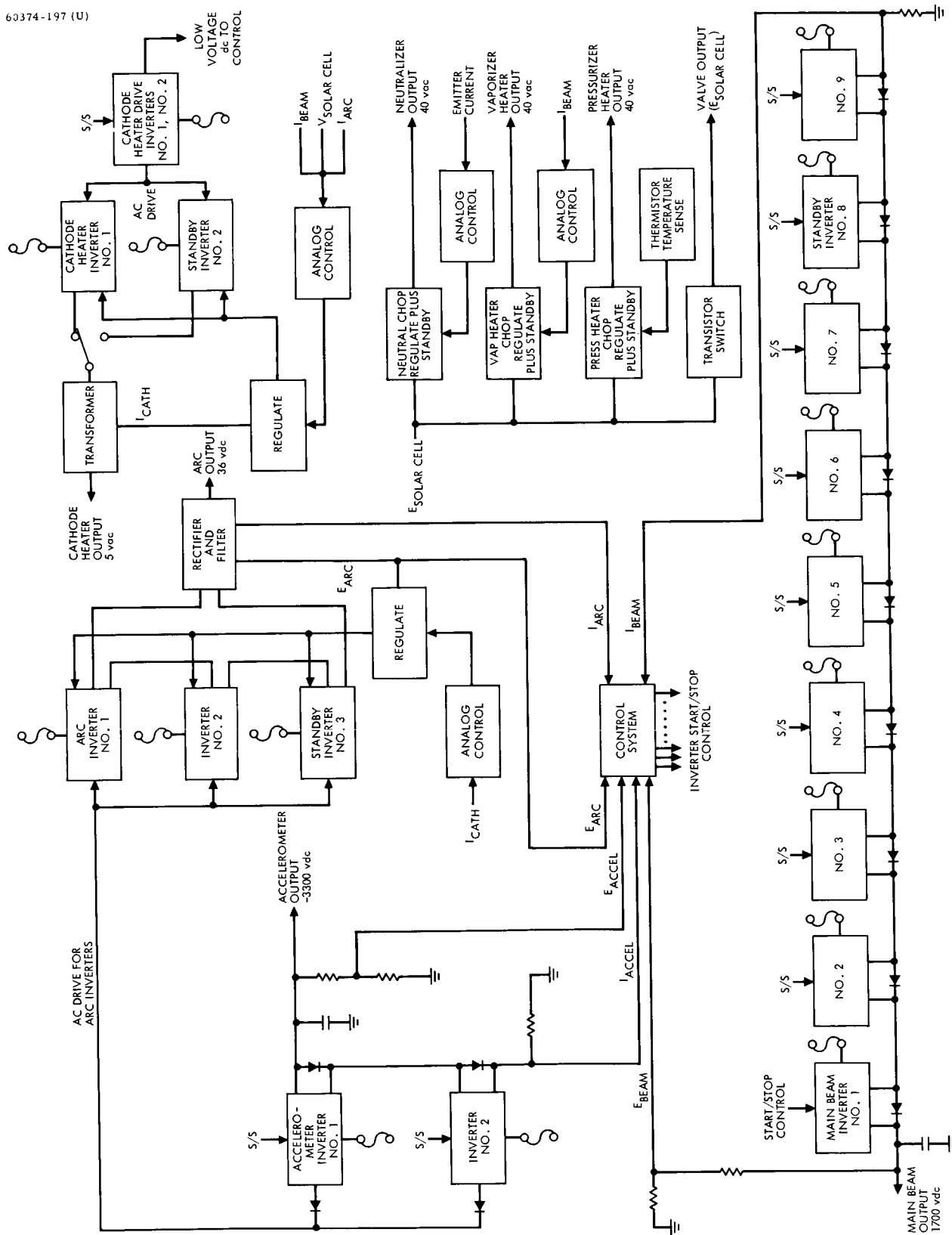
Specific weight = 6.72 lb/kw in. Overall power efficiency 92.5 percent. Power conditioning panel area = 815 in.²
= 7.3 lb/kw out.

¹ Self-oscillating inverter operating frequency ranges between 16 and 24 kc, depending on solar cell bus voltage

² Driven oscillators receive drive power from accelerator module

³ Neutralizer heater supply V, I requirements based on indirectly heated cathode

⁴ I_B is automatically programmed to match solar cell power characteristics



a) Power Conditioning System for Oxide Cathode Engine

Figure 4-19. Circuit Block Diagram, 2.5 kw Output

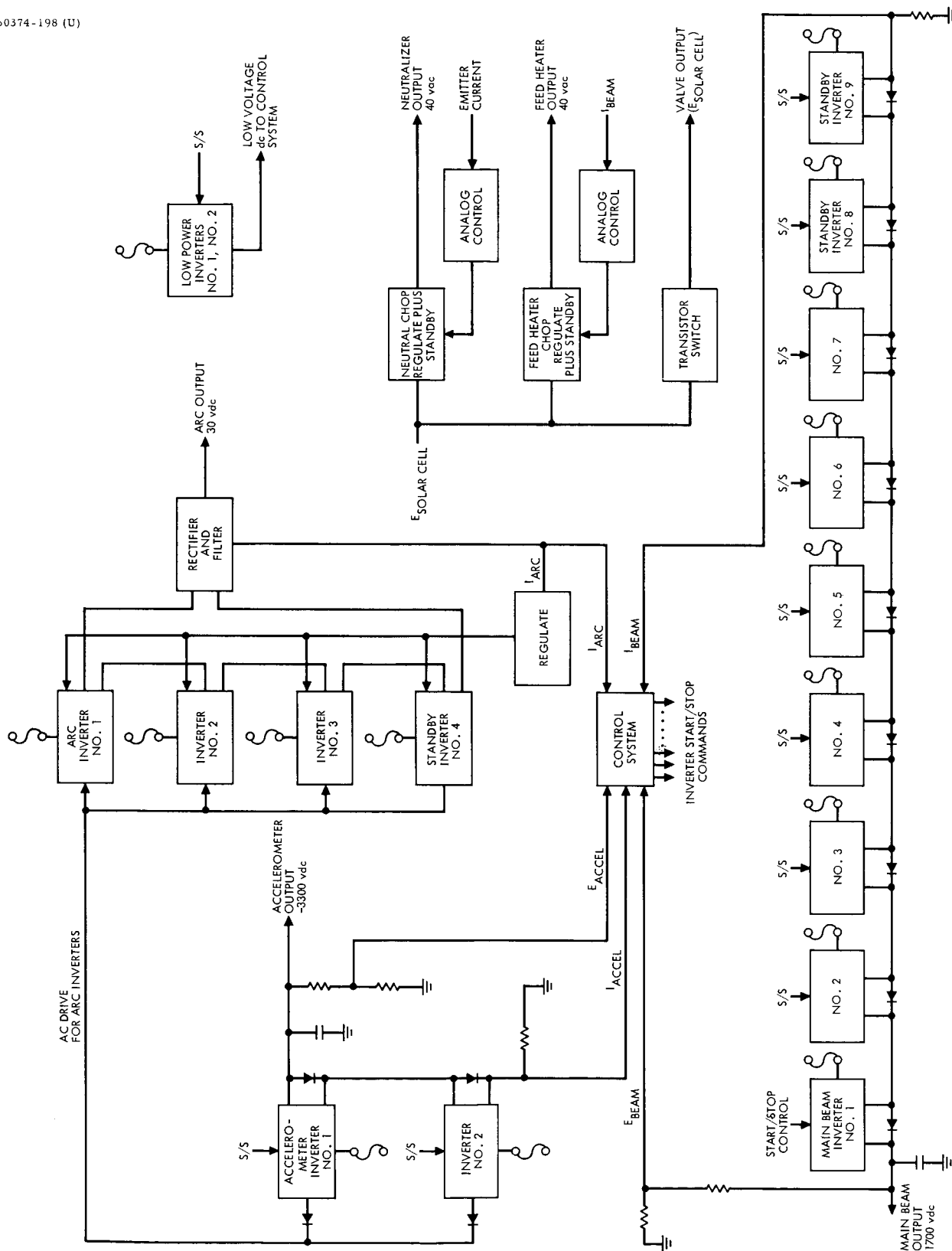


Figure 4-19 (continued). Circuit Block Diagram, 2.5 kw Output

b) Power Conditioning System for Liquid Mercury Cathode Engine

The modules used in the prototype system (Reference 3) typically were rated at 125 watts output at 8 kHz inversion, with 0.5 microsecond switching times using a Delco transistor 2N2580 and ferrite output transformer. The same size ferrite output transformer may be used to deliver 250 watts at 20 kHz, since core losses were virtually negligible at 8 kHz, and will still be small relative to the copper loss at 20 kHz. For the same voltage, half the original turns at twice the wire size would maintain the same flux density, which is saturation limited, not core loss limited. Also twice the current at the same voltage may now be used with one fourth the winding resistance (half turns, twice size), resulting in the same copper losses at twice the output power, with a corresponding improvement in transformer efficiency.

At the line voltage used with the design verification hardware (40 to 60 volts based on SERT II requirement), twice the current per module would increase the percentage saturation loss in the switching transistor (the new transistor has the same drop at the same current as the original type). However, the reduced switching loss in the new transistor, resulting from the 5 to 1 improvement in speed, will offset the increase in saturated loss, resulting in a transistor switching efficiency about equal to that of the original module.

An analysis of system efficiencies and weights (the system includes solar array and transmission line) indicates a substantial advantage to be gained by operating with a 60 to 90 volt solar panel voltage, rather than the SERT II voltage of 40 to 60 volts. (This analysis is given in Appendix H.) The weight saving is equal to 30 percent of the weight of all of the modules in the power conditioning system. Thus, the equivalent improvement in figure-of-merit (kilowatts per pound) of the modules is substantial. This comparison is an illustration of the inadequacy of heavy emphasis on power conditioning weight without a corresponding consideration of optimum solar array voltage.

PROPULSION SYSTEM INTEGRATION AND DESIGN

The design of a modularized ion propulsion system poses thermal, mechanical, and electrical integration problems which must be analyzed and solved prior to any specific definition of the capabilities and limitations of a high power ion propulsion system. The results of these system integration studies have a substantial effect on the design of the various major subsystems as well as the overall spacecraft. Analyses of some of the more important integration problems were presented in Reference 1. The results of these analyses will be reviewed briefly here prior to the discussion of propulsion system design.

Summary of Results of Integration Studies

The major thermal, mechanical, and electrical integration considerations which affect the design of a modularized ion propulsion system are summarized as follows:

Thermal

- 1) Since, in general, high thruster shell temperatures ($>600^{\circ}\text{C}$) are undesirable, ion engine clusters should not be close-packed but should employ a peripheral array type geometry (e.g., Appendix E).
- 2) The power conditioning and control system panels should be designed to provide passive temperature control of the individual modules (e.g., Appendix H). This feature requires that panels be mounted external to the spacecraft (in the freely radiating condition).

Mechanical

- 1) To eliminate the need for a structural cutout in the spacecraft and to thermally isolate the engine system from the spacecraft, the thrusters and propellant reservoirs should be mounted on a tray external to the spacecraft bus.
- 2) An engine array translation mechanism, caging device, and jettisoning mechanism must be provided to meet three specific mission requirements:
 - a) Provide the capability to orient the thrust vector in 2 degrees of freedom (longitudinally and laterally) to correct for spacecraft cg - thrust vector misalignments.
 - b) Survive the boost environment.
 - c) Permit jettisoning of the entire engine system prior to spacecraft retro maneuver. (Does not apply for flyby or out-of-ecliptic missions.)

Electrical

- 1) An electrical power transmission cable must be provided to interconnect the power conditioning and ion engine systems. The cable must be able to transmit this power at ion engine potentials and currents with reasonable cable power efficiencies and specific weights. A flexible cable mounting is required to allow the ion engine array to be translated in two directions during the course of the mission to accommodate a shift in the thrust vector. Also

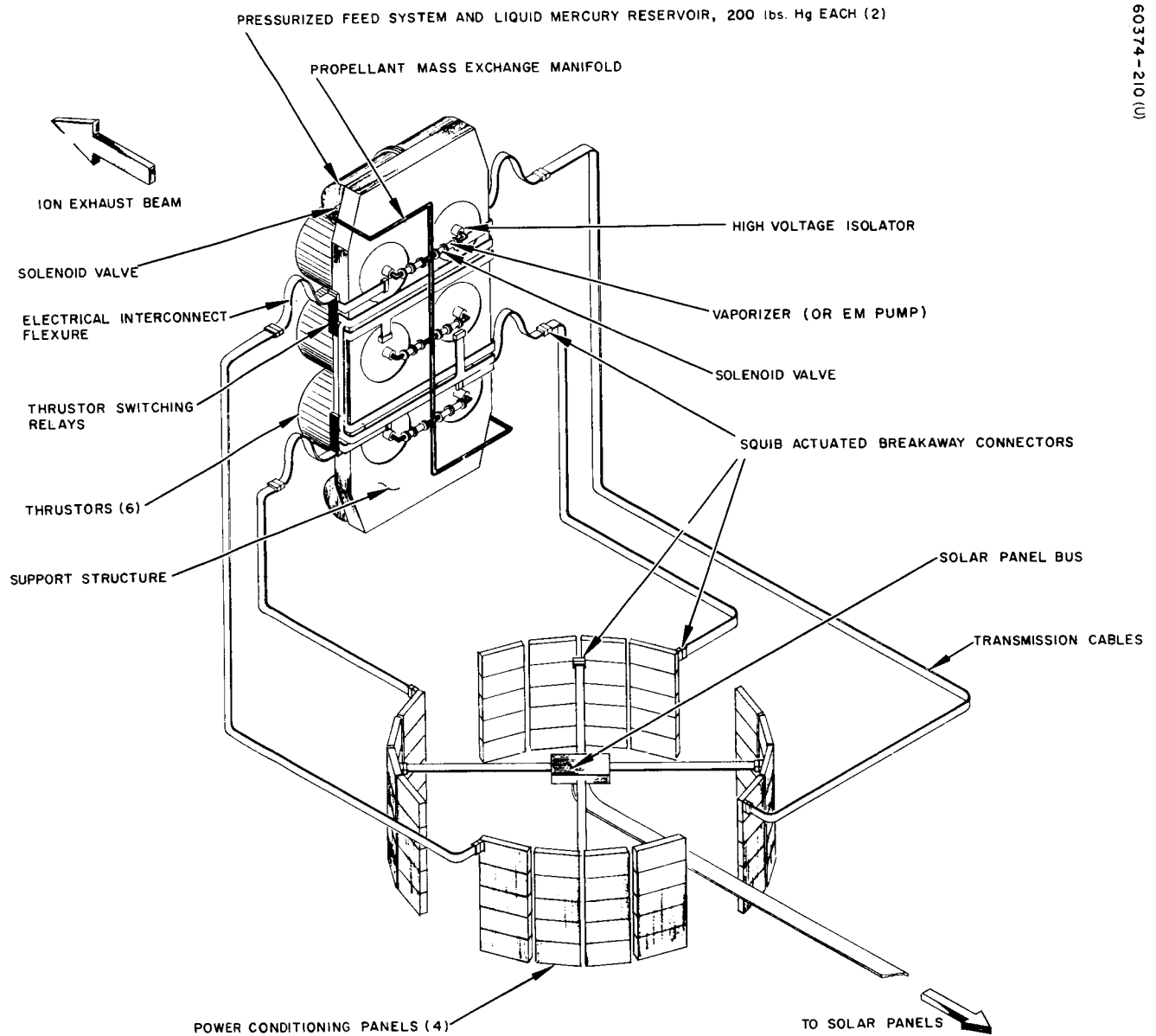


Figure 4-20. Conceptual Drawing of 9.6 Kilowatt Propulsion System

squib actuated cable connectors must be provided to permit disconnecting and jettisoning (if required by the mission) of the power conditioning panels and ion engine modules.

- 2) A switching matrix and a cable harness is required to permit the operation of two standby ion engines from any one of the power conditioning panels.

Conceptual Design of Propulsion Systems

Conceptual designs of two ion propulsion systems have been completed. These system designs result from integrating the propulsion subsystems previously specified, in accordance with the thermal, electrical, and mechanical constraints imposed by the spacecraft integration studies. The two conceptual designs, both of which are based on oxide cathode thrusters, are depicted in Figures 4-20 and 4-21. The following descriptions of the system designs are, however, applicable to either an oxide cathode or liquid mercury cathode engine system. Both systems employ the modularized subsystems which have been discussed in detail in the Subsystems Layout Section.

The basic 17 kw system considered consists of eight 2.6 kw, 25 cm diameter thruster modules, six cylindrical propellant reservoirs, plus associated valves, vaporizers, isolators, and plumbing. The arrangement presented herein assumes that each thruster occupies an envelope 35 cm in diameter to account for necessary thruster components extending beyond the 25 cm basic thruster diameter. The eight thrusters are arranged in two rows of three thrusters separated by two center row thrusters at the top and bottom. The six pressurized mercury reservoirs are mounted above and below the thruster array in two rows of three. The reservoirs have been sized to not exceed one thruster module depth, thereby not increasing the array envelope in depth. The placement of the Hg reservoirs above and below the thruster array, rather than behind the thrusters, resulted from the design constraint which required the mounting of the engine system externally to the spacecraft bus.

The mounting structure which may be fabricated from aluminum alloy sheet is basically a ribbed tray to which the eight thrusters are attached using nonconducting standoff attachments. The mercury reservoirs are also mounted to the tray supported by the flange on the mercury side of the reservoir assembly. The feed system plumbing and components are located aft of the support tray. Sizing the structure for the boost environment, assuming fabricated aluminum construction utilizing approximately 10 gage material, indicates a structural weight of approximately 40 pounds.

As shown in Figures 4-20 and 4-21 each reservoir has a separate valve as does each engine. A basic feature of this design is the capability for crosscoupling between fuel tank modules and thruster modules, provided by the one additional valve. In this manner, any number of Hg storage tanks

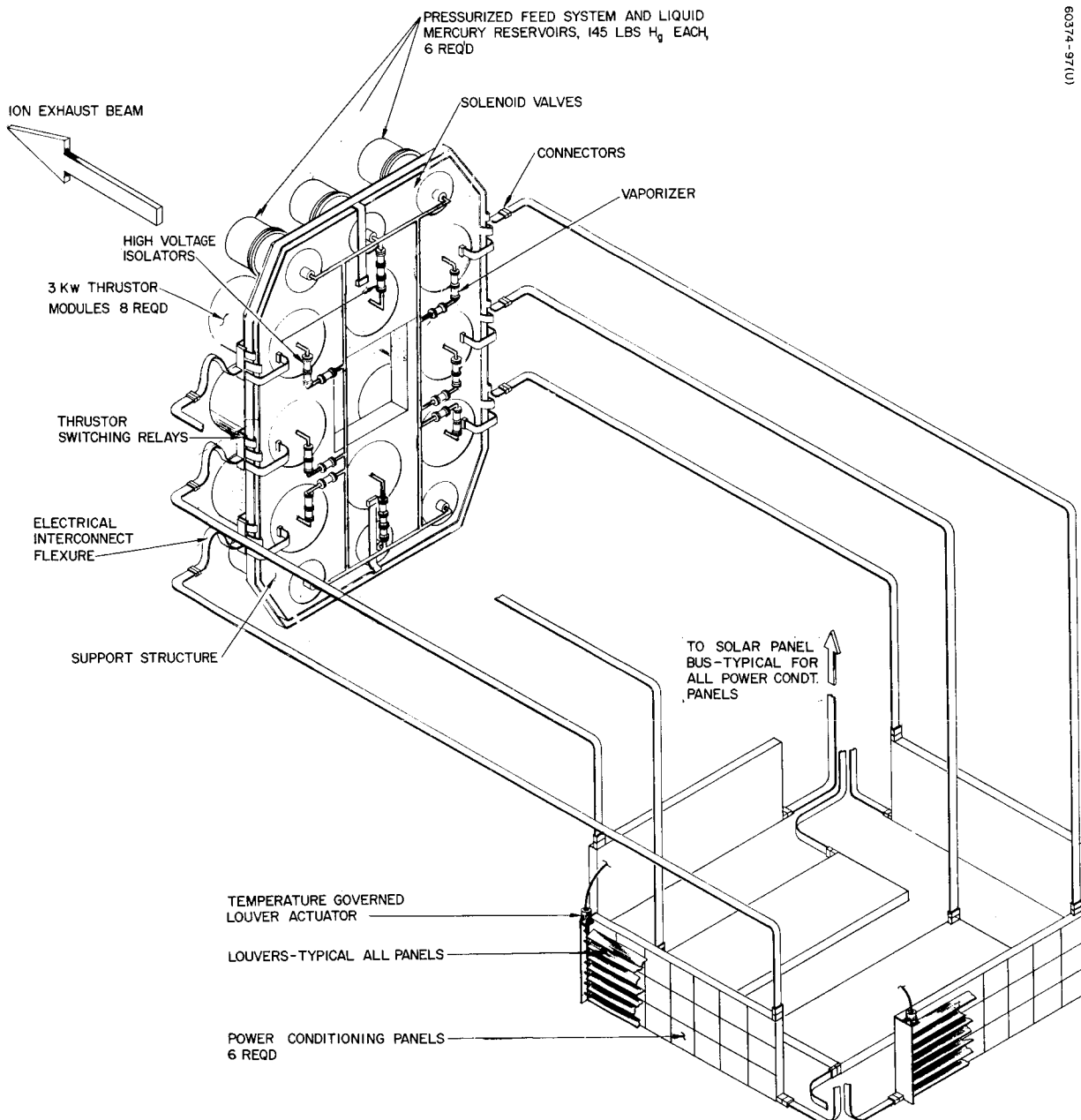


Figure 4-21. Conceptual Drawing of 17 Kilowatt Propulsion System

may be used to feed simultaneously any number of thrusters. This feature is desirable for two reasons, engine switching and reliability of propellant supply system. The general layout is shown in Figure 4-22.

Engine switching is required because the propulsion system employs the standby thruster module concept (in order to raise thruster system reliability). It may be necessary to turn off an engine at random, switching in a standby to replace it. The crosscoupling design allows the fuel, which is at first available to the original operating engine, then to become available to the standby engine.

Engine switching is also a requirement of solar cell power matching whereby the total beam current must be reduced as a function of time. Because of this requirement complete thruster units are turned off at discrete points in the mission. Therefore, an increasing number of standbys become available. The complexity in determining beforehand which engines will be operating at any given time thus becomes enormous. The cross-coupling design eliminates this consideration.

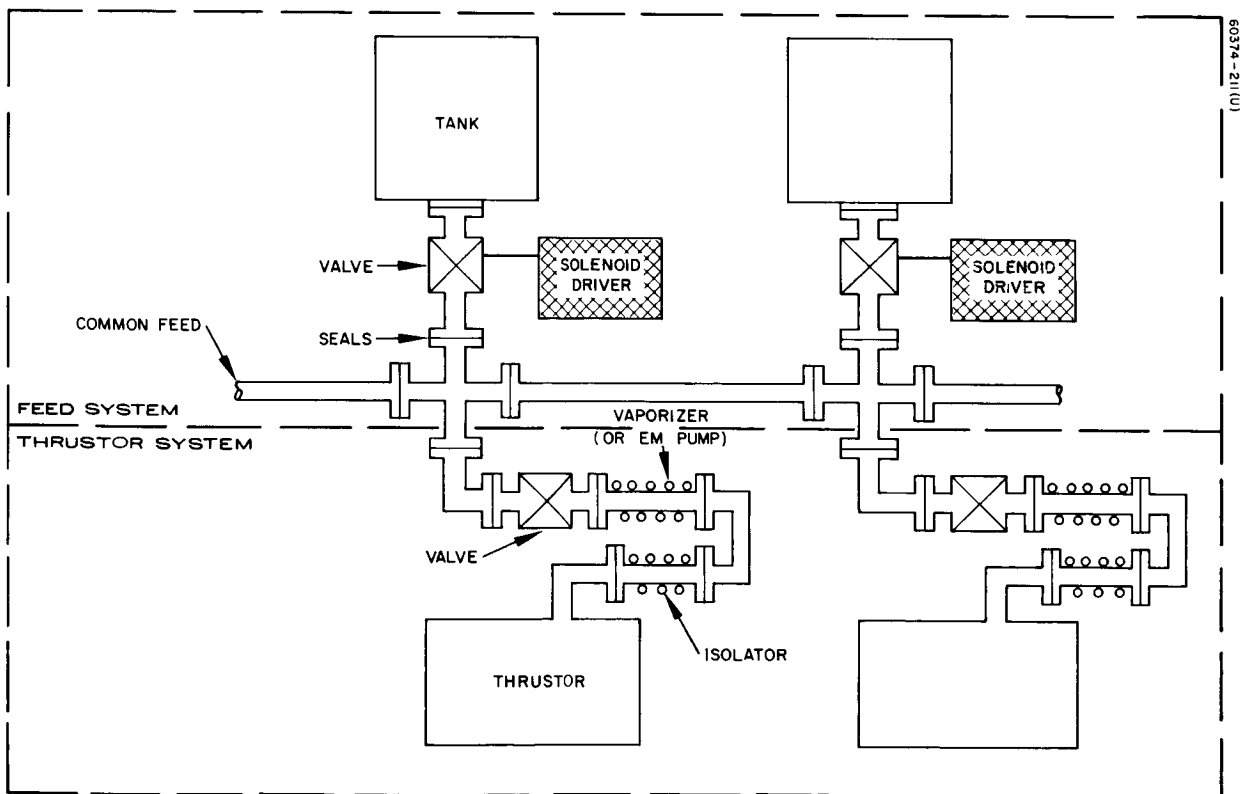


Figure 4-22. General Feed System
Arrangement for Modularized
Propulsion System

Although engine switching could be accomplished with only one propellant tank, the use of multiple tanks is desirable. For example, it is unlikely that a single tank could support, under launch conditions, the large amount of propellant required for these missions (400 to 870 pounds). Therefore, in sizing the tankage operational limitations on the expulsion system must be considered. Present work in this area has involved tanks of much smaller capacity so that extrapolation to too great a capacity would be subject to question.

Furthermore, multiple tanks are desirable from a reliability standpoint. If one tank failed (due to meteoroid puncture for example), the multiple reservoir arrangement allows that tank to be isolated from the propulsion system in such a way that thruster performance continues undisturbed. The punctured tank could be immediately isolated by its own valve, limiting the propellant loss to that remaining in the defective reservoir. To conserve propellant further a slightly more elaborate scheme may be employed. For example, upon a signal that a particular mercury reservoir was leaking, all other tanks would be shut off until all fuel in the leaky tank was utilized. The punctured tank could then be isolated and the other tanks reopened, thus maintaining continuous supply to the operating thruster modules.

The entire engine system, mechanically integrated in the manner described, occupies a 3.5 foot wide by 5.0 foot high by 1.3 foot deep envelope.

As indicated in the mechanical integration discussions, three mechanisms must be incorporated to satisfy the translation, jettison, and caging (i.e., support during the boost phase) of the engine and reservoir array. Using a single device to provide all three functions does not appear attractive since the mechanisms for translation and jettison would be penalized by having to be structurally adequate to survive the high loading (6.5 g longitudinal and 2 g lateral) imposed during boost. The design philosophy adopted allows the boost loads to bypass the engine array translation device and jettison device by caging the entire engine array structure to the spacecraft structure with an independent support system which is disengaged once the spacecraft is in a zero-g environment. In the system proposed for caging the engine tray during boost, stress levels were found to be within acceptable subyield limits. Details for the engine array translation mechanism, caging device, and jettisoning mechanism are presented in Reference 1.

The mechanical integration of the 9.6 kw thruster system utilizes basically the same concepts discussed above for the 17 kw system, although only six 2.5 kw thrusters and two mercury reservoirs are required. This array is 2.5 feet wide and 5.5 feet long.

Figure G-1 (Appendix G) shows the power conditioning panel layout for the oxide cathode propulsion system. The power conditioning panel layout indicates the number of modules and approximate panel area required for each module. The complete 17 kw system has six identical panels of the type shown, whereas the 10 kw system has four. This choice of the number of panels assumes that the reliability of the power conditioning

system using the multimodule approach can be raised internally by use of standby power conditioning modules (e.g., Reference 8). Thus, the standby ion engine systems will not require separate power conditioning systems. When one of the operating ion engine systems fails, the power conditioning system will automatically be turned off and the failed ion engine will be switched out of the circuit. Subsequently, one of the standby ion engines will be switched in and the power conditioning will be automatically restarted. The switching and cable harness arrangement which provides for transfer of a power conditioning panel to a standby thruster is indicated in Figures 4-20 and 4-21. A detailed discussion of the cable and switching circuits employed in these system designs is given in Reference 1.

A weight itemization of both oxide and liquid cathode systems (17 and 9.6 kw) is given in Table 4-11. The specific weights of all four systems are within the design goal of 25 lb/kw. For example, the total specific weights for the 17 kw and 9.6 kw oxide cathode engine systems are 20 lb/kw and 21 lb/kw, respectively. It is important to note that the specific weight of the basic propulsion system components (i.e., thrusters, feed systems, and power conditioning and control systems) for the 17 kw oxide cathode engine system, for example, is only 13 lb/kw including thruster and power conditioning redundancy. The weights of the remaining items such as cabling, structure, and propellant tankage which make up the additional 7 lb/kw are functions of spacecraft configuration and mission profile. These latter weights are, therefore, not necessarily constant and must be evaluated for each specific application.

TABLE 4-11. WEIGHT ITEMIZATION FOR BOTH OXIDE
AND LIQUID-Hg CATHODE SYSTEMS (POUNDS)

Subsystem		17 kw		9.6 kw	
		Oxide	Liquid	Oxide	Liquid
Thrustor	Number of thrustors	8	8	6	6
	Weight of each thrustor	10	15	10	15
	Subsystem total*	80	120	60	90
Propellant storage distribution	Number of reservoirs	6	6	2	2
	Weight of each reservoir	10	10	12	12
	Reservoir total	60	60	24	24
	Feed system and plumbing valves*	24	24	18	18
	Subsystem total	84	84	42	42
Power conditioning controls	Number of panels	6	6	4	4
	Weight of each panel	20	20	18	18
	Panel total*	120	120	72	72
	Cabling-switch matrix	7	7	6	6
	Subsystem total	127	127	78	78
Structure and Mechanism	Engine mount	40	40	20	20
	Translator	10	10	6	6
	Subsystem total	50	50	26	26
Total propulsion system weight		341	381	206	236
Specific weight of propulsion system, lb/kw		20	22	21	25
* Specific weight of basic propulsion subsystems, (lb/kw)'		13	16	16	19

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1. "Solar Powered Electric Propulsion Spacecraft Study," Final Report, prepared for JPL by Hughes under Contract No. 951144, December 1965.
2. N. B. Kramer, "Ion Optics for the Electron Bombardment Engine," Hughes Research Laboratories Research Report No. 324, February 1965.
3. "Development and Test of an Ion Engine System Employing Modular Power Conditioning," Final Report, prepared for JPL by Hughes under Contract No. 951144, September 1966.
4. P. D. Reader, "Experimental Performance of a 50-cm Diameter Electron-Bombardment Ion Rocket," AIAA Preprint 64-689, September 1964.
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8. R. L. Seliger and J. H. Molitor, "Reliability in Design: Solar Electric Propulsion Systems," Annals of Reliability and Maintainability (AIAA, 1966), Volume 5, pp. 256-273.
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5. SPACECRAFT SYSTEM DESIGN

The design philosophy adopted for the solar-powered electric propulsion spacecraft general arrangements presented herein was one in which designs were formulated to be adaptable to as wide a variety of interplanetary missions as possible for a specific launch vehicle, and also, that could be scaled to be compatible with a variety of the NASA family of launch vehicles. This concept was achievable through a modular design approach of the electric propulsion system allowing the use of the same thruster and power conditioning system modular designs, in varying numbers, to accomplish a variety of missions with various launch vehicles.

During the final phase of the study three baseline conceptual spacecraft designs were of primary consideration — a spacecraft suitable for a Mars orbiter and lander mission launched by a Titan III M with a hammerhead shroud, and a universal spacecraft for a Jupiter flyby or out-of-the-ecliptic mission launched by an Atlas (SLV-3C)/Centaur with a modified OAO shroud. Power level requirements at 1 AU for the electric engine systems associated with the three baseline designs were established as 9.6 kw for the Mars mission with the Titan III M launch vehicle and 17 kw for the Jupiter flyby and out-of-the-ecliptic missions with the SLV-3C/Centaur launch vehicle.

The initial spacecraft design study phase concentrated on spacecraft designs suitable for Mars orbiter missions using the Atlas/Centaur and a Mars orbiter/lander spacecraft using the Saturn IB/Centaur launch vehicle. New performance capabilities of these spacecraft based on new trajectories which are three-dimensional in both computation and control and utilize an updated power curve (see Appendix A) result in a significantly lower power requirement than previously indicated. A weight breakdown including mission payload capability and a revised detail spacecraft design for the Saturn IB/Centaur version is presented in this section. It also should be noted that all of the Mars orbiter/lander spacecraft designs presented in this report are based on a new ground rule for launch of the lander capsule, i. e., the lander is assumed to be launched

during the Mars orbiting phase of the mission rather than just prior to retro into Mars orbit as was done for the Phase I effort. However, for purposes of reference, an improved analysis was also made of the case where a 2300 pound lander was separated prior to retro. This also is shown in this section.

DESIGN CRITERIA

The following statements define the spacecraft design criteria adhered to for all SEP spacecraft considered within the scope of this study.

Launch Vehicle-Performance

The performance capability for each of the NASA launch vehicles considered is presented in Figure 5-1. Performance data was provided by JPL.

Shroud Envelope

Figures 5-2 through 5-6 depict the shroud envelopes for the five launch vehicles considered during this and the previous phase of the study. Envelopes were provided by JPL.

Solar Array Concept

Specific spacecraft designs were developed for each of the launch vehicles considered during the study. During the Phase I study effort,

maximum power levels, as limited by shroud envelope constraints and allocated spacecraft volume, were considered for specific designs of SEP spacecraft for the Saturn IB/Centaur (50 kw) and Atlas/Centaur (10 kw). During the follow-on study phase, specific spacecraft designs for a 10 kw Titan III M SEP spacecraft and a 17 kw Atlas (SLV-3C/Centaur) SEP spacecraft were formulated. As stipulated by JPL, the design study was limited to the consideration of only the rectangular folding modular type solar cell array based on the results of the Boeing Aircraft feasibility study for a 50 kw and 10 kw solar cell array. A brief summary of the results of the Boeing study is presented below to serve as a baseline for solar array design criteria:

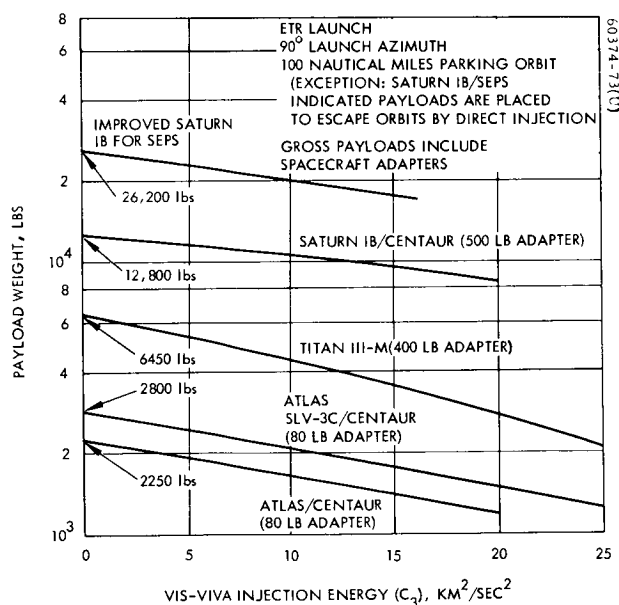


Figure 5-1. Payload Capabilities of Selected Launch Vehicles for Use with SEP Spacecraft

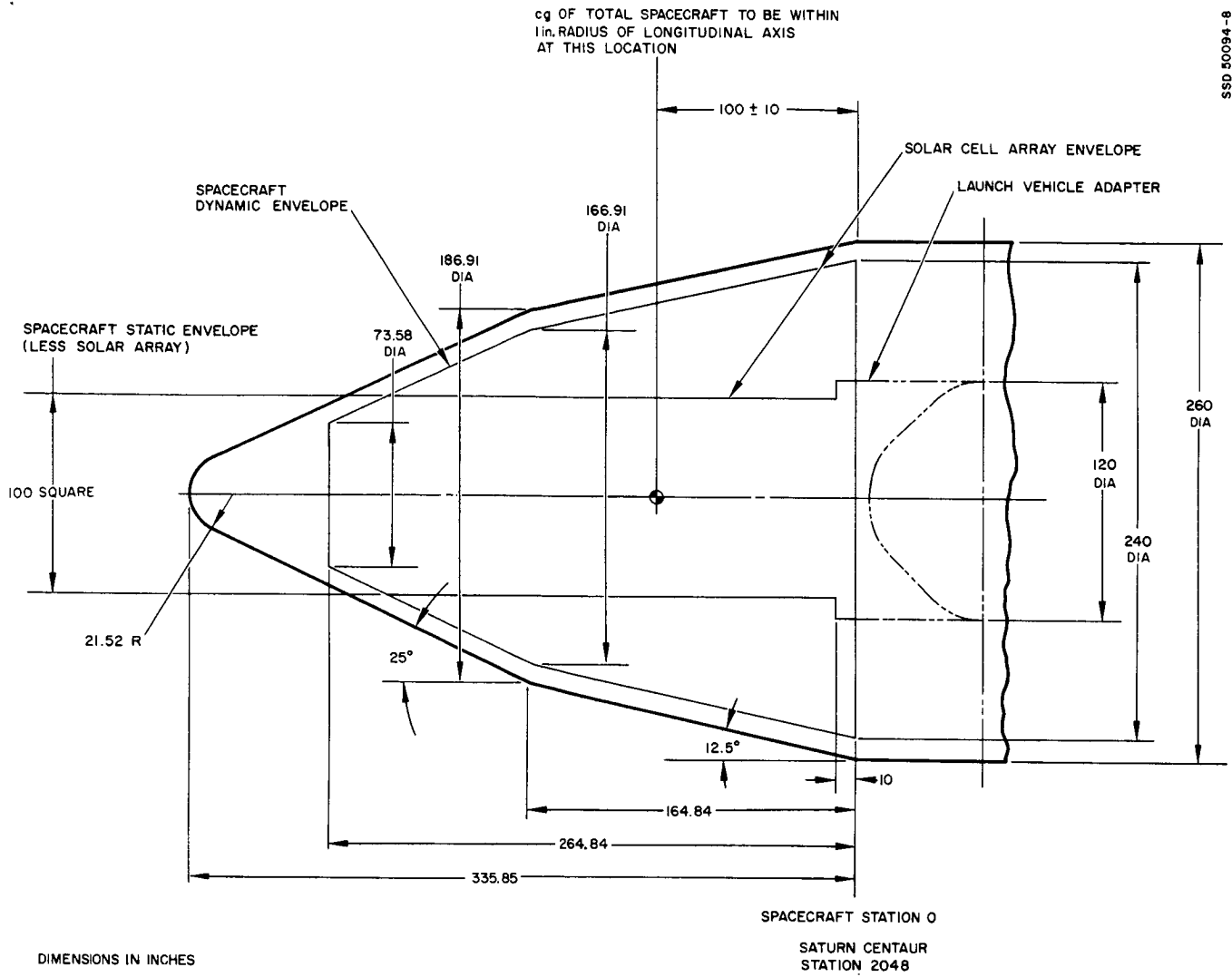


Figure 5-2. Saturn IB/Centaur Nose Fairing and Spacecraft Envelope

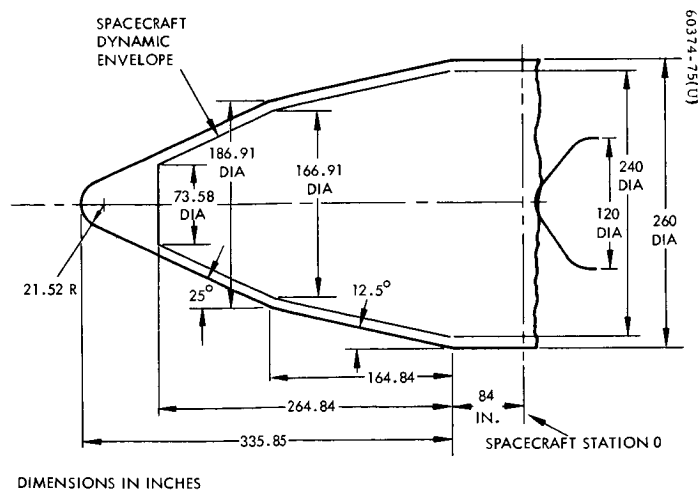


Figure 5-3. Saturn IB/Zero Stage with Extended Nose Fairing

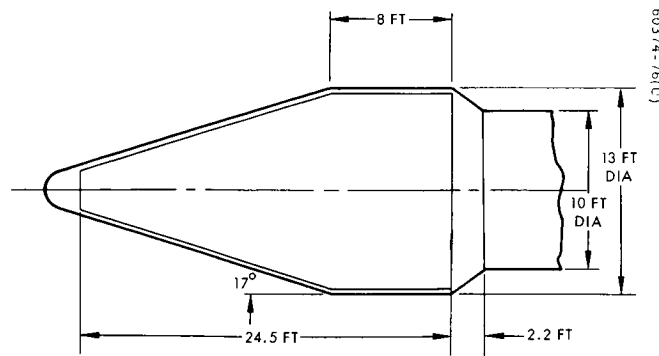


Figure 5-4. Titan III M with Hammerhead Nose Fairing

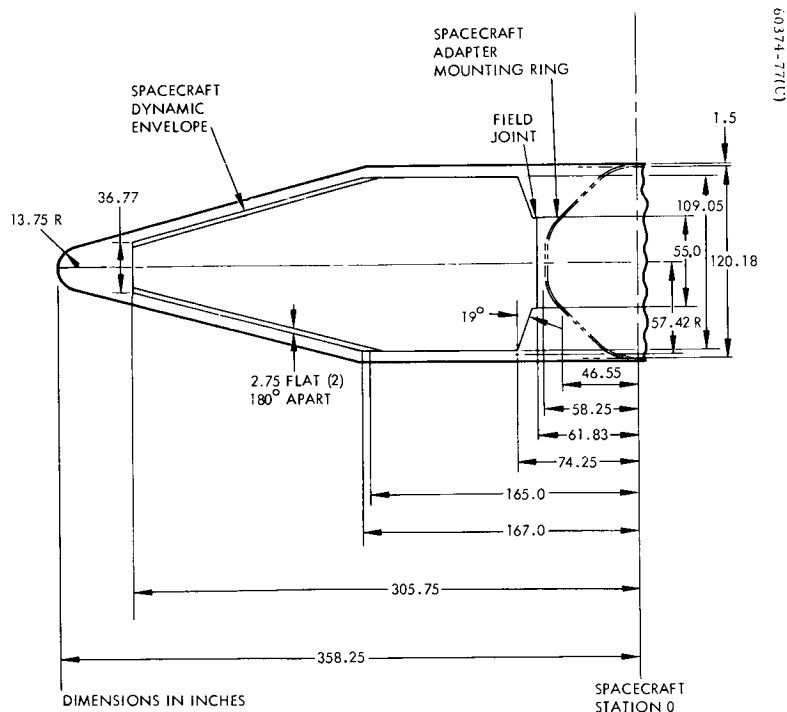


Figure 5-5. Atlas (SLV-3C)/Centaur with Modified OAO Nose Fairing

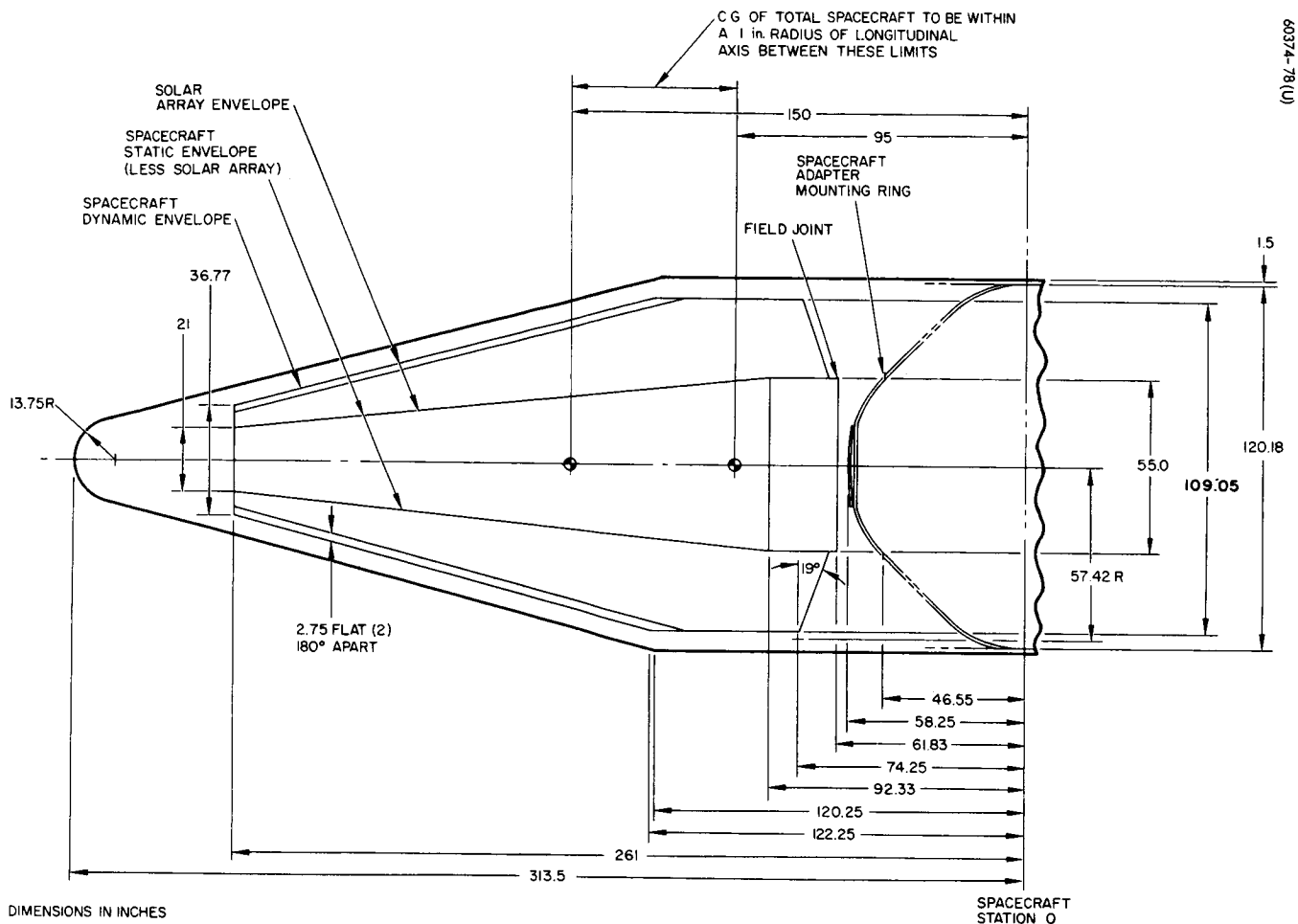


Figure 5-6. Atlas/Centaur Nose Fairing and Spacecraft Dynamic Envelope

50 kw Saturn IB/Centaur Solar Array

Gross area (deployed)	4944 square feet
Cell area	4433 square feet
89.7 percent packing efficiency	
Weight	1959.7 pounds
Power	47.7 kilowatts at 1 AU
Power-weight ratio	24.35 w/lb or 41.0 lb/kw
At 1 AU power-area ratio	10.8 w/ft ²

10 kw Atlas/Centaur Solar Array

Gross area	1004 square feet
Cell area	930 square feet
92.5 percent packing efficiency	
Weight	518.1 pounds
Power	10.18 kilowatts at 1 AU
Power-weight ratio	19.64 w/lb or 50.9 lb/kw
At 1 AU power-area ratio:	10.8 w/ft ²

Solar Array Weight Breakdown

<u>Items</u>	<u>Saturn IB/ Centaur, pounds</u>	<u>Percent of Total</u>	<u>Atlas/ Centaur, pounds</u>	<u>Percent of Total</u>
Cover glasses (0.004 inch thick)	218.16	11.1	46.33	8.9
Adhesives (RTV)	78.48	4.0	16.67	3.2
Cells (0.008 inch thick 2 x 2 cm)	464.40	23.7	98.64	19.0
Connectors	79.28	4.0	16.84	3.3
Substrate (0.003 inch thick fiberglass)	76.08	3.9	15.51	3.0
Thermal coating (0.003 inch laminar X-500)	136.40	7.0	27.79	5.4
Frame*	531.32	27.1	165.49	31.9
Mechanisms	201.24	10.3	82.36	15.9
Electric connections	174.36	8.9	48.49	9.4
Total	<u>1,959.72</u>		<u>518.12</u>	

*The space envelope that was allocated for the stowed solar array configuration of the Atlas/Centaur established tighter volume constraints than for the Saturn IB/Centaur version, thereby resulting in a less efficient frame design. A comparison of the resulting solar array specific weights shows values of 41.0 lb/kw and 50.9 lb/kw for the Saturn IB/Centaur and Atlas/Centaur, respectively. The spacecraft volume constraints for all of the configurations presented in this report are less restrictive than those formerly imposed for the Atlas/Centaur and therefore it was assumed that 50 lb/kw specific weights were attainable for all of the Mars orbiter spacecraft presented. It was further assumed that specific weights of 45 lb/kw were obtainable for all flyby spacecraft on the basis that on the flyby spacecraft the retained array is not subjected to retro loads and the jettison mechanisms are not required.

Solar Array-Shroud Envelope Parametric Study

The graphical data presented herein represents a parametric geometry study of the following five launch vehicle payload envelopes:

- 1) Saturn IB/zero stage (SEP) with 7 foot cylindrical shroud extension
- 2) Titan IIIM with hammerhead shroud
- 3) SLV-3C/Centaur with modified OAO shroud
- 4) Saturn IB/Centaur with standard shroud
- 5) Atlas/Centaur with standard shroud

The folding modular, rectangular panel solar array concept (Boeing design) was used as the basis for the layout study. The data presented permits the following information to be quickly approximated once the power level requirement for the specific spacecraft at 1 AU has been established:

- 1) Typical width of each solar array panel
- 2) Total number of panels and/or hinge lines in the array
- 3) Semispan measured (in feet) from the center of the spacecraft to outboard edge of array
- 4) Volume (in cubic feet) required for stowed solar array
- 5) Volume available for spacecraft bus proper and spacecraft subsystems
- 6) Unusable stowage volume as limited by envelope of stowed array and shroud dynamic envelope

A baseline consideration of 10 w/ft² of array area at 1 AU was used throughout, although the Boeing design study specified 10.8 w/ft².

For the Saturn IB/zero stage SEP spacecraft, the lengths of the individual solar panels in the array were arbitrarily set at 12 and 7 feet. Panels were limited to 12 feet so as to reserve sufficient payload space forward of the stowed solar array to accommodate a 16 foot diameter Apollo shaped lander capsule. Each panel was assumed to be 1.75 inches thick to account for structural frames and spars. (Note: the Boeing design for a 50 kw Saturn IB/Centaur solar array considered a thickness of 1.5 inches for each panel to accommodate the structural frame that supports the cells, cover glass, etc., plus a 0.25 inch gap between panels in the stowed configuration.)

The individual typical panel length for the Titan III M SEP spacecraft was assumed to be 8 feet so as not to extend forward beyond the cylindrical portion of the hammerhead shroud. Each panel was assumed to be 1.5 inches thick (gap considered to be included in assumed panel thickness). Limiting the panels to 8 feet in length permits consideration of possibly an 11 foot diameter scientific package stowed forward of the panels.

Considering the SLV-3C/Centaur with a modified OAO shroud the panel lengths were limited to 8.5 feet with an individual panel thickness of 1.5 inches.

For the Saturn IB/Centaur with the standard shroud the parametric study was made for 7, 8, and 9 foot wide panels with individual panel lengths of 13 feet and 8 feet. Limitation of panel lengths to 13 feet permits consideration of an 11 foot diameter stowed package in the shroud space forward of the stowed solar panels. One and three-quarter inch panels were assumed.

The data presented for the Atlas/Centaur considers that the solar panels are of maximum length, limited only by the shroud conical envelope forward of the cylindrical section. Panel thicknesses, as for the SLV-3C/Centaur with modified OAO shroud, were assumed to be 1.5 inches thick.

Figure 5-7 depicts the typical spacecraft stowed configuration considered for the parametric study. Figure 5-8 shows the data graphically relating power (kilowatts) level to typical panel width (in feet) and hence panel width to total number of panels and semispan of the deployed solar array for that specific power level. Figure 5-9 indicates the stowage volume of the solar array, the available spacecraft bus volume, and total shroud volume available for specific typical panel widths.

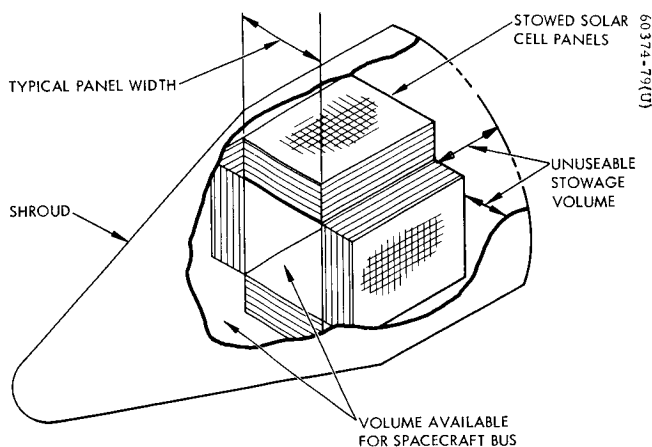


Figure 5-7. SEP Spacecraft Typical Stowed Configuration

An example of how the graphs may be used is given below. Suppose the approximate values of the parameters discussed are quickly required for a 17 kw SEP spacecraft using an Atlas (SLV-3C)/Centaur. Enter graph Figure 5-8 on the horizontal scale at the 17 kw point and move vertically to the power curve. The power curve intersects 17 kw at two places, a 2 foot and 5 foot typical panel width. Taking 5 feet for this example and moving horizontally to the two curves marked SLV-3C/Centaur for total number of panels and semispan yields a total of 40 panels and a semispan of 38 feet.

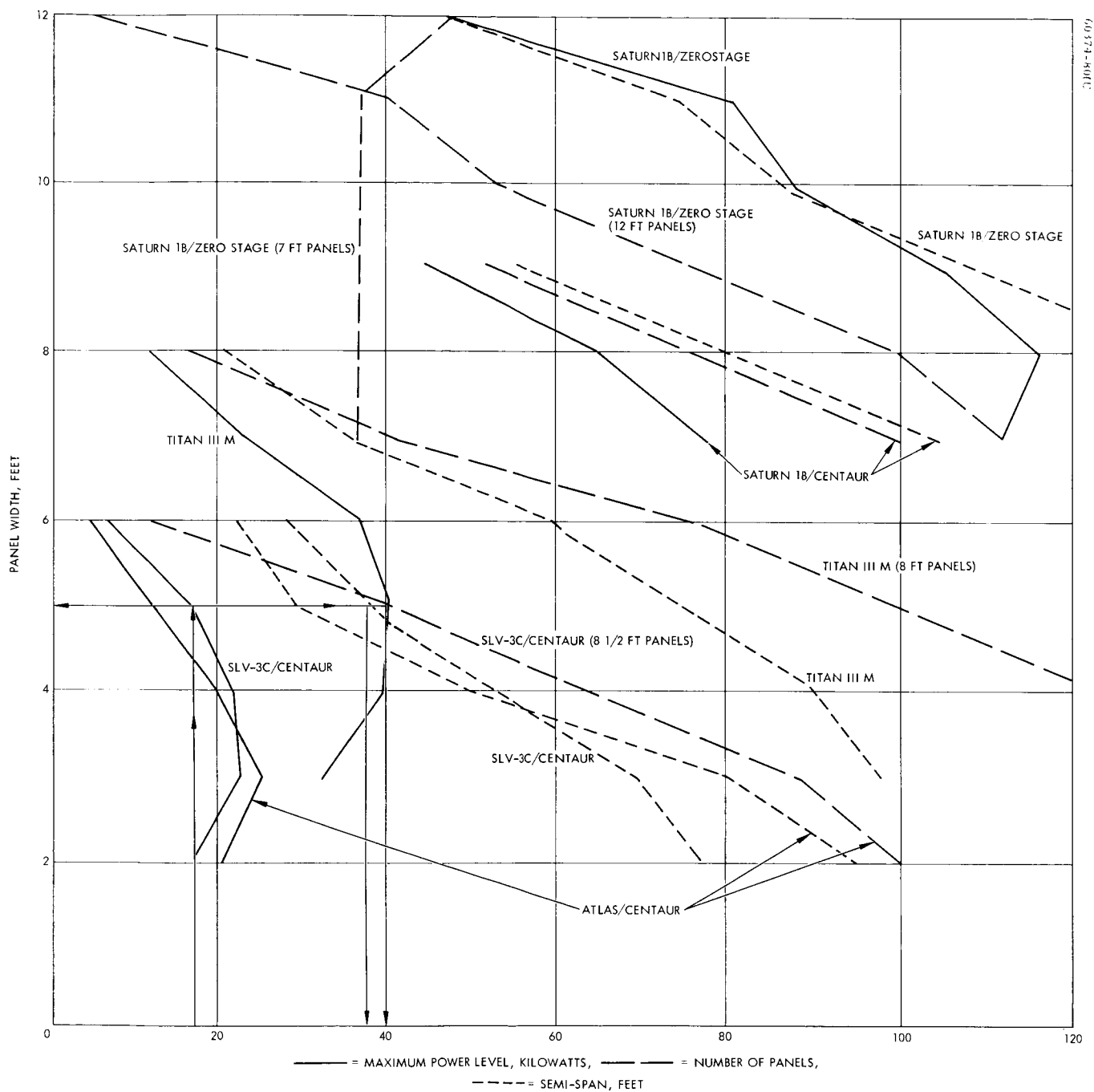


Figure 5-8. Typical Panel Width versus Power, Number of Panels and Semispan

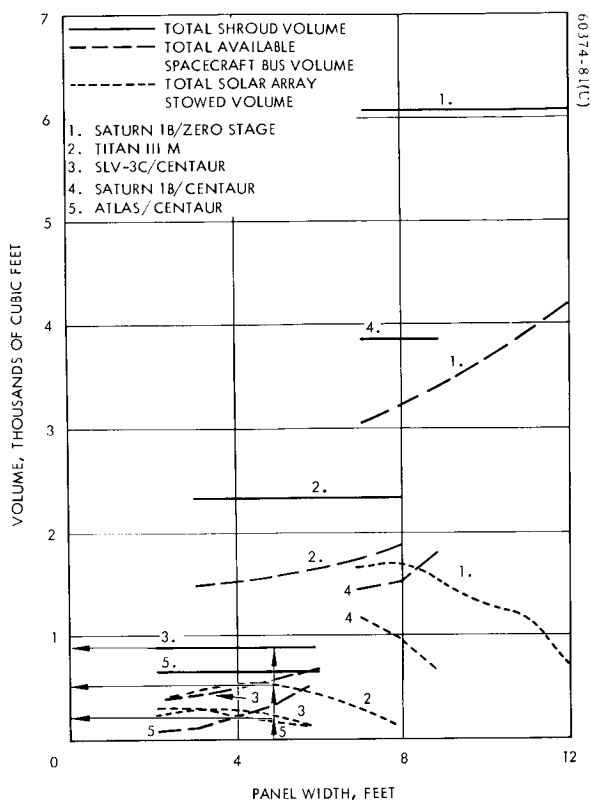


Figure 5-9. Payload Volume Allocations, SEP Spacecraft

The following volume allocations can be taken from Figure 5-9 for 5 foot wide panels:

Stowed solar array volume 220 cubic feet

Spacecraft bus volume 550 cubic feet

Total shroud volume 880 cubic feet

Electric Engine System

The following design ground rules apply to all SEP spacecraft:

- 1) The electric engine thruster array is to be "body-mounted" to the spacecraft bus and will not require deployment by means of linkages, booms, etc.
- 2) The thruster array is to be oriented so the thrust vector is parallel to the plane of the deployed solar panels with the capability of moving the engine array so as to rotate the exhaust beam away from solar panels plane (about 10 degrees could be accommodated without significant structural changes).
- 3) Two degrees of freedom translation of the thruster array with respect to the spacecraft bus is to be considered for attitude control and thrust vector - center of gravity alignment/disturbance torque corrections.
- 4) The engine system to be considered is to be limited to the Mercury bombardment type.
- 5) Thruster exhaust beam divergence of 30 degrees half-angle is to be considered for clearance requirements for the exhaust beam and adjacent spacecraft components.

- 6) For SEP spacecraft considered for planet orbit missions, the spent engine system, associated power conditioning equipment, propellant tankage, feed system, and the major portion of the solar array not required for orbiter power supply is to be jettisoned prior to the retro maneuver.

Selected Missions for Specific Spacecraft Design

Specific missions considered for SEP spacecraft baseline conceptual designs are as follows:

	<u>Mission</u>	<u>Launch Vehicle</u>	<u>Power Level, kilowatts</u>
1)	Mars Orbiter and Lander	Saturn IB/Centaur	14.4
2)	Mars Orbiter and Lander	Titan III M	9.6
3)	Jupiter Flyby	Atlas (SLV-3C)/ Centaur	17.0
4)	Out-of-Ecliptic	Atlas (SLV-3C)/ Centaur	17.0

Typical Mission Sequence of Events

- 1) The SEP spacecraft is to be boosted to the appropriate C₃ energy level by the chemical launch vehicle, the shroud jettisoned, and the spacecraft is then separated from the expended launch vehicle motor.
- 2) Deployment of low-gain antennas.
- 3) Deployment of solar cell array.
- 4) Initial solar acquisition and orientation of spacecraft for helio-centric transfer phase of mission.
- 5) Deployment of high-gain antenna and science platform may occur after proper solar panel deployment and operation. Electric engine startup initiated.
- 6) Transit phase: continuous guidance and control communications.
- 7) Approach phase: jettison of subsystems that are not required for orbiter operation.
- 8) Attitude control maneuver and retro fire into desired planet orbit.

- 9) Retro system separation, in-orbit attitude control maneuvers, and lander separation.
- 10) Continual orbital and lander data transmission to earth throughout orbiter life.

GENERAL ARRANGEMENTS

Mars Mission Spacecraft

Titan IIIM Version (Mars Orbiter and Lander)

A conceptual spacecraft design was developed for a 9.6 kw power level Mars orbiter and lander mission utilizing the hammerhead shroud configuration as stipulated by JPL. The configuration is depicted in Figures 5-10 and 5-11. The solar panels are deployed into a plane normal to the spacecraft longitudinal axis into the four quadrants at the base of the spacecraft bus. A total of 20 equal size rectangular subpanels (8 feet by 7 feet) are deployed to provide a gross area of 1120 square feet. To provide adequate thruster exhaust beam clearance with the solar array, the outboard auxiliary subpanels have been hinged to deploy asymmetrically to eliminate impingement of the exhaust flux on the panels. The main hinge attachment of the solar array to the spacecraft is at the base of the spacecraft bus, just forward of the spacecraft launch vehicle separation joint. The spacecraft longitudinal axis is along the sun-probe line with the solar cells placed on the surface away from the spacecraft bus. In this manner, the entire spacecraft bus remains in the shade thereby relaxing thermal control requirements. The solar array subpanels are purposely limited to 8 feet in length so as not to extend forward (in the stowed configuration) beyond the cylindrical portion of the hammerhead shroud. By preserving shroud stowage volume forward of cylindrical section, various lander shapes with maximum diameters up to approximately 11 feet may be accommodated.

In the four-quadrant solar array configuration, the panels are stowed in a box-like fashion around the cylindrical spacecraft bus limiting the spacecraft bus to a 7 foot diameter cross section. The spacecraft cylindrical structure is locally squared off to accommodate the electric engine array. Having the engine array buried within the spacecraft would require a large structural cutout for passage of the exhaust beam with the associated structural reinforcement weight penalty. Placement of the engine array external to the basic spacecraft frame and shell also permits a greater freedom for thruster array translation motion and thrust vector angular motions, which may be desirable on some missions.

Power conditioning modules are mounted to the spacecraft bus on each side of the engine system and are in the shade cast by the primary solar panels. The high gain antenna is stowed within the base of the spacecraft bus and gimbal mounted to satisfy clock and cone angle requirements for earth communications. A 7 foot diameter planar array antenna is indicated,

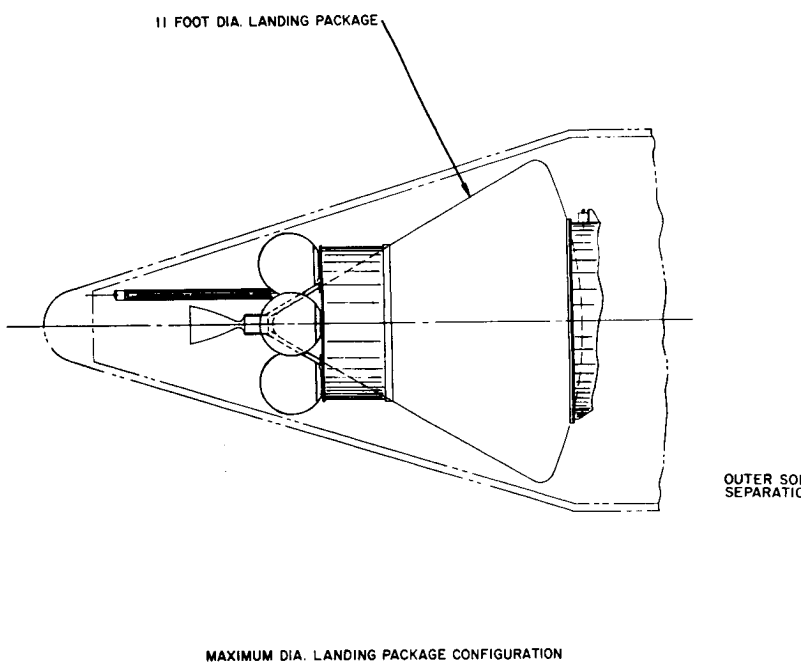
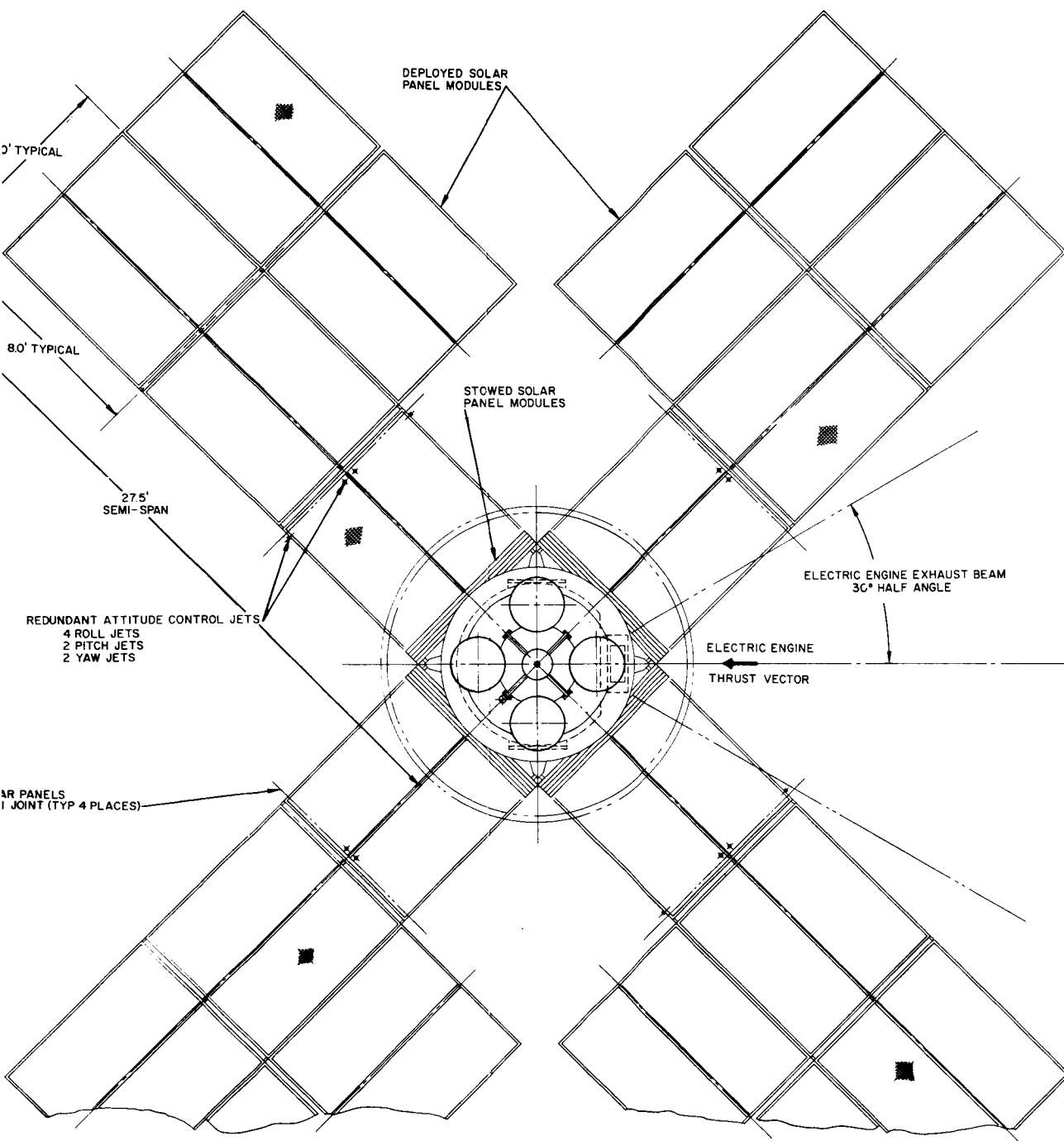
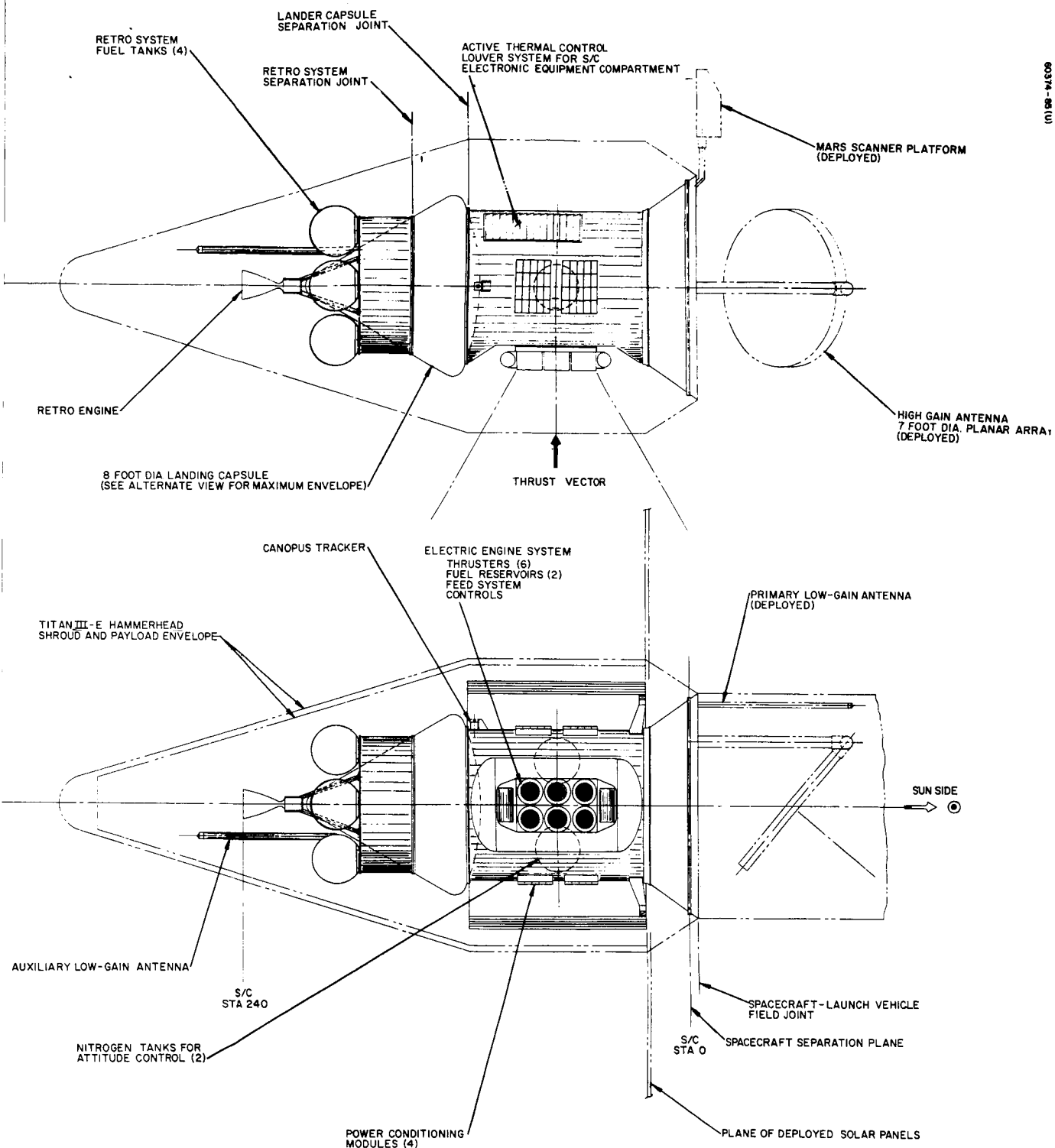


Figure 5-10. Titan III M SEP Spacecraft Gen
(10 kw power level)



General Arrangement

514-1



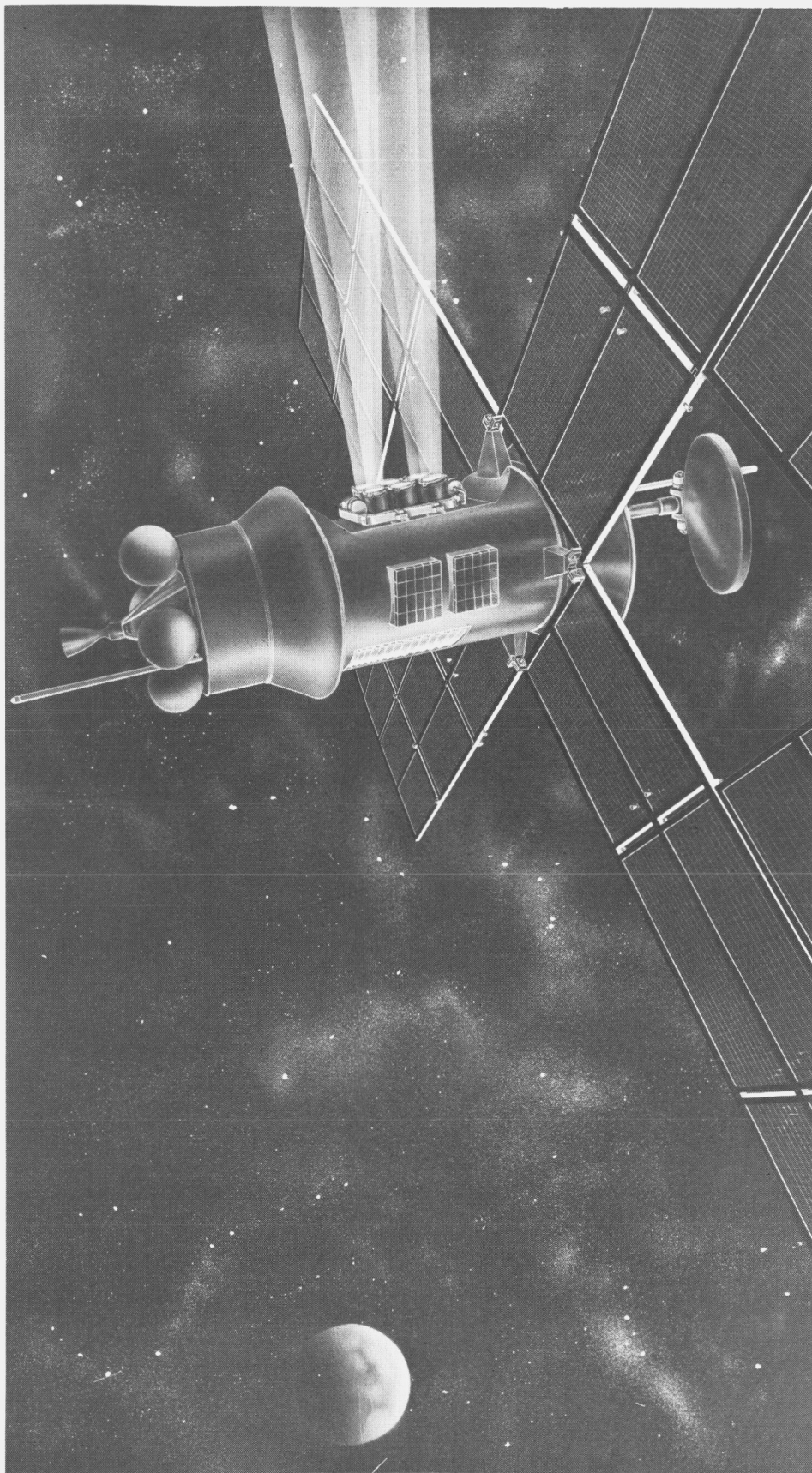


Figure 5-11. Titan III M SEP Spacecraft in Transit to Mars

although a 1 foot increase in diameter may be incorporated within the constraints imposed by spacecraft-launch vehicle support frustum interface geometry.

Major subsystems are housed within the spacecraft bus frame such as the nitrogen gas tanks for attitude control, bus or bodyfixed scientific payload, communication electronics, and guidance and control equipment. Temperature control of the spacecraft is maintained by two sensor actuated louver systems.

The lander capsule is attached to the forwardmost section of the spacecraft bus and is to be retro maneuvered into orbit with the spacecraft bus. The retro rocket and tankage are mounted such that the thrust vector is directed along the spacecraft longitudinal axis. The four solar panels beyond the outboard hinge of the primary panels are separated and ejected radially away from the spacecraft prior to the retro maneuver.

The planet scanning science equipment is mounted to deploy into the space between two adjacent solar panel assemblies on the opposite side of the thruster array to obtain maximum unobstructed viewing. Two low-gain directional antennas are indicated. They are located on the spacecraft where they are not dependent upon deployment of any other subsystem prior to their normal operation. The primary low gain antenna is deployed out of the base after spacecraft-launch vehicle separation.

The attitude control nitrogen storage tanks are mounted symmetrically about the deployed spacecraft configuration center of mass so as not to contribute to cg displacement as gas is consumed throughout the mission. The attitude control thrusters are located at the outer edge of the four innermost solar panels. The retro rocket fuel and oxidizer tanks are located symmetrically about the retro thrust chamber.

A representative weight breakdown for a Titan III M SEP spacecraft is given in Table 5-1. A 5000 kilometer circular orbit is considered for retro propulsion sizing for a 1973 mission.

Saturn IB/Centaur Version (Mars Orbiter and Lander)

A conceptual spacecraft design for a Saturn IB/Centaur solar-electric powered Mars orbiter and lander is shown in Figures 5-12 and 5-13. The spacecraft design for the Saturn IB/Centaur launch vehicle was developed during Phase I of the SEP spacecraft system study for a maximum power capability of 50 kw. The spacecraft presented herein was scaled down from the basic design to a spacecraft requiring only 14.4 kw of power. The solar array configuration is a modified version of the Boeing Company 50 kw folding modular solar array designed for the maximum power version presented in the Phase I Final Report. The solar array has been modified by reducing the number of rectangular subpanel sections to provide a gross area of 1498 square feet instead of 4944 square feet. This reduction was accomplished by omitting the outboard nine subpanels on each of two legs of the array and

by omitting ten subpanels on the remaining two legs. This configuration allows the thruster array exhaust beam to have a clear corridor to space without the beam impinging on any portion of the solar panels. A total of 14 rectangular subpanels of equal size are utilized in the configuration, three panels on two legs, and four panels on the remaining two legs of the deployed array.

The spacecraft bus is basically square in shape. The electric engine thruster bank is mounted externally to the spacecraft bus with the exhaust beam oriented to pass between two adjacent legs of the solar array. Power conditioning panels for the engine system are accommodated on each side of the engine cluster and mounted to the spacecraft structural panels. Eight 2.5 kw thrusters, six operating and two as standby, and six power conditioning modules are used.

The square base housing of the spacecraft bus is used to stow the high-gain planar array communication antenna which is 7 feet by 7 feet. The planet scan platform is also stowed within the spacecraft base. Both may be deployed from the base on masts equipped with positioning actuators once the spacecraft has separated from the launch vehicle. A base closure bulkhead just above the stowed scanner platform is used to support spacecraft electronics and orbiter body-fixed scientific experiments. The entire bus is enclosed and is thermally controlled by an active louver system on the opposite side of the electric engine array. Two nitrogen storage tanks for the attitude control system are mounted within the spacecraft bus. The attitude control thrusters for three axis control are mounted near the outboard edge of the four primary solar panels. All but the four primary solar panels are jettisoned prior to the orbiter retro maneuver into Mars orbit.

In contrast to the mission profile considered during Phase I portion of the SEP study, the lander capsule is not to be separated from the bus prior to the retro maneuver. The retro system is therefore placed on a support structure forward of the lander, with the thrust vector oriented along the spacecraft longitudinal axis, or sunline. After completion of the retro maneuver, the entire retro system and its associated supporting structure is jettisoned from the spacecraft bus to allow the lander capsule to be ejected outward along the bus longitudinal axis. The expended engine system and its power conditioning system is previously separated from the spacecraft to minimize the mass to be retro fired into orbit.

A weight breakdown for the major spacecraft subsystems is shown in Table 5-2 and for comparison purposes a comparable weight breakdown for a version of this spacecraft for a Mars mission as per the Phase I study rules. (i. e., lander launched before retro into orbit) is presented in Table 5-3.

TABLE 5-1. TITAN III M SEP SPACECRAFT WEIGHT
BREAKDOWN, 1973 MARS ORBITER AND LANDER MISSION
(9.6-kw power level)

Items		Weight, pounds	
		Injected	At Approach
Mission payload		1960	1960
Science payload	1510*		
Telecommunications	180		
Power	270**		
Solar array (portion jettisoned)		400	---
Electric engine system		250	---
Thrustor system	66		
Propellant tankage	24		
Power conditioning and controls	94		
Structure and mechanisms	34		
Feed system	32		
Propellant, Hg		273	---
Guidance and control		95	90
Structure		455	405
Thermal control		40	40
Electrical harness		60	54
Retro system		1480	1480
Total		5013	4029

* Lander capsule weight is included as part of science payload weight.

** 100 pounds of power associated with the mission payload is that portion of the solar array subsystem retained with the orbiter.

Notes:

- 1) The retro system weight is based on an approach velocity of 1.14 km/sec, propellant I_{sp} = 315 seconds, and mass fraction of 0.90.
- 2) Variation of scientific equipment weight versus selected orbits is as follows:

Orbit	Science Equipment Weight, pounds
1000 km circular	1230
5000 km circular	1510
1000 x 20,000 km elliptical	2230

ACTIVE THERMAL CONTROL LOUVER: SYS

LANDER CAPSULE (8 AND 10 FOOT
APOLLO SHAPE INDICATED)

RETRO SYSTEM SUPPORT FR
(MODIFIED TO SUIT LANDER)

RETRO ROCKET SYSTEM (FOUR 40.0" SPHERIC
TANKS FOR 5000 km. CIRCULAR ORBIT CAPA
COVERED BY THERMAL JACKET.

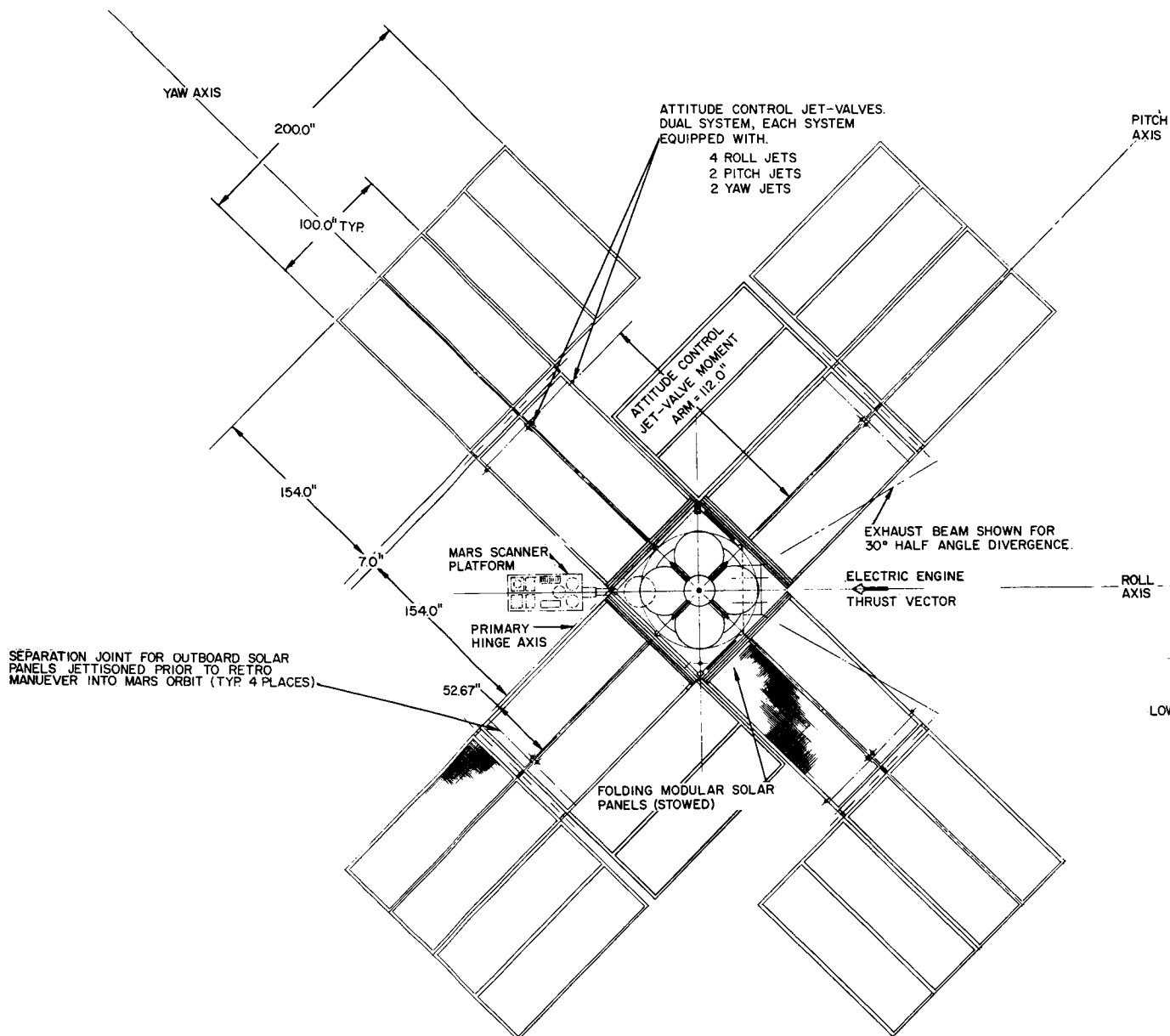
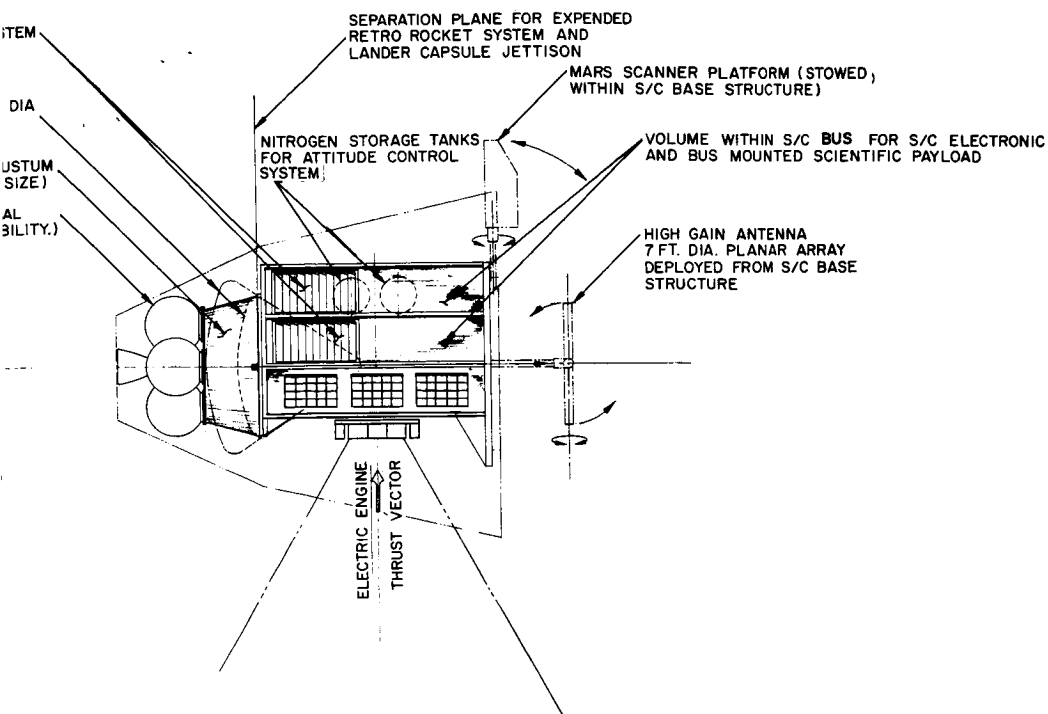
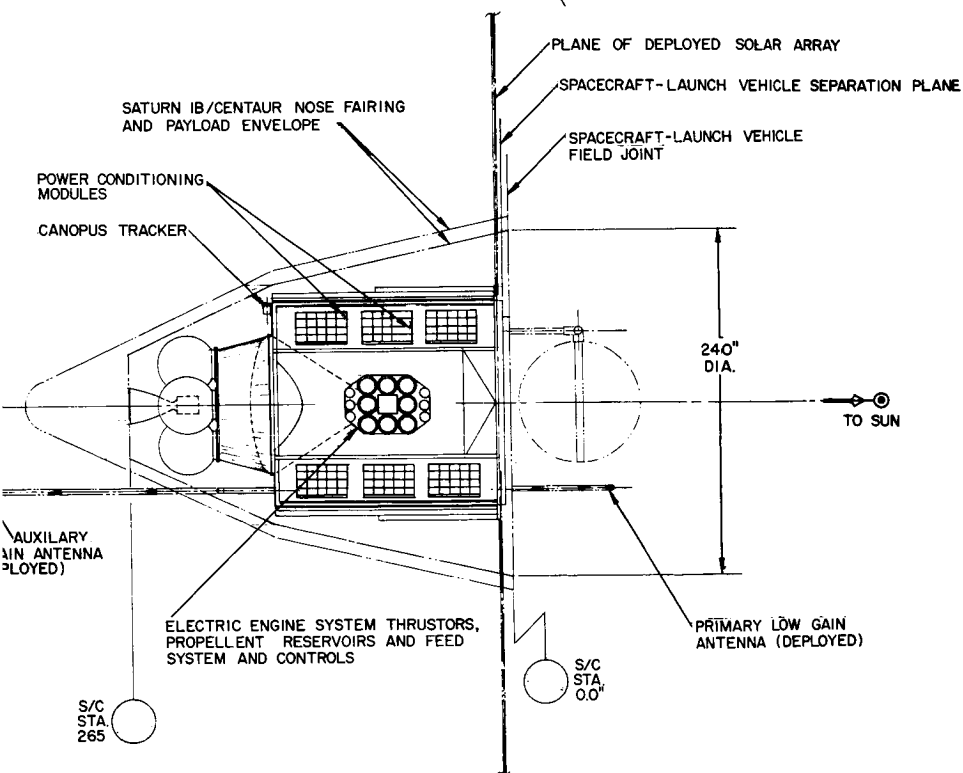


Figure 5-12. Saturn IB/Centaur SEP Spacecraft
General Arrangement
(14.4-kw power level)



60374-9810



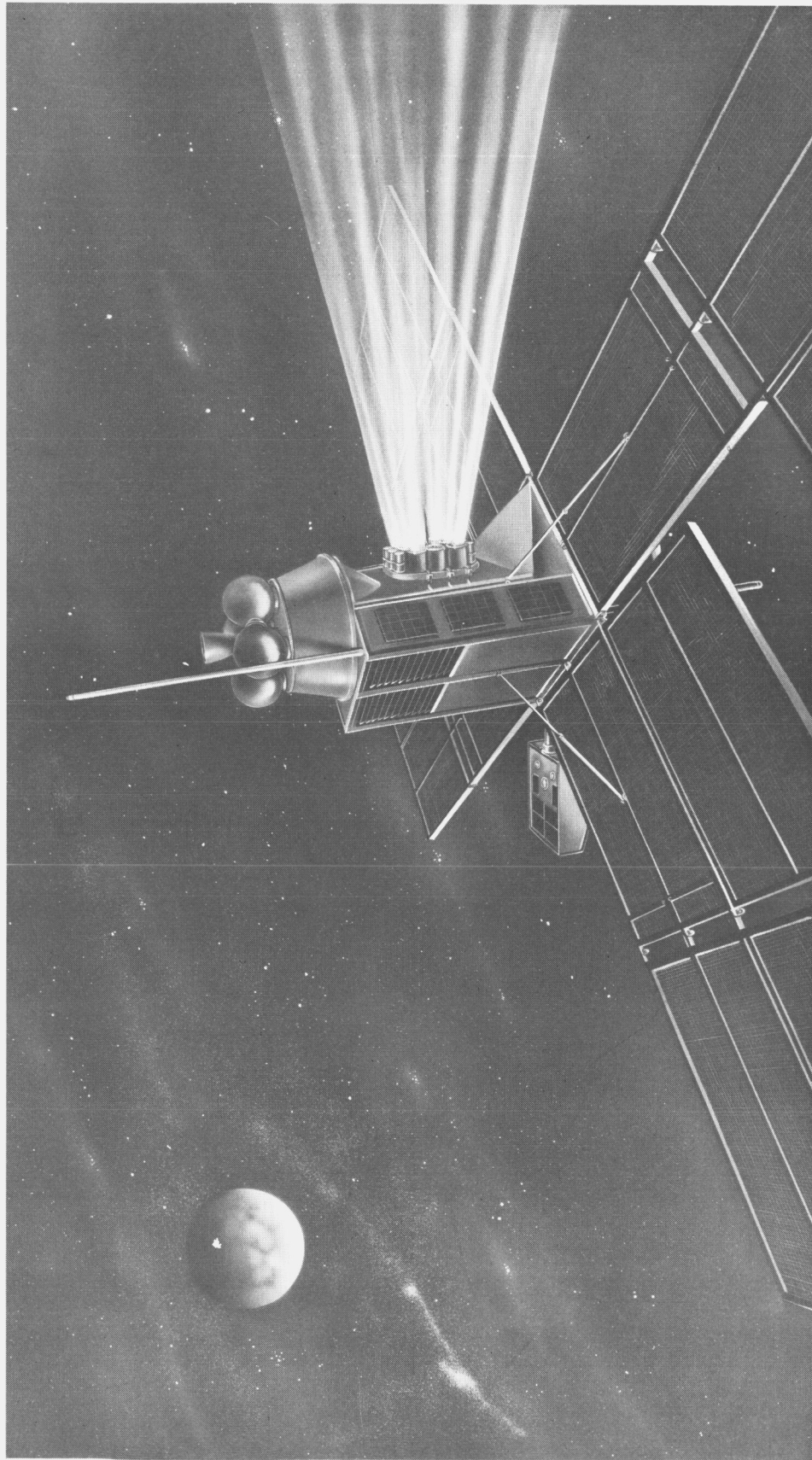


Figure 5-13. Saturn IB/Centaur SEP Spacecraft
in Transit to Mars

TABLE 5-2. SATURN IB/CENTAUR SEP SPACECRAFT WEIGHT
BREAKDOWN, 1973 MARS ORBITER AND LANDER MISSION
(14.4-kw power level)

Items	Weight, pounds	
	Injected	At Approach
Mission payload	3130	3130
Science payload		
Telecommunications	2335*	
Power	200	
Power	595**	
Solar array (portion jettisoned)	445	---
Electric engine system	360	---
Thrustor system		
Propellant tankage	95	
Power conditioning and controls	36	
Structure and mechanisms	135	
Feed system	49	
Feed system	45	
Propellant, Hg	410	---
Guidance and control	140	130
Structure	700	500
Thermal control	50	25
Electrical harness	65	58
Retro system	2220	2220
Total	7520	6063

*Lander capsule weight is included as part of science payload weight.

**275 pounds of power associated with the mission payload is that portion of the solar array subsystem retained with the orbiter.

Notes:

- 1) The retro system weight is based on an approach velocity of 1.14 km/sec, propellant I_{sp} = 315 seconds, and mass fraction of 0.90.
- 2) Variation of scientific equipment weight versus selected orbits is as follows:

Orbit	Science Equipment Weight, pounds
1000 km circular	1915
5000 km circular	2335
1000 x 20,000 km elliptical	3405

TABLE 5-3. SATURN IB/CENTAUR SEP SPACECRAFT WEIGHT
BREAKDOWN, 1973 MARS ORBITER AND LANDER MISSION
LANDER CAPSULE EJECTED PRIOR TO ORBIT
(14.4-kw power level, 5000-km circular orbit)

Items	Weight, pounds	
	Injected	At Approach
Mission payload	1665	1665
Science payload	870	
Telecommunications	200	
Power	595**	
Lander capsule	2300	---
Solar array (portion jettisoned)	445	---
Electric engine system	360	---
Thrustor system	95	
Propellant tankage	36	
Power conditioning and controls	135	
Structure and mechanisms	49	
Feed system	45	
Propellant, Hg	410	---
Guidance and control	140	130
Structure	700	500
Thermal control	50	25
Electrical harness	65	58
Retro system	1385	1385
Total	7520	3763

** 275 pounds of power associated with the mission payload is that portion of the solar array subsystem retained with the orbiter.

Notes:

- 1) The retro system weight is based on an approach velocity of 1.14 km/sec, propellant I_{sp} = 315 seconds, and mass fraction of 0.90.
- 2) Variation of scientific equipment weight versus selected orbits is as follows:

Orbit	Science Equipment Weight, pounds (Science Payload including 2300 pounds Lander)
1000 km circular	2885
5000 km circular	3170
1000 x 20,000 km elliptical	3840

Jupiter Flyby Mission SEP Spacecraft

Atlas (SLV-3C)/Centaur Version

A conceptual design of a spacecraft for a Jupiter flyby mission is shown in Figures 5-14 and 5-15. The selected mission requires an electric engine system rated at 17 kw at 1 AU. Major design considerations dictated by the mission profile were imposed over the previously considered Mars spacecraft. Of primary concern was the large dynamic thermal range introduced by virtue of considering a swing-in type trajectory. For thermal control reasons, the spacecraft bus was placed on the shadow side of the solar array plane. The size of the high-gain antenna (6 foot diameter parabolic dish) negated the feasibility of stowage within the base of the spacecraft due to the space limitations imposed by the package envelope of the solar array and the 57 inch launch vehicle interface. In lieu of a circular parabolic dish, a highly elliptical dish was also considered to be stowed longitudinally within the spacecraft to be deployed out of the base. This design was rejected due to the articulation required for deployment, plus the large volume required for stowage.

The only space solely restricted by the launch vehicle shroud envelope void of spacecraft interface considerations exists at the top of the spacecraft. To not require that the high-gain antenna be articulately deployed using a mast and associated mast mounted pointing actuators, the main solar panel hinges and high-gain antenna were placed at the top of the spacecraft bus, thereby satisfying both thermal control and communications considerations. Placing of the power conditioning modules remotely from the solar panels significantly reduces thermal coupling. The modules are mounted at the base of the spacecraft bus. Adequate area exists in the base structure of the spacecraft to accommodate two of the six power conditioning panels with an unobstructed view to radiate to space. For spacecraft mass balance and thermal isolation considerations, the bulk of spacecraft electronics and science payload is situated in a compartment just above the power conditioning equipment. The equipment in this compartment is thermally controlled by an independent active louver system.

Two low-gain antennas are deployed from the spacecraft bus, one to the normal sunside position and the other on the shadow side from the base of the spacecraft. The deployment of either low-gain antenna is not contingent upon deployment of any other spacecraft subsystems. A medium gain horn is mounted on top of the feed supporting tripod of the high-gain dish antenna.

The electric engine system is composed of eight thrusters, six propellant feed system reservoirs, and associated control and switching components. Placement of the thruster array is such that the exhaust beam is directed between two adjacent deployed solar panel assemblies. The engine cluster is fixed so its thrust vector is directed at 90 degrees to the sunline. To accommodate the exhaust beam divergence of the electric engine thrusters without exhaust impingement on the outboard solar panels, the auxiliary subpanels are asymmetrically hinged to provide maximum clearance

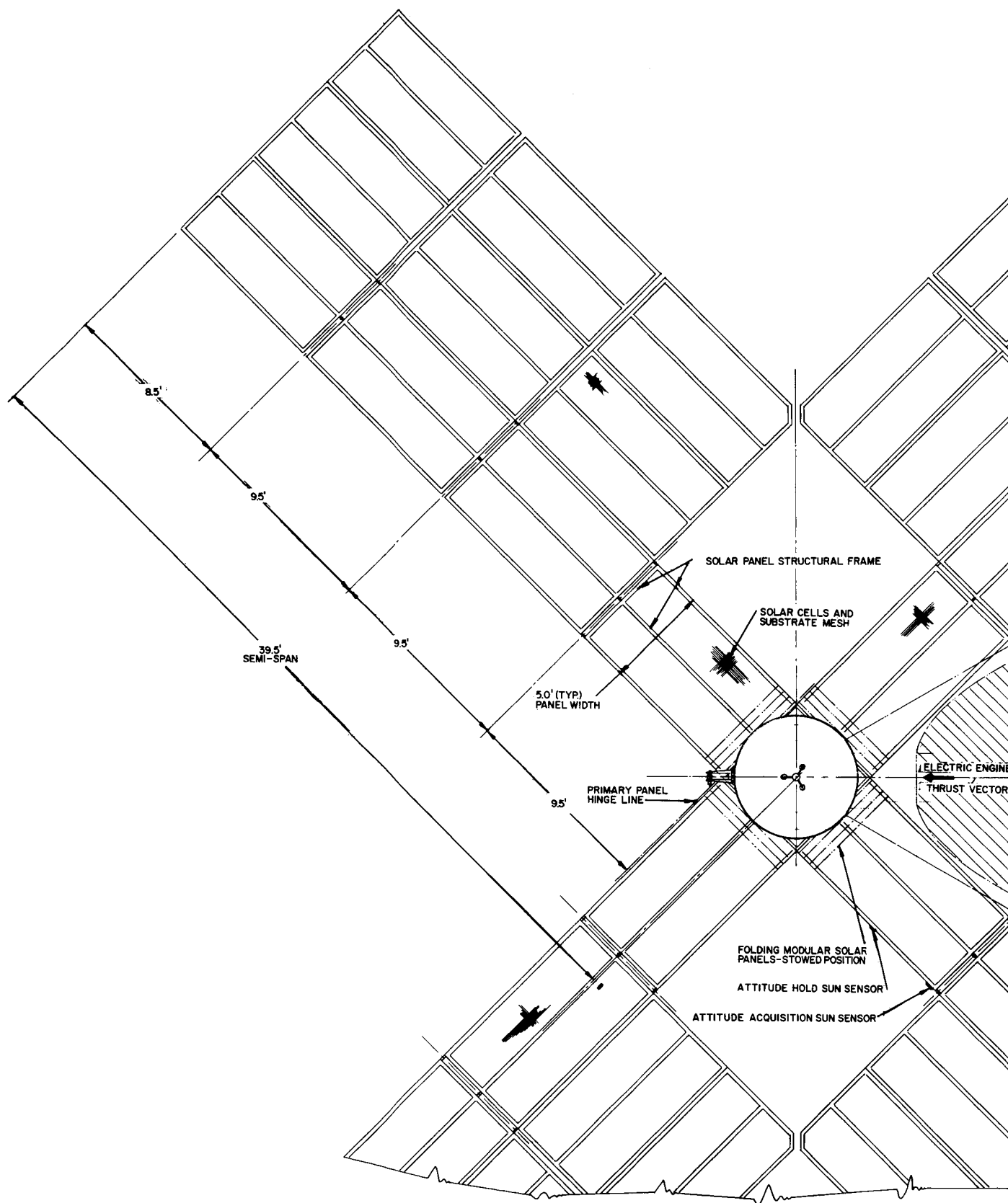
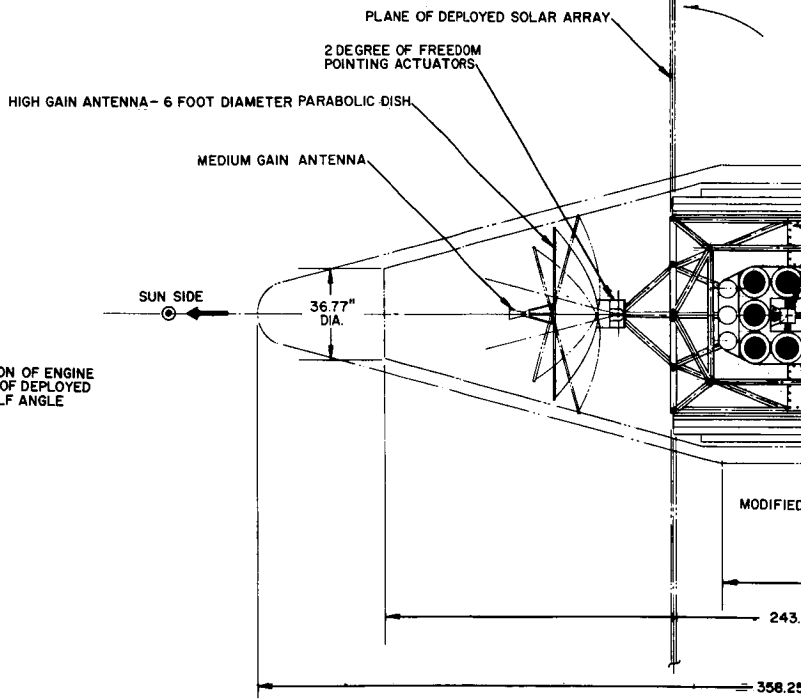
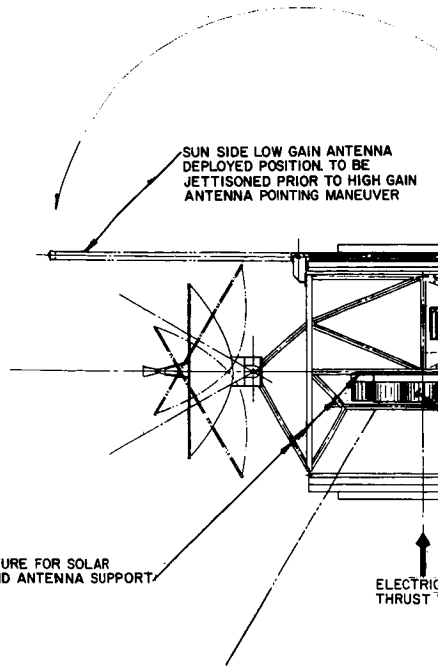
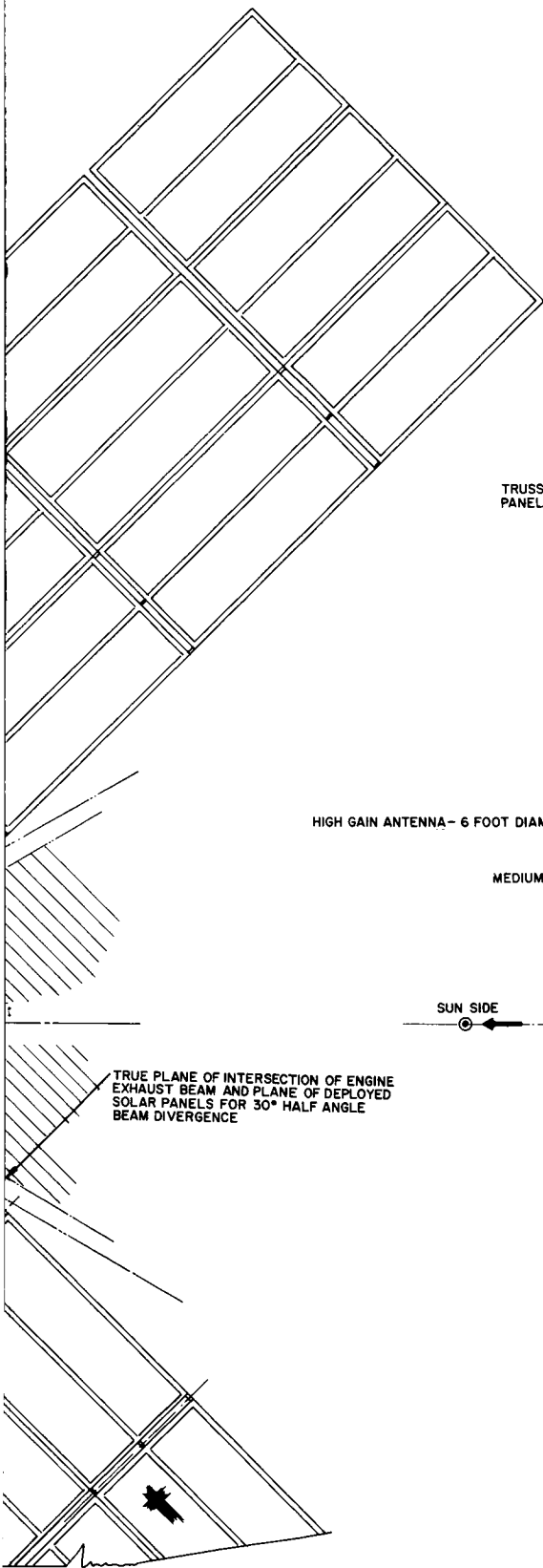
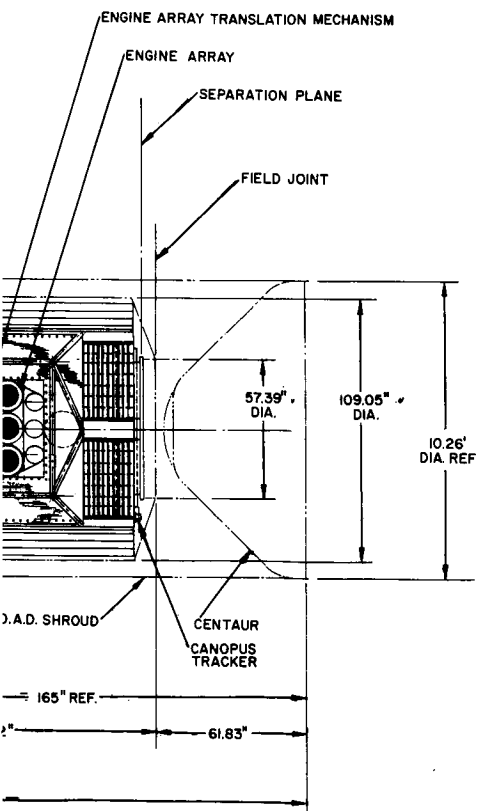
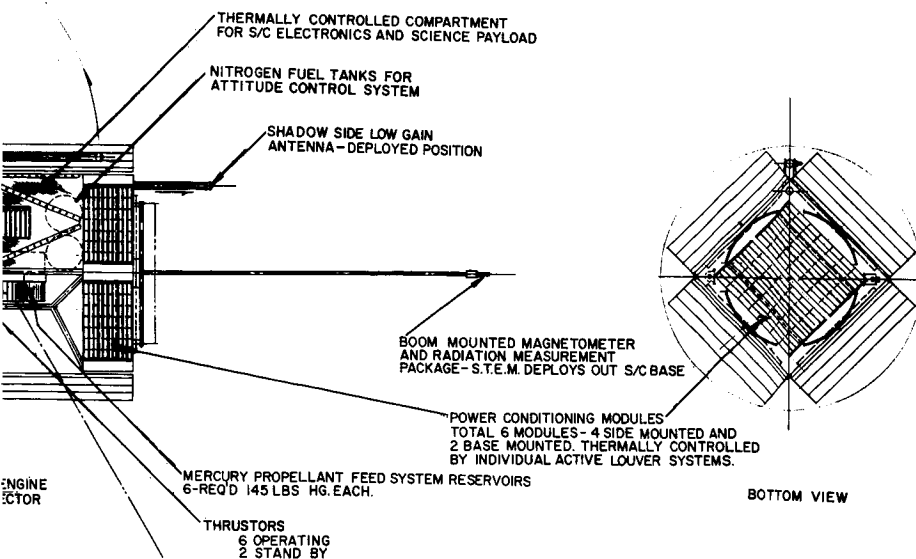


Figure 5-14. Atlas (SLV-3C)/Centaur SEP Spacecraft
General Arrangement
(17 kw power level)



TRUE PLANE OF INTERSECTION OF ENGINE
EXHAUST BEAM AND PLANE OF DEPLOYED
SOLAR PANELS FOR 30° HALF ANGLE
BEAM DIVERGENCE



5-26-2

in the thruster quadrant. To maintain symmetrical area distribution, the panels on the side opposite the engine are likewise arranged.

The major portion of the spacecraft base is enclosed for thermal control requirements. The central and upper section of the spacecraft is purposely void of equipment since the most effective location for compensating for the weight of the solar panels is at the base (bottom end) of the spacecraft. The attitude control N_2 gas tankage is mounted within the equipment enclosure while the attitude control thrusters are mounted on the outboard edge of each of the four solar panel assemblies. Flexible lines are utilized at each of the solar panel hinges to accommodate the panel deployment motions.

A truss structure extends upward to provide a square framework at the top for the main hinge attachment of the four solar panel assemblies. A light tripod supports the high-gain antenna and pointing actuators on top of the solar panel hinge frame. The truss structure in the vicinity of the engine cluster is constructed to allow longitudinal and lateral translation of the entire engine system.

The spacecraft in the stowed configuration has been designed to be compatible with the envelope provided by a modified OAO Centaur nose fairing. Without the cylindrical extension of approximately 4 feet as provided by the modified shroud, the 17 kw power level utilizing a folding modular solar cell array could not be realized. The standard shroud configuration accommodates a maximum of 10 kilowatts of solar array. A realistic maximum power level based on envelope and geometry considerations appears to be 22 kw for the modified OAO Centaur shroud.

A weight breakdown for the Jupiter flyby spacecraft is presented in Table 5-4.

Out-of-Ecliptic Probe SEP Spacecraft

Atlas (SLV-3C)/Centaur

The general arrangement of a spacecraft for an out-of-ecliptic probe is shown in Figures 5-16 and 5-17. The Atlas (SLV-3C)/Centaur with a modified OAO shroud is the launch vehicle considered for the out-of-ecliptic mission. Although the thermal considerations and communication requirements for an out-of-ecliptic probe do not pose design constraints nearly as severe as for the Jupiter flyby SEP spacecraft, the spacecraft arrangement developed for the Jupiter mission is well suited to serve as an out-of-ecliptic probe with only minor modifications. For communications, only the low-gain antennas are required. The sun side low-gain antenna need not be deployed as on the Jupiter spacecraft but may be permanently mounted on top of the spacecraft simplifying the antenna mount design. The overall spacecraft in essence is identical to the Jupiter flyby spacecraft and conceptually would be feasible for use on either mission. For an out-of-ecliptic spacecraft, the space previously utilized by the high-gain antenna for a Jupiter mission may be used to accommodate approximately 25 square feet of fixed solar cell panel.

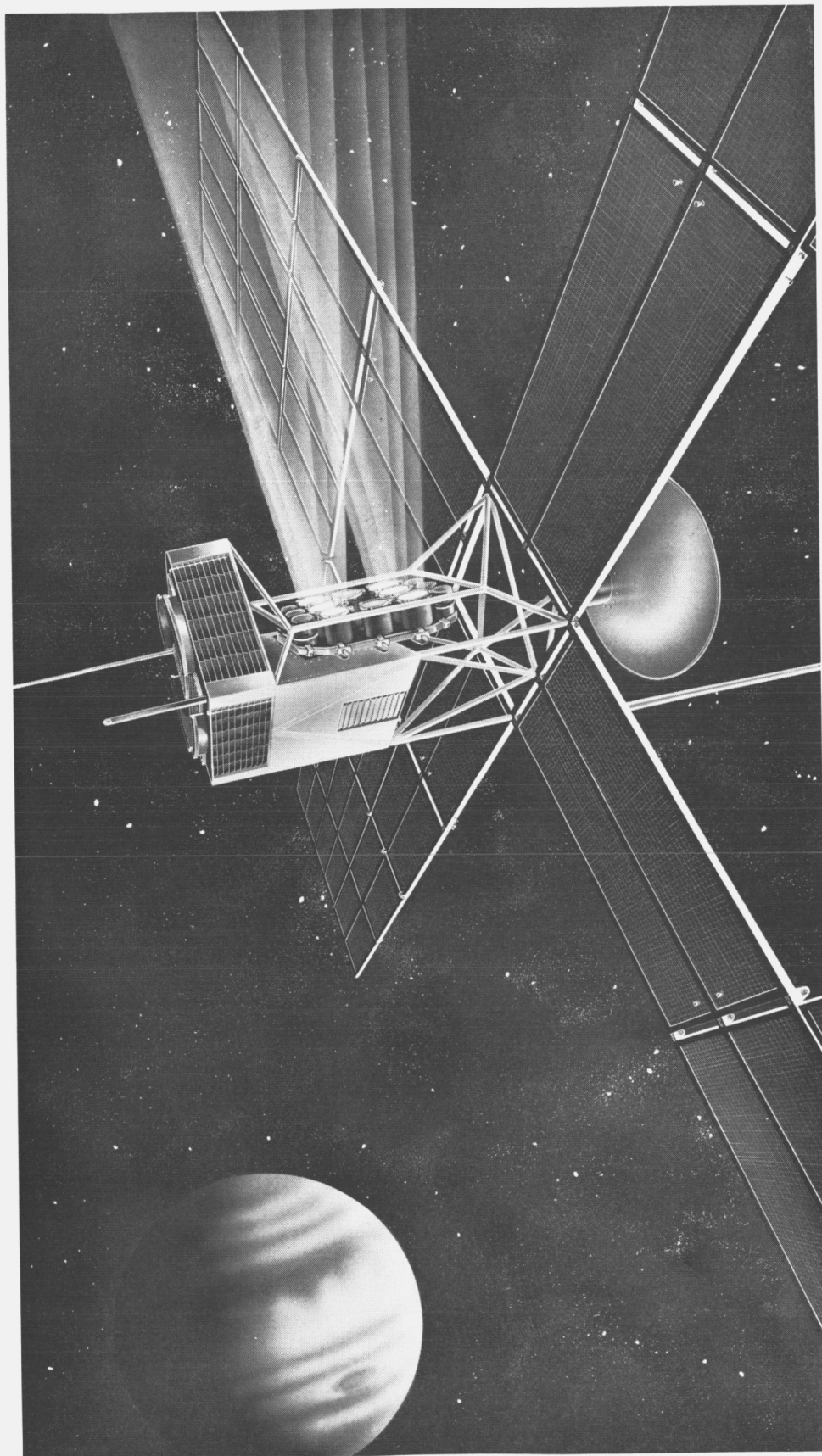


Figure 5-15. Atlas (SLV-3C)/Centaur SEP Spacecraft
in Transit to Jupiter

TABLE 5-4. ATLAS (SLV-3C)/CENTAUR SEP SPACECRAFT
WEIGHT BREAKDOWN, 1973 JUPITER FLYBY MISSION
(17.0 kilowatts power level)

<u>Items</u>	<u>Injected Weight, pounds</u>
Mission payload	238
Scientific payload	23
Telecommunications	165
Battery power subsystem	50
*Solar power subsystem	(765)
Solar array, 45 lb/kw	765
Thruster system	80
Propellant tankage	60
Power conditioning and controls	126
Structure and mechanisms	50
Feed system	24
Propellant, Hg	865
Guidance and control	107
Structure (including micrometeroid shielding)	205
Thermal control	45
Electrical harness	50
Total	2615

*Entire solar array power available for mission applications at Jupiter flyby.

Note:

$$C_3 = 1.22 \text{ (km/sec)}^2, \text{ 900 days mission time.}$$

No change in the engine system is required and a weight savings in the active thermal control for the power conditioning modules would likely result.

A weight breakdown for the out-of-ecliptic probe is presented in Table 5-5.

TABLE 5-5. ATLAS (SLV-3C)/CENTAUR SEP SPACECRAFT
WEIGHT BREAKDOWN, OUT-OF-ECLIPTIC MISSION
(17.0-kw power level)

<u>Items</u>	<u>Injected Weight, pounds</u>
Mission payload	298
Scientific payload	137
Telecommunications	111
Battery power subsystem	50
*Solar power subsystem	(765)
Solar array, 45 lb/kw	765
Electric engine system, 20 lb/kw	340
Thrustor system	80
Propellant tankage	60
Power conditioning and controls	126
Structure and mechanisms	50
Feed system	24
Propellant, Hg	865
Guidance and control	107
Thermal control	10
Electrical harness	50
Total	2615

*Entire solar array power available for mission applications in out-of-ecliptic final orbit.

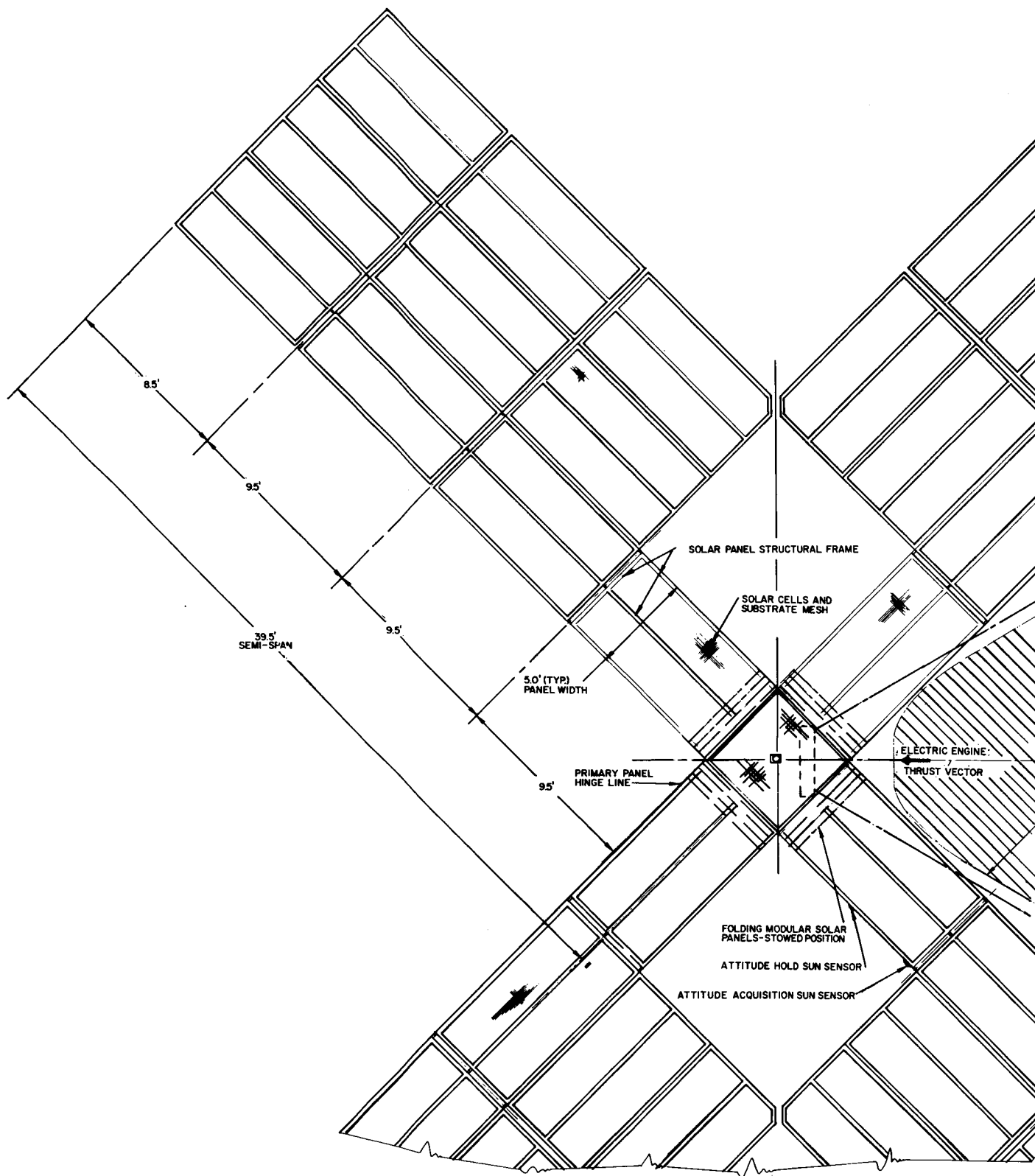
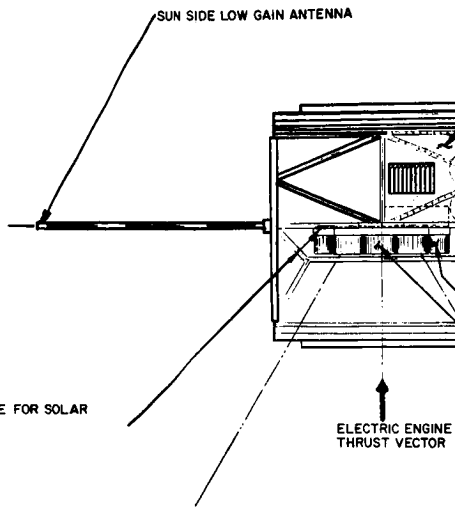
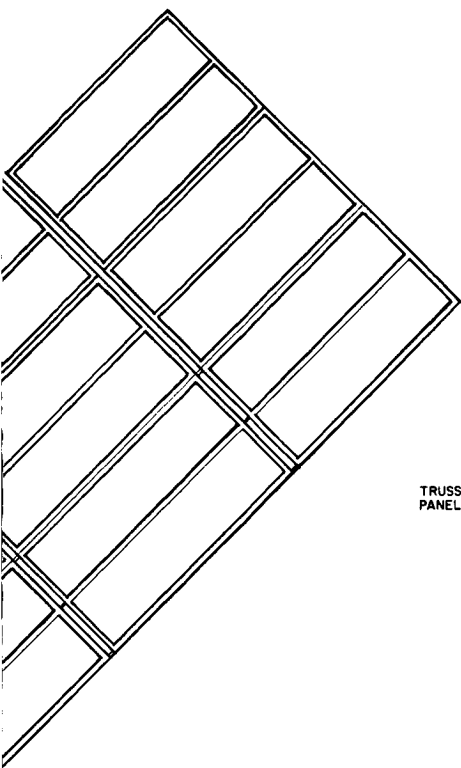
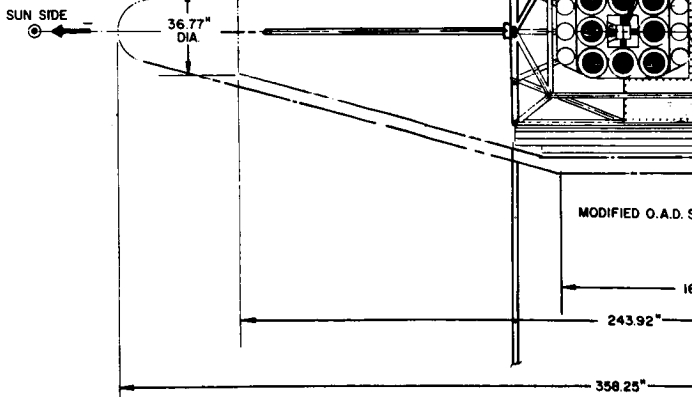


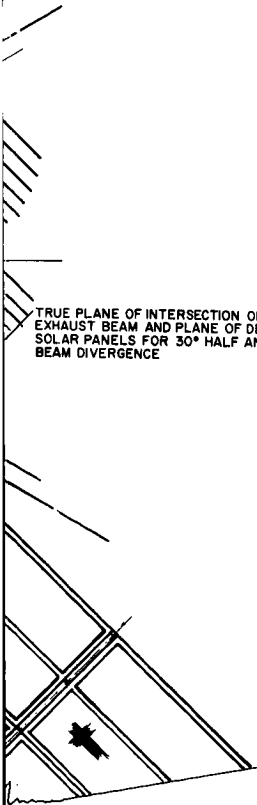
Figure 5-16. Atlas (SLV-3C)/Centaur SEP Spacecraft General Arrangement (Out-of-Ecliptic Mission, 17 kw Power Level)



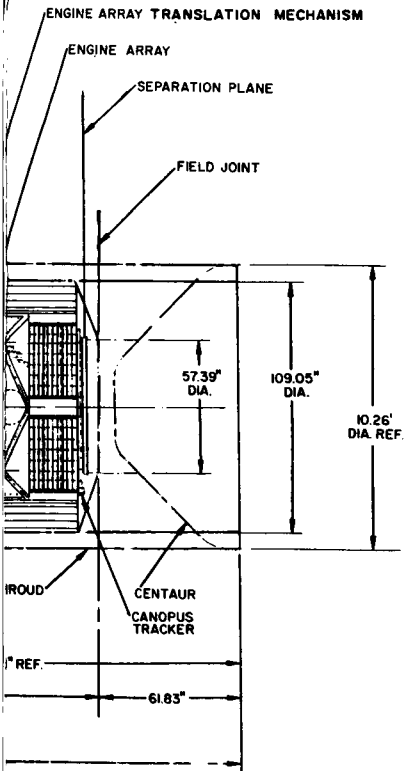
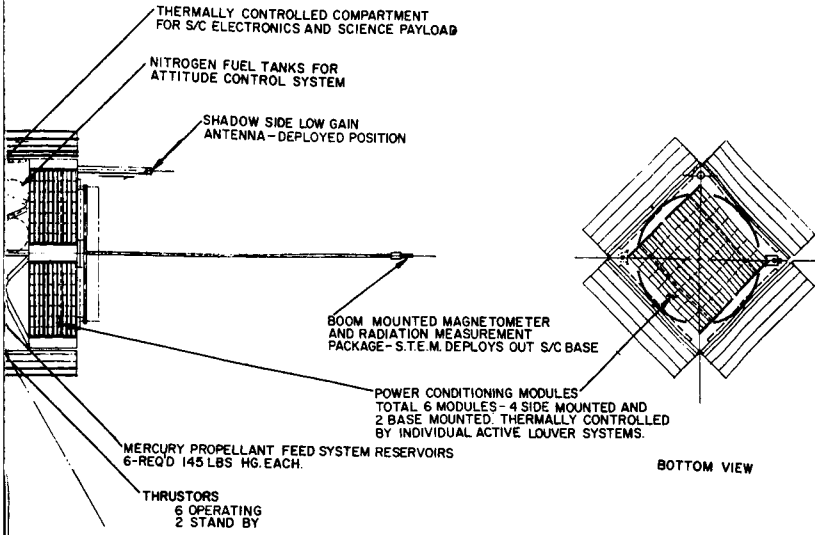
PLANE OF DEPLOYED SOLAR ARRAY



TRUE PLANE OF INTERSECTION OF ENGINE EXHAUST BEAM AND PLANE OF DEPLOYED SOLAR PANELS FOR 30° HALF ANGLE BEAM DIVERGENCE



5-32-1



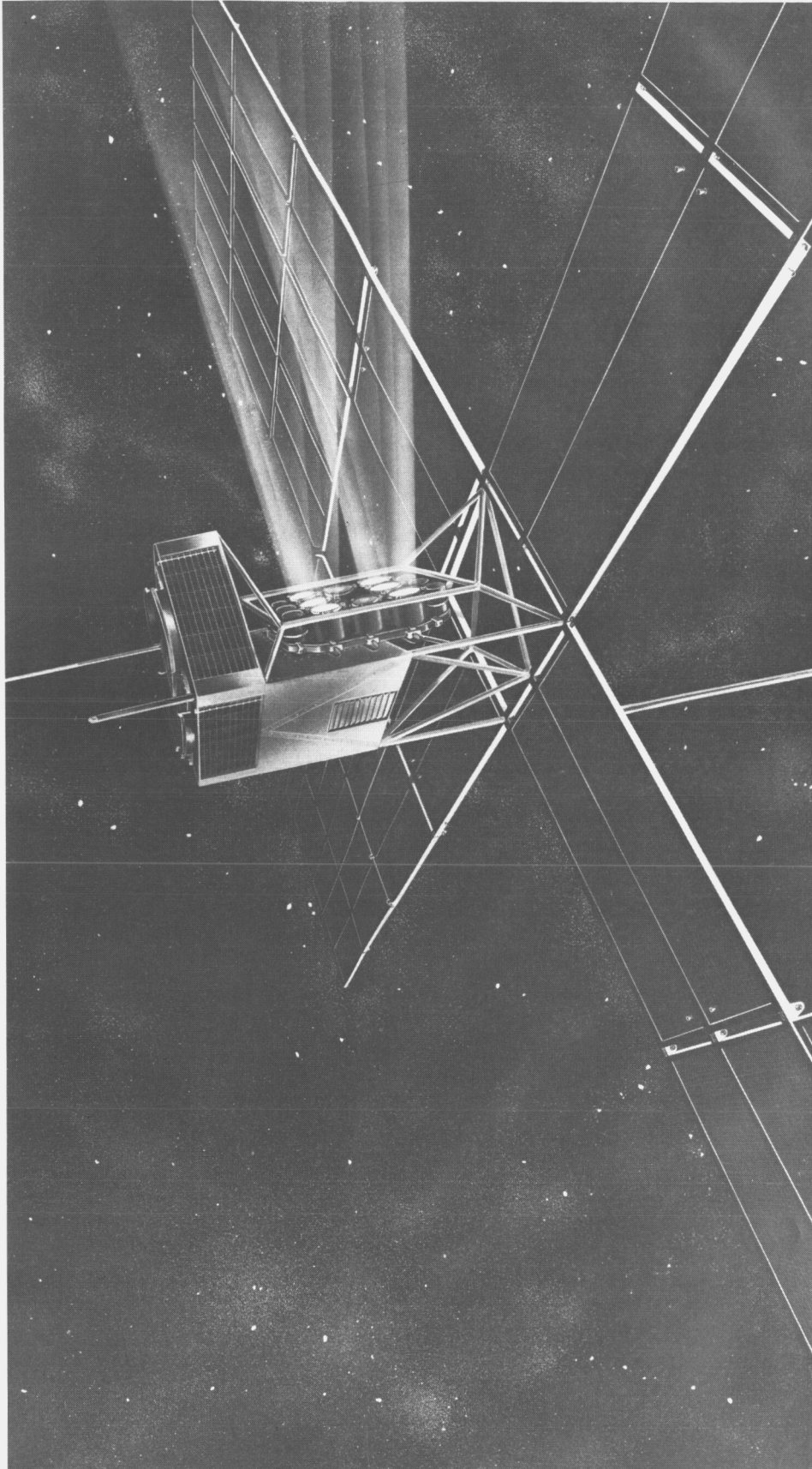


Figure 5-17. Atlas (SLV-3C)/Centaur SEP Spacecraft in Transit
During Out-of-Ecliptic Mission

SCIENTIFIC PAYLOAD CONSIDERATION

Representative scientific payloads have been devised to ascertain the desirable locations of the various instrumentation requiring movable platforms. These payloads are used only to illustrate the general problems of location, orientation, and field of view in the design of the spacecraft. Three specific scientific objectives have been considered in this analysis:

- 1) Determination of characteristics of Martian space by orbiters and landers.
- 2) Determination of characteristics of Jovian space by a flyby and orbiter.
- 3) Determination of properties of solar space above ecliptic plane.

However, consideration was also given to monitoring the space in between the planets — the solar atmosphere. The approach to selection of representative payloads is to obtain "reasonable" scientific coverage of the area of interest within certain payload weight boundaries established by a parameterized astrodynamic study.

One further step was to establish the minimum payload to be of a worthwhile scientific value below which the gain would not be sufficient to justify the mission.

The scientific payloads of any of the landers are not considered in this study. Only the total weight of the landers is estimated as a specific subsystem of the main scientific payload for the purpose of evaluating the integration problems.

Scientific Payload Constraints

The basic constraint in establishing a scientific payload is the allowable weight with which to work. For this study the weight constraints are as follows:

Planetary payloads:

Mars	Orbiter	720 to 910 kg
	Lander	42 kg
Jupiter	Flyby	27 to 46 kg
	Orbiter	15 kg
Out-of-the-ecliptic		30 to 50 kg
(at 25 degrees maximum orbit inclination)		

Any number of experiments can be considered in each of these weight categories depending upon other scientific value of the experiment and the degree of accuracy required. One controlling factor that would tend to place an upper limit on the number of experiments is the data handling capacity. This is a function of the transmitted power and the communication distance. The details of the telecommunications system are discussed in another section.

The number of scientific experiments must be evaluated against the data transmission capabilities including command signals, coding to minimize probable error, consideration of channel capacity, and power available. The amount of data available will be a function of the type of orbit obtained. The greatest load of data comes from the television cameras.

There are other constraints on location of the experiments to minimize the effects of the spacecraft and maximize the field of view. These are discussed under the section on specific instruments.

Planetary Experiments

The scientific objectives of a planetary flyby, orbiter, or a lander are the same except for the extent and accuracy of measurement. Some of these are:

- 1) Measure presence of any planetary magnetic fields
- 2) Determine amount and composition of any atmosphere that is present
- 3) Determine presence and intensity of any trapped radiation
- 4) Determine presence of cosmic dust
- 5) Observe surface features as to contours, temperature, and color, as well as natural planetary satellites
- 6) Determine properties of surface deduced from indirect and/or direct measurements
- 7) Life detection by techniques of Items 5 and 6.

The various representative payloads that meet these objectives for planetary exploration and interplanetary monitoring in the ecliptic plane are shown in Tables 5-6 through 5-8, covering a lander for the planet Mars, and a flyby and orbiter for the planet Jupiter.

Scientific Payload for Mars Lander

The allowable payload for the Mars orbiter mission permits the use of a Surveyor-type spacecraft for soft landing on the Martian surface. While the main emphasis is on the scientific payload landable on the surface, it is important to consider an orbiting payload to back up or enhance the lander payload.

TABLE 5-6. SCIENTIFIC PAYLOAD FOR MARS LANDER

Experiment	Instrument	Range	Weight Kg	Power, ergs sec ⁻¹ x 10 ⁻⁷	Sample Rate	Bit Rate	Total Bits
Environment							
Magnetic field	He magnetometer	-400 to 10 ³ T	3	5	10 per minute	24 bps	2.88 x 10 ⁴ (2 hours)
Charged particle ionization rate	Neher chamber		1.1	0.5			
Cosmic ray plus trapped radiation	GM tube array	1 to 10 ⁴ p. cm ⁻² rec ⁻¹	1.2	1	10 per minute before touchdown 1 hr ⁻¹ after touchdown	10 bps	
Electron flux (ionosphere)	Electron counter	E > 40 kev 10 ² to 10 ⁴ el. cm ⁻² rec ⁻¹ kev ⁻¹	2.4	1.5			
Surface							
Landing site observations	Approach TV camera		9	15	10 pictures	2.4 x 10 ⁵ bits/picture	2.4 x 10 ⁶
Pictures after touchdown	Survey TV camera	Surveyor	9	15		2.4 x 10 ⁵ +02.2 x 10 ⁶ bits/picture	
Surface composition	Gas chromatograph		3	3	First 30 minutes	600 bits per channel	1800
Hardness	Strain gages on legs	Three channels	0.1	0.02			
Atmosphere							
Pressure profile	Ionization gages (2)	0 - 0.5 torr	0.9	1			
Temperature profile	Thermistors	-100° to +30°C	0.4	0.05	10 per minute	40 bits per sample	
Composition	Mass spectrometer	1 - 30 amu	6.8	10			
Exobiological							
Nonexistence of life	Gulliver detector (2)		2.7	2	One per hour	4 bps each	
Presence of organic material	Multivator (2)		2.7	2		4 bps each	
	Total		42.3 (93 lbs)	56.07 (56 watts)			

TABLE 5-7. SCIENTIFIC PAYLOAD FOR ORBITER SUPPORT FOR MARS LANDER

Experiment	Instrument	Range	Weight, kg	Power, ergs sec ⁻¹ x 10 ⁻⁷	Sample Rate	Bit Rate
Radiation						
Ionization rate	Neher chamber		1.1	0.5		
Cosmic ray	GM tube array	1 - 10 ⁴ pont. cm ⁻² sec ⁻¹	3.6	2		10 bps
Trapped radiation	GM tube array					
IR radiation	Hughes radiometer*	5 channel	6	7		4 bps/channel
UV radiation (looking toward ϕ)	Mariner III*	3 channel	3	5		4 bps/channel
Atmosphere						
Occultation experiment	Occultation receiver (Mariner IV)		3	6	600 sec/orbit	2000 bps
Surface						
Photographs	Mariner IV TV*		6.2	9	10 per periopsis pass	2.4 x 10 ⁵ bits/picture
Cosmic Dust						
	Capacitor detector Mariner IV		2.5	2	1 hr ⁻¹	4 bps
		Total	25.4	31.5		
	Platform and servos and electronics		8	4		
		Total	33.4 (73.5 lbs)	35.5 (35.5 watts)		

*Platform for Planet Orientation Experiment

TABLE 5-8. SCIENTIFIC PAYLOAD FOR JUPITER FLYBY

Experiment	Instrument	Range	Weight, kg	Power, ergs sec ⁻¹ x 10 ⁻⁷	Bit Rate, bps	Sample Rate or t	Total Bits
Magnetic field	Magnetometer						
	He*	0.5 to 10 Gauss	3	5	24	10 per minute	(40 hours) 3.46 x 10 ⁶
	Flux gate (electronics)	±500 T	0.9 0.9	0.2 1.5	4	1 hr ⁻¹	1.44 x 10 ⁴ hr ⁻¹
Particle radiation	Counters						
		10 ⁹ to 10 ¹³ el. cm ⁻² rec ⁻¹ kev ⁻¹			10,000		
Electrons	Low E 40 kev	10 ⁴ to 10 ⁶ el. cm ⁻² rec ⁻¹ mev	3.6	3	1000	Interplanetary 7 hr ⁻¹ Planetary 7 sec ⁻¹	(8 hours) 4.6 x 10 ⁸ bits
Protons	High E	10 ⁸ to 10 ¹¹ cm ⁻² sec ⁻¹	(1.15)	(0.4)	5000		
(Trapped radiation)	GM tube array*						
Atmosphere							
			5	5	2000	1200 seconds	2.4 x 10 ⁶
Ionosphere scale Height (composition)	Occultation receiver Fresnel diffraction pattern						
Composition	UV photometer	3 channels	3	5	4/channel		(40 hours) 1.73 x 10 ⁶
Temperature	IR radiometer	5 channels	6	7	4/channel		(40 hours) 2.88 x 10 ⁶
Cosmic dust	Capacitor detection*		1.2	1.5	4	Interplanetary 1 hr ⁻¹ Planetary 6 hr ⁻¹	1.44 x 10 ⁴ /hr ⁻¹ 3.46 x 10 ⁵ /hr ⁻¹
Visual observation clouds (surface?)	Mariner IV TV camera		6.2	9	240,000 bits/picture (10 ⁴ bits/sec ⁻¹)	100 pictures	2.4 x 10 ⁷
	5.35 kg reg.	6.9 x 10 ⁷ reg. sec ⁻¹	29.8 (65.6 lbs)	40.2 (40.2 watts)			

* Minimum payload.

The main objective of this mission is to make direct measurements of the Martian atmosphere and the surface characteristics and some of its properties. A representative payload that would accomplish this end is shown in Table 5-6. The experiments selected and instruments chosen are to illustrate to the designer the problems of visibility, mounting, and data requirements. Most of the experiments have already been successfully conducted in outer space; the two listed that have not are the experiments of exobiological nature. Because of their limited duration, two of each of the instruments are considered.

In the event of any failure whatsoever, it is considered advisable to have some scientific backup so that the mission would not be a total loss. The representative payload is extended to consider a reasonable orbital package with scientific capability considerably beyond the Mariner IV. The scientific experiments and instruments used for spacecraft design considerations are listed in Table 5-7. Three of the instruments are required to point toward the planet and will require the use of a servoed platform to maintain the proper orientation.

Scientific Payload for Jupiter Mission

The allowable scientific payload for exploration of the planet Jupiter does not allow the versatility of landers or subsatellites as does the Mars mission. Much more care must be exercised in the selection of experiments because of the reduced allowed weight and the greatly reduced data transmission capacity.

The main objective of the initial exploration of Jupiter space is to determine the existence or nonexistence of a magnetic field and its strength at a given altitude. The other objective that is related to the magnetic field is the extent of the hypothesized trapped radiation zones. Various combinations of magnetic field and trapped particle fluxes have been hypothesized to produce the 3 to 30 cm radio signals that come from around Jupiter and not at its center.

This then would be considered the minimum scientific objective and the payload to accomplish as shown by the asterisks on Table 5-8.

Out-of-the-Ecliptic

It is hypothesized that the greater part of the solar atmosphere and its contaminants are confined to or near the ecliptic plane of the solar system. The thickest part will be around the nucleus of the solar system but the thickness is not known.

The scientific objectives of solar system exploration above the ecliptic plane are:

- 1) Measure interplanetary dust distribution to see if major portion is confined to or near the ecliptic plane.

TABLE 5-9. SCIENTIFIC PAYLOAD FOR OUT-OF-THE ECLIPTIC MISSION

Instrument	Weight, kg	Power $\times 10^7$ ergs sec ⁻¹	Purpose	Instrument	Weight, kg	Power $\times 10^7$ ergs sec ⁻¹	Purpose	
TV camera	10	10	Photosphere and coronal pictures	Magnetometer, He	*3.0	5.0	Magnetic field	
Servoed platform	5	4		Fluxgate	*0.9	1.7		
UV spectrometer (radiometer 3 frequency)	3	6		Solar plasma probe	*2.1	2.0	Solar wind	10 < p < 1000 km sec ⁻¹
	18	20		Cosmic ray detection				
				Ionization (Neher)	1.2	0.5		
				Solid state telescope	1.2	0.4		0.6 < p < 200 mev
				Nuclear abundance detector	4.1	1.7		10 - 100 mev/nuclear
				Solar proton detector				
				(High energy) GM tubes	1.15	0.4	Proton distribution	
				Positron detector	3.4	1.4		
				Cosmic dust detector	*1.2	1.5	Dust distribution	
				Pulse height analyzer	0.9	2.0		
				Total	19.15 (42.1 lbs)	16.6 (16.6 watts)		
				Lower level of payload				
				Total Weight = 37.15 kg (81.7 lbs) Power = 36.6×10^7 ergs sec⁻¹ (36.6 watts)				

* Minimum payload: Weight = 7.2 kg
Power = 10.2×10^7 ergs sec⁻¹

- 2) Map solar magnetic field and detect if possible a solar-galactic magnetic field boundary.
- 3) Determine change in charged particle radiation in terms of cosmic rays (galactic and solar), and solar wind.
- 4) Measure ratio of H^+ to He^{++} .
- 5) Observe sun from "above" the ecliptic plane (orientation platform required).
- 6) Detect interaction of galactic magnetic field and particle flow with the solar magnetic field.

A representative scientific payload that satisfies the objectives is presented in Table 5-9. Also shown is the minimum payload that would have scientific significance to justify the mission.

Some of the experiments will require orientation toward the sun — those that are to observe the sun. Because the solar panels must also be oriented toward the sun the instrument orientation is greatly simplified. However, a servoed platform still may be used since the attitude control requirements for the solar panel orientation are not as stringent as the requirements for the experiments.

It is extremely difficult to reduce the spacecraft magnetic moment to a value that would not interfere with the magnetometer output without the use of a boom. The largest contributor to the magnetic moment would be the operating solar panels augmented by components and subsystems that have ferromagnetic materials, electro and permanent magnetic moments and currents in electrical wiring. The separation distance between the magnetometer and the spacecraft magnetic center is determined by the required ambient field in the vicinity of the magnetometer. This in turn is constrained by the magnitude of the measurements to be made (i. e., the range of the instrument). Assuming a 3 percent error in cancelling the current induced magnetic moments of the solar panel, the magnetometer must be located at least 8 meters from the solar cell array. Also, assuming that the magnetic moment contribution of the remainder of the spacecraft doubles the solar cell value of magnetic moment, then the distance must be increased to 15 meters to have an ambient spacecraft magnetic field of 0.5 γ .

The ion engines with magnetic beam modulators produce a large residual magnetic field at distances comparable to the dimensions of the spacecraft. Assuming that the magnetic modulator around the thrust chamber acts as a dipole, the maximum value of the flux density at a distance of 10 meters (32.8 feet) from the spacecraft is 14.6 gamma (1.46×10^{-4} Gauss). To reduce the field to 0.5 gamma would require a distance of 30.3 meters (100 feet). These estimates are for a 25 cm ion engine.

The modulator can be either a permanent magnet or an electromagnet. With a permanent magnet modulator, compensation would be required throughout the entire mission even with the engines shut down. With electromagnetic modulation compensation would be required only during the engine operating time. Compensation of the magnetic modulator fields at the ion engines cannot be done without influencing the magnetic field inside the thrust chamber and thereby affecting the beam characteristics. If compensation is necessary, it must be done in the vicinity of the sensitive instrument. This would require knowledge of the direction and magnitude of the field lines at the instrument so that a counter field line can be placed at that location. It might be more desirable to operate the magnetically sensitive instruments after the ion engines have been turned off.

Some of the instruments used for planetary or solar observations will require the use of a hinged platform with a servo control loop to orient the instruments toward the planet while the solar panels and main spacecraft remain oriented toward the sun. The weights of these servoed platforms and power requirements are considered as part of the payload. For the Jupiter flyby the platform can be oriented to a fixed position just prior to the pass and locked in during the pass (similar to the Mariner IV mission). In the case of an orbiter this may not be so where continual reorientating may be necessary for obtaining pictures and measurements from various sections of the observable planet surface.

THERMAL CONTROL

The three items on the Mars orbiter and the Jupiter flyby spacecraft that have a major influence on the thermal design are the electric propulsion engines, the power conditioning equipment for the engines, and the subsystem electronic equipment. The changes in power dissipation from this equipment that must be accommodated by the thermal control system during the mission are quite large. Providing thermal control to the scientific payload is also of primary importance but should not be difficult because the power dissipation in this equipment is relatively low.

Several general spacecraft configurations have been examined to determine the best way to meet the known spacecraft temperature specifications. The goals of the vehicle thermal control system are generally to:

- 1) Provide a spacecraft internal temperature level during transit of $70^{\circ}\text{F} \pm 40^{\circ}\text{F}$ ($21^{\circ} \pm 22^{\circ}\text{C}$) on all subsystem electronics equipment.
- 2) Maintain temperatures of all power conditioning equipment between -22° and 266°F (-30° and 130°C) for all operating conditions where the spacecraft distance to the sun varies between 0.8 AU and 5 AU on the Jupiter flyby spacecraft and 1 AU to 1.67 AU for the Mars orbiter spacecraft.

- 3) Assure that remote subsystems and scientific payload are maintained at reasonable temperature during the complete mission.

These spacecraft thermal control studies showed that these goals can best be met by:

- 1) Isolating the thruster array from the rest of the spacecraft to minimize thermal interactions between the thruster array and the rest of the spacecraft.
- 2) Locating all power conditioning panels for the electric engines together so that the thermal coupling by conduction and thermal radiation can take place between these panels. This thermal coupling is required to keep temperatures of the nonoperating power conditioning panels above their lower limit. The temperature of each power conditioning panel will be controlled with a temperature sensitive louver system for the Jupiter flyby spacecraft.
- 3) All other spacecraft subsystem equipment that does not have to be mounted remotely will be enclosed in a temperature controlled compartment, with louver systems used as necessary to control the temperature.
- 4) On the Mars orbiter spacecraft the temperatures of the remotely located equipment will be controlled by appropriate thermal surface finishes and insulation. On the Jupiter flyby spacecraft all remotely located equipment will have to be controlled with insulation and electrical heaters.

Power Conditioning

Four power conditioning panels, as shown in Section 4, will be used on the Titan III M Mars orbiter spacecraft and six panels will be used on the Jupiter flyby spacecraft. Each of these panels will have an area of about 6.22 square feet. On the Mars orbiter spacecraft the maximum power dissipation per panel will be about 180 watts (2.4 kw at 92.5 percent efficiency) and on the Jupiter flyby spacecraft the maximum power dissipation will be about 225 watts (3 kw at 92.5 percent efficiency). This heat will be allowed to dissipate to space by opening the thermal control louvers. The effectiveness of these thermal control louvers is strongly dependent upon the location of these louvers with respect to the solar panels. The thermal coupling between a louvered radiator on the side of the spacecraft and the solar panel can be quite large. A measure of this thermal coupling can be found by calculating the view factor between the louvers and the solar panels. This view factor was computed for the two solar panel and four solar panel spacecraft configurations with height of the louvers from the solar panels as a parameter. The effect of this coupling is to partially block the view of the louver system to space and to thermally radiate heat from the solar panels into the louver system.

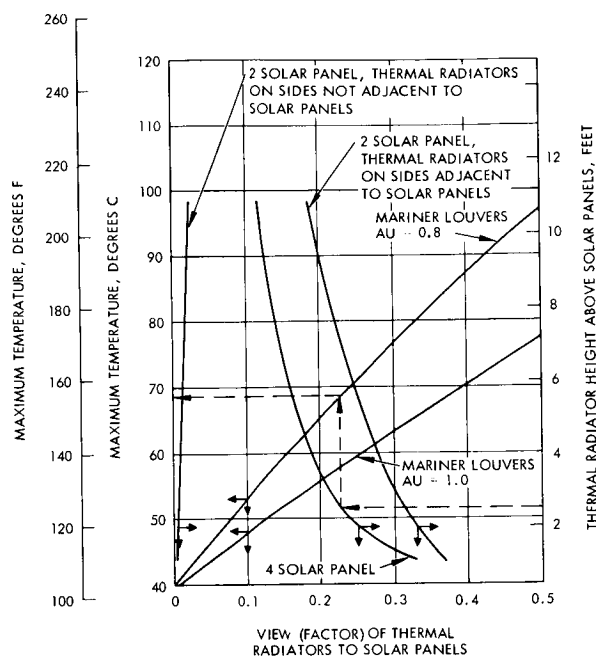


Figure 5-18. Maximum Temperature of Power Conditioning Equipment Versus Thermal Radiator Location

the closed position is 0.12. The weight of the louvers is approximately 0.83 lb/ft^2 and the weight of the actuator is about 1 pound.

All of the power conditioning equipment for the electric propulsion engines will be operating when the spacecraft is near the sun. When this equipment is operating its own power dissipation and the louver system will keep its temperature in a safe operating range. As the distance between the sun and the spacecraft increases, these power conditioning panels will be switched off as shown in Section 4. The temperature of a nonoperating power conditioning panel must also be controlled so that it can be used as a spare in the event of failure of another unit. Approximately 75 watts of power dissipation is required to protect each power conditioning panel when the louver system is closed. On the Mars orbiter spacecraft, the minimum of two panels will be operating. These two panels will dissipate about 360 watts of heat which will be sufficient to keep all of the nonoperating panels above their lower temperature limit if all the panels are located in close proximity to one another. On the Jupiter flyby spacecraft a minimum of at least 450 watts heat dissipation will be required to keep all of the panels above their lower operating temperature. At a distance from the sun greater than 2 AU the power dissipation from the operating power conditioning panels will be less than 450 watts; therefore, additional electrical heating will be required to bring the temperatures of the nonoperating panels above the lower temperature limit prior to turnon.

Shown in Figure 5-18 are the results of these view factor calculations and an estimate of the maximum temperature of the power conditioning equipment when the spacecraft is 0.8 AU and 1 AU distance from the sun. This maximum temperature was calculated for a maximum power dissipation of 225 watts/panel. An example of how to estimate the maximum temperature is shown by the dotted lines on this figure.

The louver system for each power conditioning panel will be actuated by a temperature sensitive pressure transducer. This temperature sensitive pressure transducer will be mounted on the power conditioning panel so that it senses directly the temperature of the panel. The design of the louvers will be similar to the one used on the Mariner spacecraft. The effective emissivity of this louver in the open position is 0.72 and in

Thermal Control for Subsystem Equipment

All of the spacecraft subsystem equipment that does not have to be mounted remotely will be located in a temperature controlled compartment. The temperatures of this compartment will be controlled to 70°F (21°C) $\pm 40^\circ\text{F}$ ($\pm 22^\circ\text{C}$) for all mission conditions. This compartment will be insulated with multilayer aluminized Mylar insulation. The temperature will be controlled by a louver system activated by a temperature sensitive pressure transducer.

The power dissipation in the subsystem equipment for the Mars orbiter and Jupiter flyby spacecraft will vary between 35 and 200 watts. This will require a louver system having an area of approximately 1.7 square feet.

The method to be used to achieve temperature control on the mercury supply tanks and payload for the Mars orbiter spacecraft is dependent on whether the payload end of the spacecraft is in the sun or shade. If the payload end of the spacecraft is in the sun, temperature control on these units can easily be obtained by the proper selection of insulation and thermal finishes. Using these techniques the temperatures of this equipment near Mars (1.67 AU) will be approximately -10°F (-23°C) if the temperature on this equipment is set at 120°F (49°C) for near earth conditions (1 AU). If the payload end of the spacecraft is in the shade, thermal insulation and electrical heating will be required to protect the payload. Thermal protection for the mercury supply tanks can be obtained by insulating the tanks and by conducting heat to them from the hot engine array.

ATTITUDE CONTROL

Attitude Control System Selection

General Attitude Control Requirements

The initial function of the attitude control system is to acquire the sun, then Canopus, from a random orientation — randomness arising from the several hours' interim between spacecraft separation and solar panel deployment.

Once acquisition has been accomplished and electric thrusters are operating, the thruster array is translated to cause the thrust vector to pass through the center of gravity. Thus, misalignment and center-of-gravity position uncertainties are removed. Disturbance torques remaining are those caused by solar pressure which have been estimated at 10^{-3} lb-ft on the 5000 square foot solar array.

Accuracy of control is determined by the high gain antenna. A requirement of approximately 1 db maximum loss means an antenna pointing error of approximately 1 degree. Hence vehicle attitude should be held to $\pm < 1$ degree.

Attitude reference instrument requirements are not out of the ordinary except for the Canopus tracker. This instrument will need an effective field of view of ± 15 degrees to encompass the apparent motion of Canopus as seen from opposite sides of the sun. At least an additional 10 degrees will be needed to permit course corrections obtained by vehicle rotation to rotate the thrust vector. In case terminal correction is required, either a 1-mil accuracy planet tracker (for the Mars mission) or image orthicon will be required; otherwise instrument accuracy of 4 mils is adequate.

No special control requirements arise for the Jupiter flyby or for the out-of-ecliptic missions. However, during the orbiting phase of the Mars mission a horizon scanner may be needed for reference while performing measurements near the planet. Inertial reference during eclipse periods for the Mars orbiter is provided by rate-integrating gyros.

Control Mode Summary. A summary of attitude control system modes showing actuator and sensor utilization for the various mission phases is given in Table 5-10. The phases listed, with exception of the orbit phase, are common to all missions considered in this study.

TABLE 5-10. ATTITUDE CONTROL MODE SUMMARY

Mission Phase	Axes 1 and 2	Axis 3	Attitude Sensors
• Initial stabilization and acquisition	Cold N ₂	Cold N ₂	Integral rate gyros sun, Canopus sensors
• Heliocentric phase			
Thrust phase	Engine translation	N ₂ resistojet	Sun, Canopus sensors
Coast phase	N ₂ resistojet	N ₂ resistojet	
• Reacquisition (if required)	Cold N ₂	Cold N ₂	Integral rate gyros sun, Canopus sensors
• Orbit phase	Cold N ₂	Cold N ₂	Sun, Canopus sensors. Integral rate gyros during eclipse

Resistojets are rockets in which transitions from electrical to thermal to kinetic energy take place. The electrical to thermal conversion is accomplished by resistance heating and the energy is transferred to the propellant principally by convection. This thermal energy is changed to a directed kinetic form in the nozzle. The particular jet proposed here is a close-coupled heat exchanger and exhaust nozzle and is used in an otherwise conventional cold gas type propulsion system. Resistojets are used together with ion engine translation to generate control moments. Resistojets and ion engine translation are utilized whenever full solar power is available.

Following separation from the launch vehicle, the solar panels are deployed and initial angular rates are nulled to the system deadspace by the reaction jet system. The resistojets are employed in the cold gas mode since, prior to sun acquisition, electrical power capacity is insufficient to power the hot gas mode. Caged rate-integrating gyros are utilized to sense spacecraft body rates.

It is recommended that solar panel deployment be completed prior to initiation of rate stabilization. This results in a fuel saving due to increased jet moment arms, and avoids the problem of gas jet location and interference while in the folded configuration. No adverse effect on deployment is expected from separation body rates, which are initially 50 mrad/sec and decrease in proportion to inertia as the panels are unfolded. A block diagram of a single-axis reaction jet control loop for the rate stabilization mode is shown in Figure 5-19. As indicated, derived-rate feedback is preferred for damping because of its simplicity and superior noise-suppression characteristics.

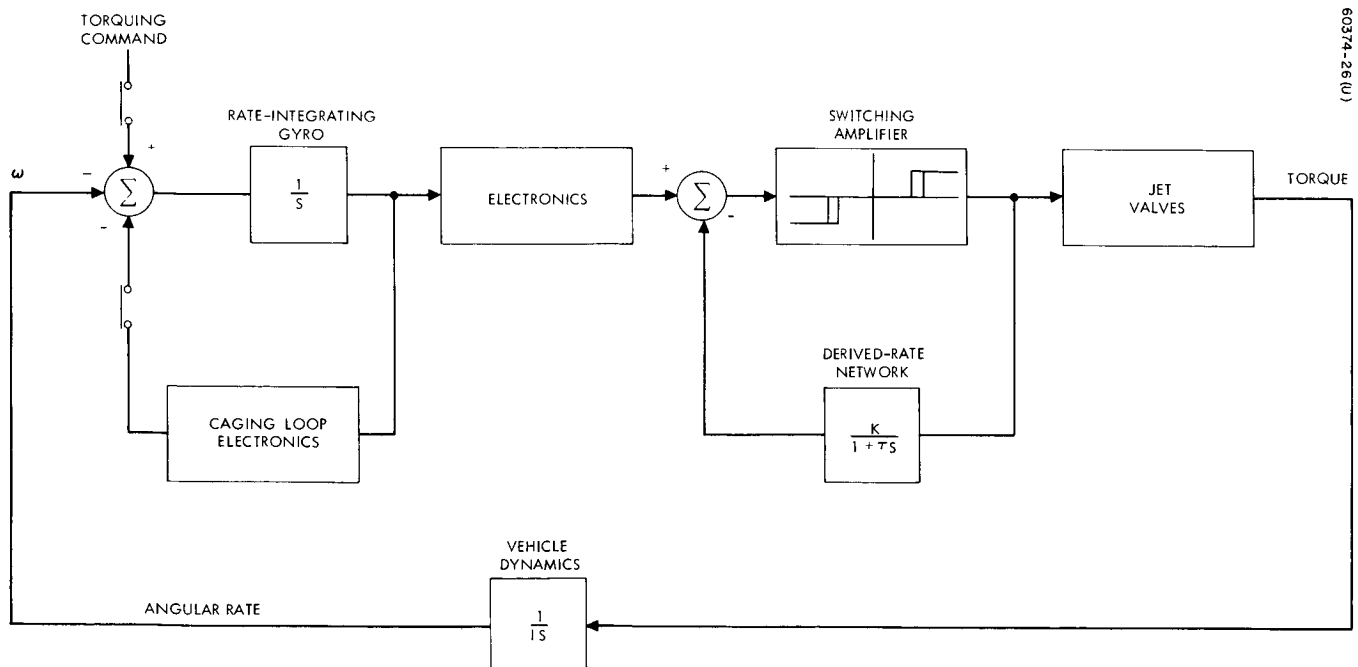


Figure 5-19. Single-Axis Rate Stabilization and Acquisition Control Loop Configuration

Following rate stabilization the sun must be acquired from a random initial orientation. A sequence of two ordered rotations is recommended. The first rotation (pitch) brings the sunline within approximately 2 degrees of the pitch-roll plane, and the second rotation (yaw) then brings the sun into the sun sensor linear range. During this phase all gyros are uncaged and operated in the inertial mode. Gyro torquing at constant rate (0.5 deg/sec) is commanded by the saturated output of the appropriate sun sensor channel.

Following sun acquisition, a roll rotation about the sunline is utilized to locate the star Canopus. Once the celestial attitude reference has been established the gyros can be turned off to ensure their operability during subsequent reorientation maneuvers and orbit phase utilization. Small thrust vector rotations during transit (± 10 degrees) are accomplished by applying bias signals to the optical sensor outputs. A block diagram of a single-axis reaction jet control loop utilizing optical sensor error signals is shown in Figure 5-20.

Also following sun acquisition and verification of full power capacity, all operating gas jets are converted to the hot-gas resistojet mode to obtain maximum efficiency. After ion engine startup, two-axis attitude control is provided by bidirectional translation of the engine bank relative to the remainder of the vehicle. Control about the third axis requires operation of only two of the resistojets. Thus, during the transit thrusting phase the attitude control function is accomplished by engine translation plus two resistojets. During the transit coast phase of the Jupiter mission, all six jets are operated in the hot-gas mode for three-axis control.

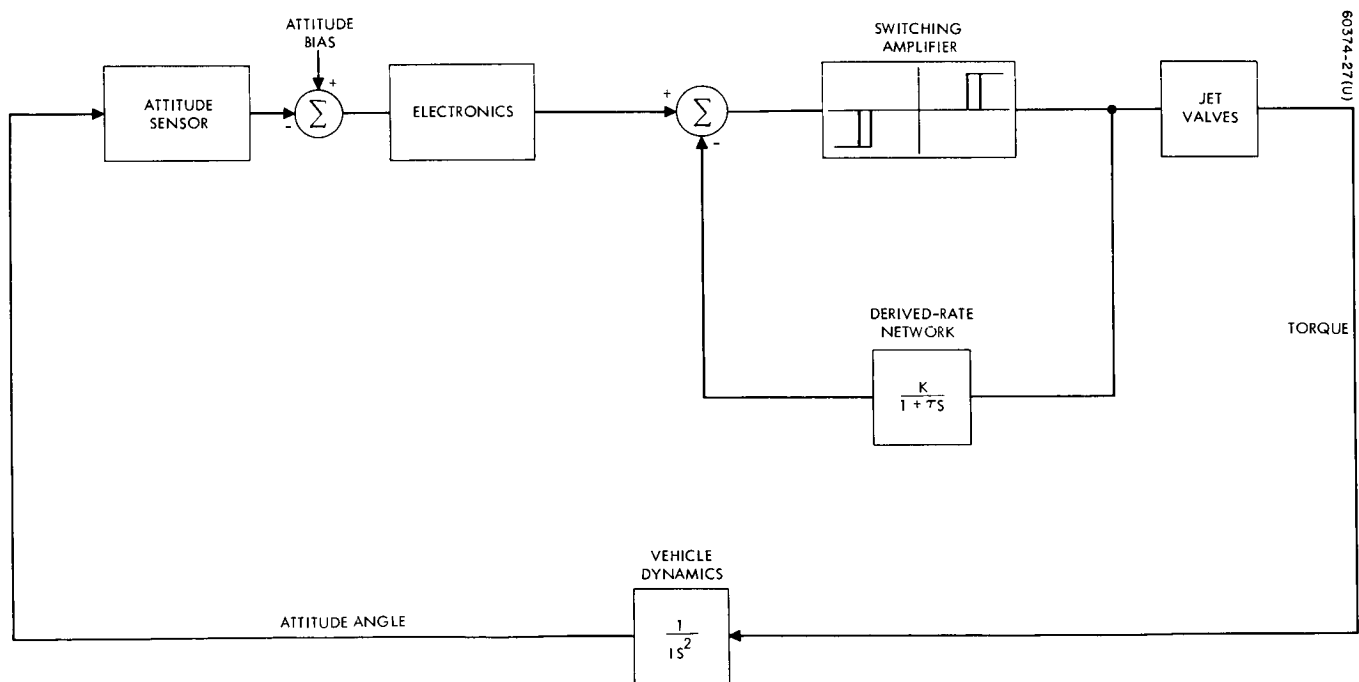


Figure 5-20. Single-Axis, Optical Reference Attitude Control Loop Configuration

In the Mars orbit phase all jets are operated in the cold gas mode due to the lower capacity after solar panel ejection, and because operation in Mars shadow is required. The gyros are operated continuously in orbit, providing inertial reference during shadow period, and are operated in the caged mode during sunlight periods. The spacecraft therefore maintains a sun-pointing attitude throughout the orbital phase unless commanded to do otherwise for the purpose of experiment pointing.

Weight Summary. A summary of the reaction jet control system weight for the Mars orbiter, out-of-ecliptic, and Jupiter flyby spacecraft, for both two and four quadrant solar panel configurations, is given in Table 5-11. Included in the totals is the fixed propulsion hardware required to implement a quad-redundant valve system. As indicated, this hardware represents a significant fraction of the total system weight. For example, it is 42 percent of total system weight in the case of the four quadrant Jupiter flyby configuration. Weight breakdown of the fixed propulsion system hardware for the Mars orbiter and Jupiter flyby spacecraft are given in Tables 5-12 and 5-13, respectively.

Also included in the reaction jet system totals of Table 5-11 is a solar panel weight increment required to furnish electrical power for hot-gas resistojet operation. This is determined on the basis of two jets at 92 watts each, and a solar panel weight of 50 lb/kw at 1 AU. For the Mars and Jupiter trips trajectory studies show that this weight must be approximately doubled due to increased probe-sun distance. The requirement is based on two jets since only this number will be used for third-axis control during ion engine thrusting. During transit coast phase of the Jupiter flyby mission, adequate solar panel power will be available for resistojet operation since the electric propulsion system has been turned off.

TABLE 5-11. REACTION JET CONTROL WEIGHT SUMMARY

Item	Titan III M		Atlas (SLV-3C)/Centaur			
	Mars Orbiter - Four Quadrant, pounds	Mars Orbiter - Two Quadrant, pounds	Out-of Ecliptic - Four Quadrant, pounds	Out-of Ecliptic - Two Quadrant, pounds	Jupiter Flyby - Four Quadrant, pounds	Jupiter Flyby - Two Quadrant, pounds
Fuel (300 percent)	15.2	17.3	12.2	23.7	14.3	30.3
Fuel tanks (2)	18.2	20.8	14.6	28.4	17.2	36.4
Fixed propulsion hardware	30.1	30.1	45.4	45.4	45.4	45.4
Sensors plus electronics	12.0	12.0	12.0	12.0	12.0	12.0
Solar panel increment	18.4	18.4	9.2	9.2	18.4	18.4
Total system weight	93.9	98.6	93.4	118.7	107.3	142.5

TABLE 5-12. MARS ORBITER SPACECRAFT
ATTITUDE CONTROL SYSTEM

		<u>Pounds</u>
Temperature transducer	$0.2 \times 2 =$	0.4
Pressure transducer	$0.2 \times 2 =$	0.4
Fill valve	$0.25 \times 2 =$	0.5
Squib valve (N. C.)	$0.5 \times 3 =$	1.5
Filter	$0.2 \times 6 =$	1.2
Pressure regulator	$1.1 \times 2 =$	2.2
Solenoid valve	$0.4 \times 4 =$	1.6
Thruster assembly	$0.7 \times 24 =$	16.8
Lines	$(11.5 \text{ feet})(0.02 \text{ lb/ft})(24) =$	<u>5.5</u>
		30.1

TABLE 5-13. JUPITER FLYBY AND OUT-OF-ECLIPTIC
SPACECRAFT ATTITUDE CONTROL SYSTEM

		<u>Pounds</u>
Temperature transducers	$0.2 \times 2 =$	0.4
Pressure transducers	$0.2 \times 2 =$	0.4
Fill valves	$0.25 \times 2 =$	0.5
Squib valves (N. C.)	$0.5 \times 3 =$	1.5
Filters	$0.2 \times 6 =$	1.2
Pressure regulators	$1.1 \times 2 =$	2.2
Solenoid valves	$0.4 \times 4 =$	1.6
Thruster assemblies	$0.7 \times 24 =$	16.8
Lines	$(43.5 \text{ feet})(0.02 \text{ lb/ft})(24) =$	<u>20.8</u>
		45.4

A reaction jet system fuel budget for the Mars orbiter spacecraft (four-quadrant configuration) is presented in Table 5-14. This is based on a maximum of 423 days in transit and 180 days in Mars orbit. As indicated, a fuel allowance has been made for various contingencies, including meteorite impact and solar panel ejection disturbances. Valve leakage was not included in the budget since with the proposed system, serial failure of two valves is required to produce leakage, and if this should occur, a spare tank and valves are available.

A detailed summary of the calculations leading to the fuel budget in Table 5-14 is given in Appendix I. Fuel requirements for other spacecraft listed in Table 5-11 were arrived at in an identical manner with the following deviations arising from differences in configuration and mission:

- 1) Mars orbiter, two-quadrant configuration:
 - Different jet locations and vehicle inertia
- 2) Jupiter flyby mission:
 - Jet locations and vehicle inertia
 - 540 days transit thrust phase
 - 360 days transit coast phase
- 3) Out-of-ecliptic mission:
 - Jet locations and vehicle inertia
 - 500 days transit phase

In addition to the reaction jet system weight, an engine translation mechanism weight of 10 pounds and an additional 3 pounds of electronics are required. These weights are included in the structures and mechanisms weight breakdown.

General Comments. The reaction control system weight estimates listed in Table 5-11 are somewhat lower than estimates made previously in the Phase I study report. This stems directly from decisions to use ion engine translation for continuous two-axis control during the long duration transit phase and resistojets in place of straight cold gas thrusters. The previous study was based on 1967 state of the art when cold gas thrusters appeared to be the best choice. For the present study, based on 1970 state of the art, resistojets are preferred for fuel economy. Furthermore, they have been successfully used on the Vela satellite program.

The decision to use ion engine translation for continuous two-axis control is based on analysis showing that entirely adequate performance and reliability are achieved with this type of system. It is estimated that the bidirectional engine translation mechanism reliability is 0.975 for a 500 day utilization period.

TABLE 5-14. REACTION JET SYSTEM FUEL BUDGET
MARSORBITER, FOUR-QUADRANT CONFIGURATION

Mission Phase	Fuel Required, pounds	Remarks
1) Initial rate dissipation	0.325	Cold gas mode
2) Sun and star acquisition	0.335	Cold gas mode
3) Transit phase	3.29	• 423 days • Third-axis control by two resistojets
4) Meteorite impact	0.335	• Cold gas mode
5) Panel ejection disturbance	0.325	• Cold gas mode
6) Mars orbit phase	0.44	• 180 days • Cold gas mode

Total fuel requirements: 5.05 pounds

300 percent fuel requirement: 15.2 pounds

Reaction jet control system redundancy is provided by a dual-tank, quad-redundant valve system. Each tank contains 150 percent of the estimated gas requirement, and four complete sets of six valves are included. In addition, the dual tank system is connected by a squib valve for increased fuel utilization capability. Nine hundred day reliability has been estimated at greater than 0.98 for this system.

Failure of an ion engine during thrust phase causes no particular attitude control problem. On-board logic immediately turns off the opposing engine such that no substantial thrust vector shift occurs, and shortly thereafter, allowing time for small transient motions to decay, a spare symmetrical pair of engines are activated. No loss of attitude reference should occur as the response of the engine translation mechanism is adequate to maintain control during such perturbations.

Reaction Jet System Considerations

Comparison of Mass Expulsion Thrustors

Before arriving at a decision to employ resistojets for attitude control actuators, a comprehensive survey of the available alternatives was conducted. Critical evaluation narrows consideration to a few that might be

suitable for long duration missions of this type. A summary of the comparison of available alternatives is given.

Cold Gas. Nitrogen or argon are possible fuels for cold gas mass expulsion thrusters. Argon has a 25 percent lower I_{sp} than cold N_2 (70 seconds versus 53 seconds) but has a higher density and is more compressible. Hence the total system weight for either fuel is about the same. Cold gas mass expulsion thrusters have low power requirements, high reliability, and have been proven operational in many space applications. It has been shown that approximately 60 percent of the system weight for a cold gas system is tankage.

Hydrogen Peroxide. Hydrogen peroxide would be a likely choice as a propellant if usage were restricted to long-pulsed operation. When used as a constant flow thruster the I_{sp} of hydrogen peroxide is 250 seconds; when used in short pulses the I_{sp} is 100 seconds. This lower value for pulsed I_{sp} operation, its high cost, detract from its desirability as a propellant for limit cycle operation.

Resistojets. Resistojets are electro-thermal thrusters which consist of the following parts: 1) A heater whose temperature is raised by I^2R drop. This heat energy is transferred to the propellant by convection as the propellant flows by. 2) A nozzle where this thermal energy of the propellant is converted into a directed kinetic form. They are capable of being operated either in the hot-gas resistojet mode or the cold-gas mode, thereby providing a degree of flexibility.

Resistojets using nitrogen, ammonia, helium, and hydrogen have been built. Their relative efficiencies are shown in Figure 5-21, a plot of I_{sp} versus operating temperature for a specific design. Hydrazine is not considered as a propellant for resistojets because it has presumably been rejected by TRW, GE, AVCO, Marquardt, and Grumman (all resistojet developers). Hydrazine is a common propellant in hypergolic systems. The specific impulse for resistojets varies from 123 seconds for a design using nitrogen to 840 seconds for a hydrogen design. An additional consideration when using resistojets is the electrical power which they require. The only operational resistojet design, the 0.042 pound thruster for the Vela satellite, requires 92 watts of electrical power. The AVCO hydrogen design develops approximately 0.12 pound thrust using a 3 kw heater.

The tankage weight penalty should be approximately the same as for cold gas; this was found to be true on Vela.

Resistojets have the additional reliability constraint of requiring the heaters and heater circuits to work. However, should a heater failure occur the thruster will still operate as a cold gas thruster.

Hypergolic Bipropellants. A bipropellant system using hypergolic oxidizer and fuel is an example of a hot gas system. The thruster for such a unit would be about 0.25 to 0.50 pound for an I_{sp} of 150 seconds. An additional tankage penalty must be paid for the second propellant. The

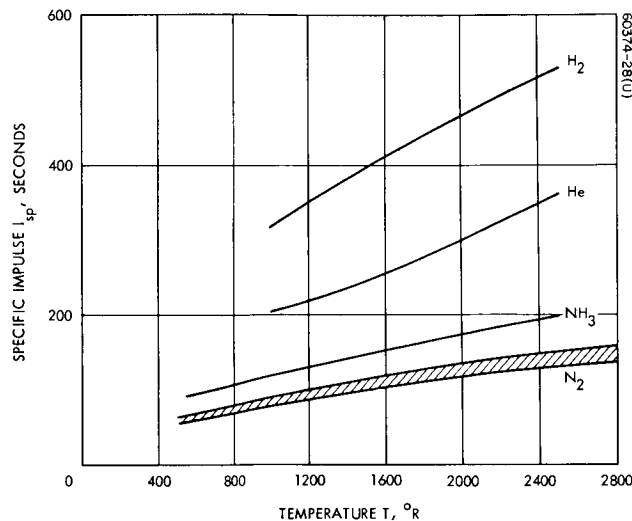


Figure 5-21. Specific Impulse with Ammonia, Helium, and Hydrogen

separation of the gas and liquid, and how to maintain the water solution between 30° and 100° F.

Conclusion. It is recommended that a resistojet thruster system be used for attitude control on the electric propulsion vehicle. A nitrogen resistojet is recommended for the following reasons:

- 1) It has been proven on the Vela satellite.
- 2) It is reliable — approximately equal in reliability to the most reliable thruster system, cold gas.
- 3) It has a higher I_{sp} than cold gas— 123 seconds. For later missions it may be possible to switch to a still more efficient fuel if such has been proven operational by then.
- 4) The electric power requirements are moderate.

Reaction Jet System Configuration

The reaction control system consists of two connected subsystems each utilizing two sets of six thrusters fed from a regulated gas source stored in a high-pressure reservoir. A schematic diagram of the system is shown in Figure 5-22.

Two titanium alloy pressure bottles constitute the gas reservoirs. Each reservoir holds 150 percent of the estimated total gas requirement. The nitrogen gas is regulated to the thruster supply pressure by a single-stage pressure reducer. Solenoid valves situated adjacent to each thruster are activated on command to provide control torques about the pitch, yaw, or roll axes.

reliability of such a system is lower since leakage could result in a catastrophic failure and both tanks are subject to failure. Because of the presence of an oxidizer, hypergolic systems pose the additional problem of corrosion. Corrosion resistant materials are generally weaker than most steels and so result in a heavier tank design for a given shear strength requirement.

Water Electrolysis Rocket.

Water electrolysis rockets were eliminated from consideration for this spacecraft mainly because they are not space proven. Also the spacecraft will not be spinning and the following problems would have to be circumvented: how to ensure

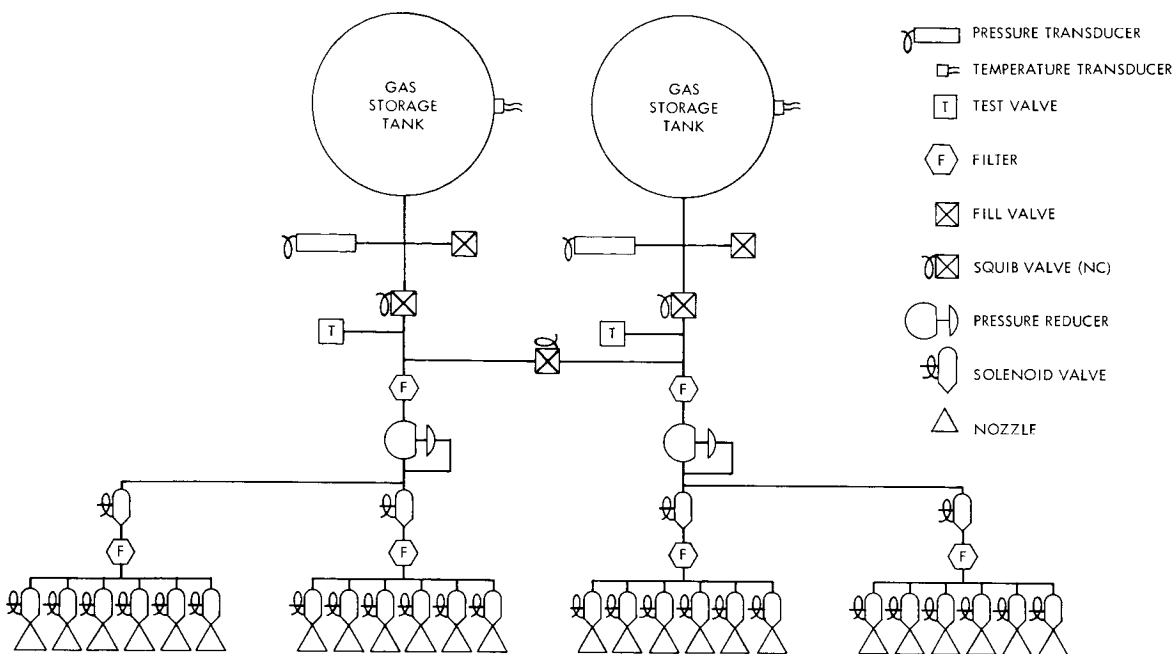


Figure 5-22. Schematic Diagram of Reaction Jet System

A normally closed squib valve (pyrotechnically actuated) isolates the gas supply from the pressure reducer until a command is given (just prior to launch) to fire the squib. A larger solenoid valve, the main valve, is placed in series with the pressure reducer and in series with each set of six thruster valves. The appropriate main valve is opened each time a thruster solenoid is activated. Filters, fill, and checkout valves complete the subsystem. The two subsystems are connected via a normally closed squib valve. Thus, gas from one subsystem can be diverted to the other as required by ground command.

In the case of the four quadrant Mars orbiter spacecraft the thrusters are all located at the outer edges of the innermost solar panels. In the two quadrant configuration four jets per set of six are mounted on the inner panels and two are mounted on opposite ends of the body. For the Jupiter flyby and out-of-ecliptic spacecraft the jets are mounted similarly, but on the outermost panels, since for these cases the panels are not ejected. Flexible lines similar to those employed on the Surveyor spacecraft are required to handle the bending during solar panel deployment. The method successfully employed on Surveyor consists of placing stainless steel bellows sections surrounded by braided steel jackets in the gas lines at the bend point.

Reliability

The requirement for reaction control system redundancy on the long-duration missions considered in this study is adequately met by the dual-tank, quad-redundant valve system recommended here. Reliability analyses

conducted at Hughes on a similar dual-tank system involving only doubly-redundant valves yielded a reliability figure of 0.99275 for 1 year's operation. For 500 and 900 days' operation this becomes 0.9901 and 0.9821, respectively. With the additional two sets of valves the proposed system should yield substantially better reliability figures than these.

The component reliability figures on which these analyses were based are given in Table 5-15. The valve reliability numbers are given in terms of percent failure during a 10^5 hour exposure; however, these data can also be applied to valves whose cycle life is as critical as exposure life.

TABLE 5-15. REACTION CONTROL SYSTEM COMPONENT FAILURE RATE DATA

Component	Failure Rate (Percent Failures per 10^5 Hours)	Failures per Year	Source
Main valve total	0.430	0.03766	AVCO
Open	0.108	0.00946	Computed
Closed	0.108	0.00946	Computed
Leak	0.215	0.01892	Computed
Thrustor valve total	0.140	0.01226	AVCO
Open	0.035	0.00307	Computed
Closed	0.035	0.00307	Computed
Leak	0.070	0.00613	Computed
Nitrogen tank	0.004	0.00035	AVCO
Solenoid driver	0.060	0.00526	ATS
Pressure reducer	0.036	0.00315	AVCO
Pressure transducer	0.700	0.00613	AVCO
Manifolding	0.030	0.00263	AVCO
Fill valve	0.010	0.00088	AVCO
Filters and nozzles	0.002	0.0017	AVCO

Also included in the reliability analysis of the doubly-redundant valve system is a comparison with other system configurations. The results of this comparison are rather interesting and are shown in Table 5-16. It is noted that the next best system, involving the same number of thrusters but without the main solenoid valves or cross-strapping of the fuel supply, yielded a 60 percent higher failure rate. The other systems considered in Table 5-16 — one a single system with a main valve, the other without a main valve — yield failure rates of 15 and 19 times that of the dual tank, doubly-redundant valve system. These results substantiate the design proposed for the electric propulsion spacecraft application, both in selection of the dual connected systems and in the use of main solenoid valves to isolate a malfunctioning thruster valve.

As indicated by the component failure rate data given in Table 5-16 the solenoid valves (main valve and thruster valves) cause the greatest degradation to overall reliability. Various solenoid valve failure modes are listed in Table 5-17 together with relative likelihood of occurrence and the resulting effect. Only simple failure modes involving a single component failure are covered in the table, but none of these modes results in a half-system failure. In two of the cases listed, a thruster valve failed open or closed, the system simply reverts to the second set of valves for that half-system and thereafter never opens the original main valve.

A half-system failure would be caused, for example, by a main valve and thruster valve serially failing open. The probability of this occurring during a 900 day mission is approximately 10^{-3} . In such an event the thruster opposing the failed thruster would be opened automatically by the attitude control system, and thrusting would continue until the half-system fuel was depleted. The second half-system would then be activated for the remainder of the mission.

TABLE 5-16. REACTION CONTROL SYSTEM RELIABILITY COMPARISON

Configuration	Description	Reliability for 1 Year	Relative Unreliability
1	• Dual, connected systems • Two sets of thrusters • Main solenoid valves	0.99275	1.0
2	• Two parallel systems • No main solenoid valves	0.98869	1.6
3	Single (nonredundant) system with main valve	0.88834	15.4
4	Single (nonredundant) system without main valve	0.85779	19.6

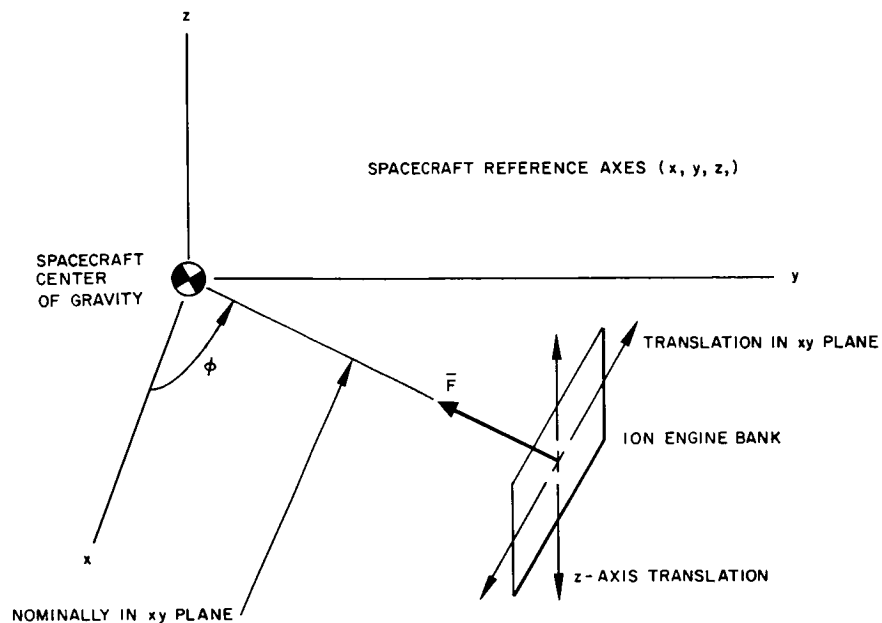
TABLE 5-17. FAILURE MODES AND EFFECTS ANALYSIS

Component	Failure Mode	Relative Likelihood	Immediate Effect	Result on Half-System
Main Valve	Open	1/4	None	None
	Leak	1/2	None	None
	Closed	1/4	Blocks passage of gas	Revert to alternate set of valves
Thrustor Valve	Open	1/4	Erroneous torque loss of gas	Revert to alternate set of valves
	Closed	1/4	Loss of torque capability	Revert to alternate set of valves
	Leak	1/2	Slow loss of gas during operation of other valves only	None

Discussion of Two-Axis Attitude Control by Means of Ion Engine Translation

During the heliocentric thrusting phase of the electric propulsion spacecraft missions considered in this report, attitude control about two axes is accomplished by bidirectional translation of the bank of ion engines. The justification for this approach is as follows. It is clear that some sort of thrust translation (continuous or intermittent) is necessary, since disturbance torques from thruster vector to cg displacement would otherwise require an inordinate amount of momentum dumping via mass expulsion, and hence a huge weight penalty. Given that ion engine thrust is available for control, and that a continuously operating translation mechanism can be built which satisfactorily and reliably performs the control function, it is concluded that the optimum approach is to use thrust translation for continuous control about two axes. Thus, other than a gas jet system for backup in case of anomolous performance or meteorite impact, no supplementary control capability need be provided about these axes.

The general spacial relationship of the engine bank relative to spacecraft axes is shown in Figure 5-23. Attitude control about the third axis, which coincides with the engine thrust vector, is provided by the resistojet reaction control system. Since the control axes do not coincide with the spacecraft XYZ axes, a simple coordinate transformation is required to convert error signals from the sun and Canopus sensors into control axis error signals. This is accomplished by straightforward electronic mixing of the sensor error signals. During mission periods when ion engine thrusting is not used (acquisition, Mars orbit phase) this coordinate transformation



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Figure 5-23. Spatial Relationships of Spacecraft and Ion Engine Bank

is by-passed and the sensor errors are used directly by the reaction control system.

Translation of the bank of ion engines in each of two orthogonal directions is accomplished by a low-level stepper motor which is geared down to produce 0.006 inch of linear travel per step. Command pulses are applied to the motor from a voltage controlled oscillator (VCO) with an output pulse rate proportional to input signal.

A block diagram of the preferred attitude control loop configuration, involving an open-loop translation mechanism, is shown in Figure 5-6. Performance characteristics are discussed in detail in following sections.

Design Alternatives

From a control loop design standpoint either a stepper motor, dc torque motor, or ac servomotor could be used to provide engine translation control. A stepper motor was selected as being well qualified for long-duration spacecraft applications. The greater inherent reliability of this type of motor makes this choice relatively clear cut.

A bang-bang or on-off type of system, obtained by introducing a dead-band into the attitude control loop, was considered but found to offer no real advantages. Actually, the proposed system (Figure 5-24) operates

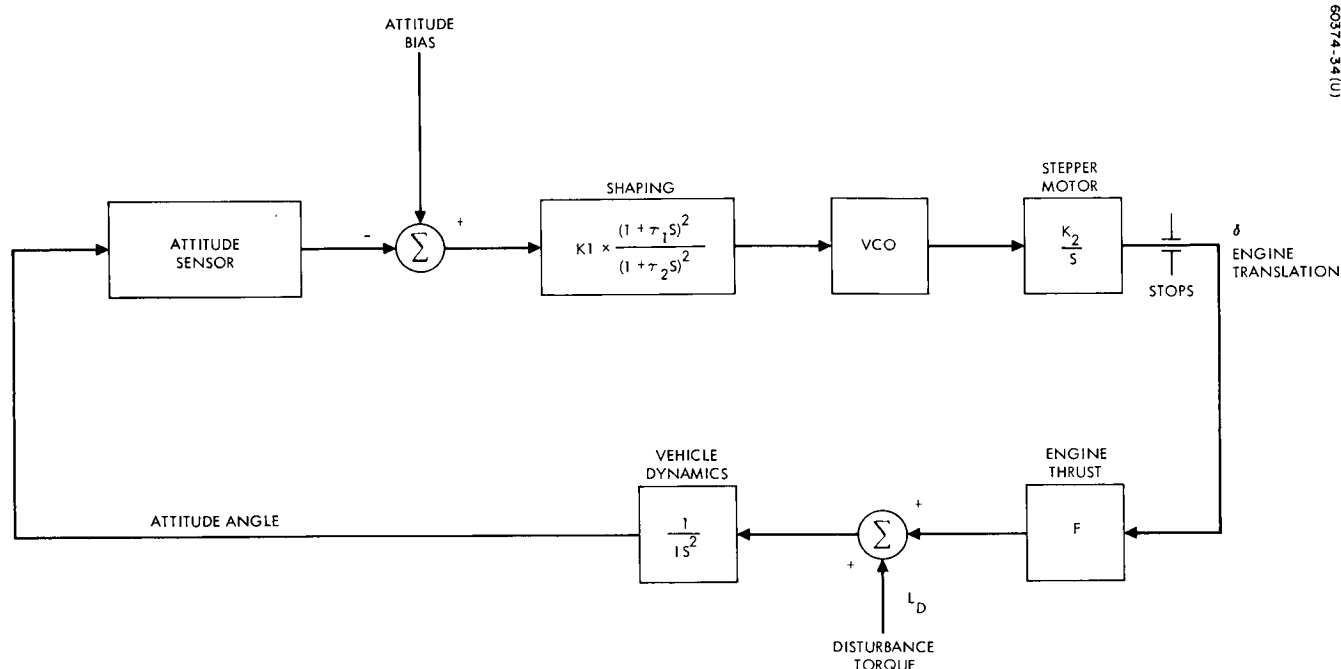


Figure 5-24. Single-Axis Attitude Control Loop Utilizing Open-Loop Engine Translator (Configuration B)

in an on-off mode at the stepper motor input. Thus, the advantages of on-off operation in allowing highly efficient and highly reliable power transfer are realized.

The proposed design involves using the engine translator in an open-loop configuration (i. e., without translator position feedback) within the attitude control loop. An alternate to this would be to employ position feedback to form a subsidiary translator loop within the attitude loop. Analysis shows that the translator loop approach yields lower reliability and greater susceptibility to solar panel interaction, although otherwise adequate attitude control characteristics are obtained. A detailed comparison of these two configurations is given below.

Preliminary Design of Translator Attitude Control Configurations

Preliminary design of two attitude control system configurations involving ion engine translation has been worked out for the Titan III M/Mars orbiter spacecraft. These designs are discussed and quantitatively compared in the following sections. The primary purpose for doing this was to allow detailed evaluation of the continuous engine translation control concept for a specific design case. This is necessary to obtain quantitative answers to questions relating to the following important aspects of system performance:

- Stability and dynamic performance
- Control of disturbance torques
- Steady-state attitude accuracy
- Total motor cycles required
- Interaction with solar panel structure
- Reliability

Each of these items is discussed in relation to each of the two alternate system configurations considered.

System Design With Closed-Loop Translator

A functional block diagram of a single-axis attitude control loop, designated as configuration A, involving a closed-loop engine translator is shown in Figure 5-25. As indicated, in addition to the translator loop, the system includes an attitude sensor and electronic shaping. An attitude biasing capability is provided to allow in-flight trajectory corrections.

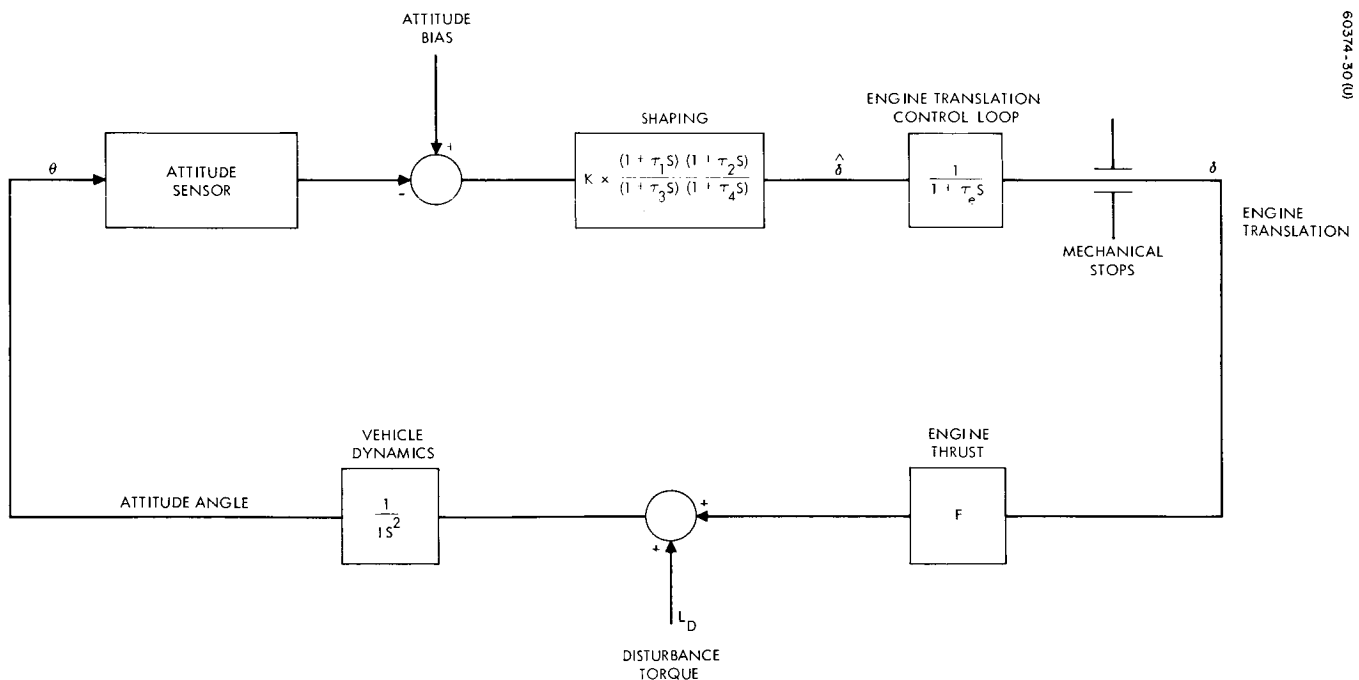


Figure 5-25. Single-Axis Attitude Control Loop Utilizing Closed-Loop Engine Translator (Configuration A)

The closed-loop engine translation mechanism is shown schematically in Figure 5-26. The basic elements of the translation loop are

- Noise filter
- Voltage-controlled oscillator (VCO)
- Stepper motor
- Engine bank position indicator

A simple low-pass noise filter is included to minimize the effect of sensor and electronics noise on system operation. Extremely slow operation of the system makes it possible to provide substantial filtering without appreciably influencing closed-loop dynamics. By means of a VCO, the filtered position error signal controls the rate at which pulses are applied to the stepper motor.

Characteristics of the stepper motor chosen for preliminary design purposes are listed in Table 5-18. As indicated, each VCO pulse results in one-quarter revolution of the motor or 0.006 inch linear travel of the ion engine bank. Maximum pulse rate capability of the motor is 215 pulses per second, which yields a maximum linear travel rate of 1.29 in/sec. In normal operation, motion rates will be one or two orders of magnitude below these maximum values.

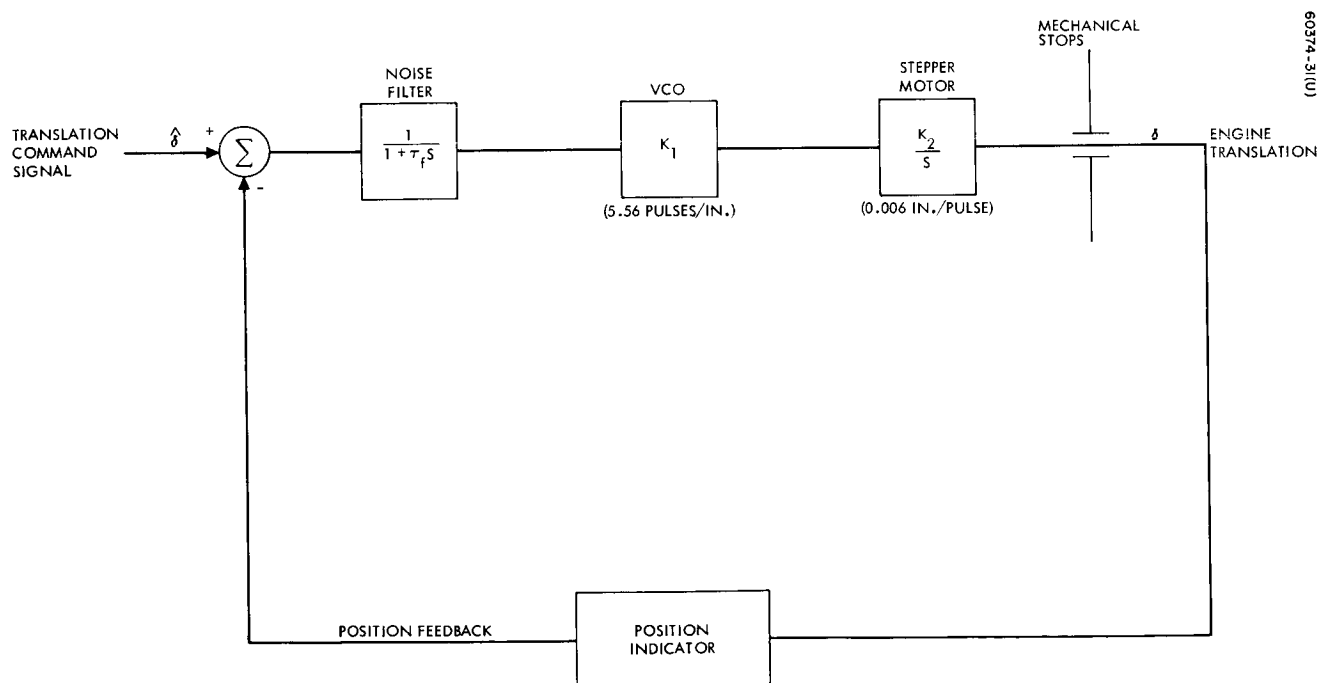


Figure 5-26. Engine Translation Control Loop

TABLE 5-18. TRANSLATION MECHANISM CHARACTERISTICS

Maximum linear travel	±5 inches
Maximum rate capability	±1.29 inch/second
Stepper motor revolution per pulse	90 degrees
Linear travel per pulse	0.006 inch
Maximum pulse rate capability	215 pps
Total cycles capability	>10 ⁹ pulses
Maximum motor power	2.8 watts

Translator Loop Parameter Selection. Translator loop parameters which yield satisfactory closed-loop performance are chosen as follows:

- VCO responsivity, K_1 5.56 pps/inch
- Stepper motor and drive mechanism responsivity, K_2 0.006 inch/pulse
- Noise filter time constant, τ_f 2 seconds
- Loop time constant, $\tau_e = \frac{1}{K_1 K_2}$ 30 seconds

The closed-loop transfer function of the translation loop, from translation command to actual output travel, is

$$\frac{\delta}{\delta} = \frac{1}{1 + \tau_e S + \tau_e \tau_f S^2}$$

With the parameter values listed above this becomes

$$\frac{\delta}{\delta} = \frac{1}{(1 + 28S)(1 + 2.1S)}$$

which, for the purpose of preliminary analysis, may be approximated by

$$\frac{\delta}{\delta} = \frac{1}{1 + \tau_e S} = \frac{1}{1 + 30S}$$

In arriving at this closed-loop transfer function the VCO/stepper motor combination has been linearized. This approximation is valid for all but very small signal operation since time between pulses is small compared with system response time. Small signal behavior during steady-state, limit-cycle operation is discussed in another section.

It can be noted that the noise filter time-constant of 2 seconds provides attenuation of frequencies above 0.08 Hz. Since the frequency of internally generated noise from sensors and electronics is typically much higher than this, spurious motor pulsing due to this effect is greatly reduced.

Attitude Control Loop Parameter Selection. Spacecraft parameters which influence attitude control loop operation are as follows:

- Spacecraft moment of inertia, I 5,000 slug-ft²
- Net ion engine thrust, F 0.06 pound

Control system parameters that yield satisfactory performance characteristics are chosen as follows:

- Control channel static gain, K 5 in/deg
- Shaping time constants
 - τ_1 30 seconds
 - τ_2 50 seconds
 - τ_3 3 seconds
 - τ_5 5 seconds
- Net loop gain, $C = \frac{KF}{I}$ $2.86 \times 10^{-4} \text{ sec}^{-2}$

The basis for selecting these parameters is: first, since attitude control accuracy is required to be approximately 1 degree, and since ± 5 inches of engine translation capability must be provided, the control channel static gain is determined to be 5 in/deg. With this quantity fixed, little freedom exists in selecting the shaping time constants. τ_1 is set at 30 seconds to compensate for the translator lag, and τ_2 is set at 50 seconds to yield a reasonable degree of closed-loop stability. A lead-lag ratio of 10:1 is used since this is readily realized physically.

The characteristic equation for the closed-loop roots can be written as

$$s^2 + \tau_2 Cs + C = 0$$

where, for simplicity, the shaping time constants τ_3 and τ_5 are neglected since they have only second-order effect. The undamped natural frequency (ω_n) and damping ratio (ζ) of the characteristic roots can therefore be determined from the parameter values listed above as

$$\omega_n = \sqrt{C} = 0.017 \text{ rad/sec}$$

$$\zeta = \frac{\omega_n \tau_2}{2} = 0.425$$

A root locus diagram for this system is shown in Figure 5-27.

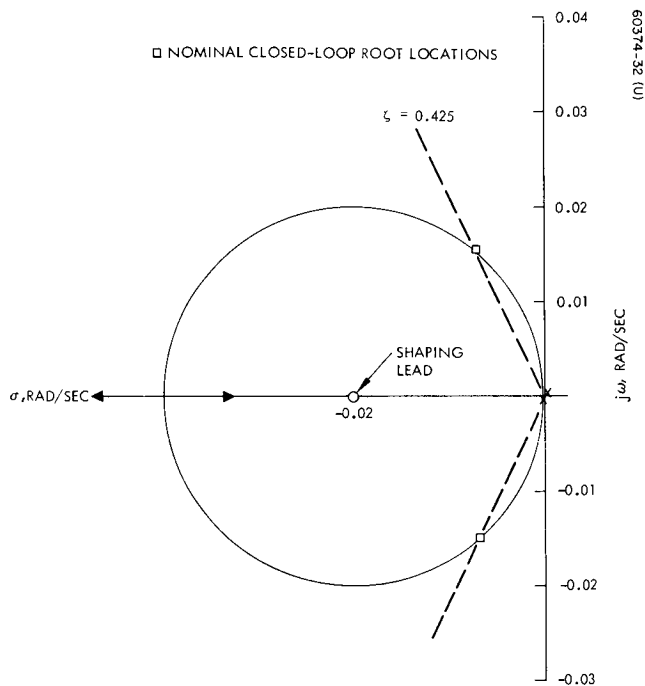


Figure 5-27. Attitude Control Loop Locus (Configuration A)

Steady-State Limit Cycle Operation. In the steady-state condition, when disturbance torques have been corrected by thrust vector translation, system operation will settle into a low-level limit cycle about the equilibrium condition. This mode of operation will prevail during most of the transit phase. Deviations from this mode will occur only when a relatively rapid change in cg location or thrust vector position occurs.

Behavior of the system during steady-state limit cycling can be determined by considering the response to periodic stepping of the motor. For simplicity, it is assumed that successive motor steps cause the thrust vector to be symmetrically displaced ± 0.003 inch from the spacecraft cg. The behavior of spacecraft attitude can then be deduced to be the periodic fluctuation as shown in Figure 5-28. With the maximum thrust vector to cg offset known (0.003 inch), peak attitude motions and the time between pulses (T) are computed by the following procedure:

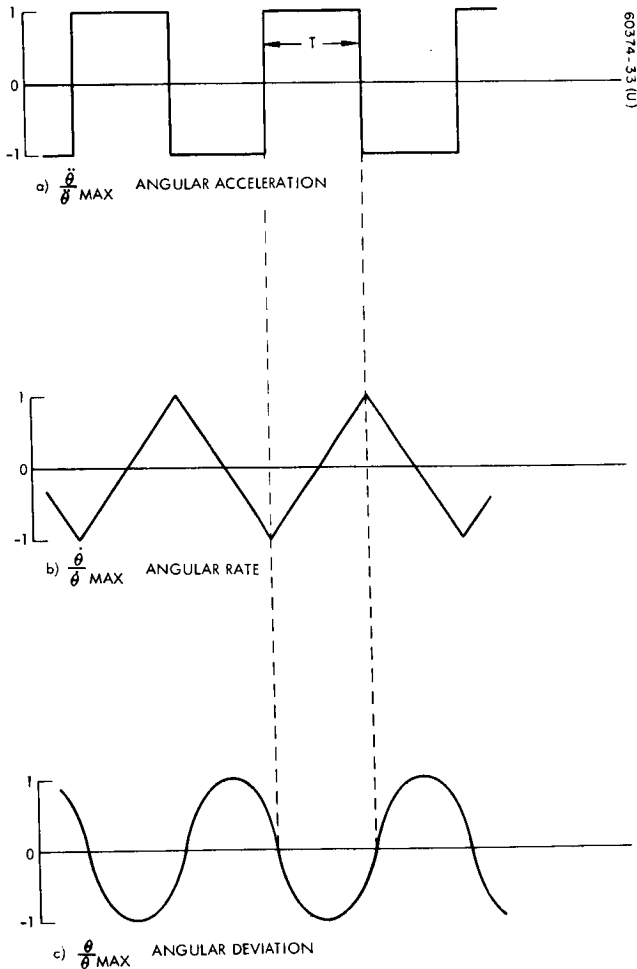
- Maximum cg offset is $\delta_{\max} = 0.003 \text{ inch}$
- Angular acceleration: $\ddot{\theta}_{\max} = \frac{F}{I} \delta_{\max} = 1.72 \times 10^{-7} \text{ deg/sec}^2$

- Maximum angular rate: $\dot{\theta}_{\max} = \frac{1}{2} \ddot{\theta}_{\max} T$
- Peak attitude deviation: $\theta_{\max} = \frac{1}{8} \ddot{\theta}_{\max} T^2$
- Time between pulses, T , is determined by considering the VCO input signal immediately following a pulse execution. The next pulse (T seconds later) is generated when

$$\int_0^T (\hat{\delta} - \delta) dt = \frac{1}{K_1}$$

Evaluation of this integral yields the relation,

$$\frac{1}{T} = K_1 \delta_{\max} \left\{ 1 + C \left[\tau_1 \tau_2 = \frac{T^2}{12} \right] \right\}$$



With the system parameter values listed above, this yields for time between pulses, $T = 43$ seconds. Thus, the theoretical minimum cycling rate of the system is one motor step every 43 seconds. Actual steady-state operation will, of course, be influenced by noise and other system imperfections, but it can be concluded from this result that steady-state cycling will not be at a sufficient rate to cause wear-out problems for missions up to several years' duration.

Attitude Control Loop Design With Open-Loop Translator

A functional block diagram of attitude control loop configuration B, which utilizes an open-loop engine translator, is shown in Figure 5-24. This differs from configuration A (Figure 5-25) only in that translator position feedback is not used, and system gain and time-constants are selected differently. The principal functional difference arises from the integrating nature of the stepper motor which removes the constraint on system static gain and allows considerably greater freedom in selection of parameter values.

Figure 5-28. Steady-State Translator Limit Cycle Operation (Symmetric Limit Cycle Assumed)

The following system parameters are selected to yield satisfactory overall performance characteristics:

- Control channel responsivity, including sensor gain, shaping, and VCO, K' 0.91 pps/deg
- Stepper motor responsivity, K_2 0.006 in/pulse
- Shaping time-constants
 - τ_1 200 seconds
 - τ_2 20 seconds
- Loop gain constant, $C^1 = \frac{K^1 K_2 F}{I}$ $0.312 \times 10^{-6} \text{ sec}^{-3}$

For this system the constraint on parameter values becomes the size of the shaping lead time-constants. These are chosen as large as possible in order that system loop gain can be minimized at higher frequencies to minimize interaction with solar panel structural resonances. Since modern micro-electronic circuitry can achieve time constants on the order of 200 seconds, this value was chosen for the two lead terms, and a lag-lead ratio of 10 was again used. The gain constant K' is then chosen to yield acceptable stability and dynamic performance characteristics, which are determined to be as follows:

- Damping ratio of oscillatory roots, ζ 0.40
- Phase margin 40 degrees
- Gain margin
 - Gain decrease 5.0
 - Gain increase 8.0

It should be noted that this system configuration yields a low-gain divergent instability mode that does not exist with configuration A. This is due to the additional integration provided by the open-loop translator.

Steady-state limit cycle characteristics will be similar to those of the previous case. Behavior of attitude variables will again be generally as depicted in Figure 5-28. In this case, however, time between motor steps (T) is determined from the relation

$$\frac{1}{T} = \frac{\delta_{\max} C^1}{K_2} \left[\tau_1^2 - \frac{T^2}{12} \right]$$

With $\delta_{\max} = 0.003$ inch, and the system parameter values listed above, this yields $T = 170$ seconds as the theoretical time between successive motor steps. Again, system noise and effects of other imperfections can be expected to reduce this somewhat.

Comparison of Translator Attitude Control Alternatives

Preliminary design analysis of two alternate systems utilizing ion engine translation for attitude control has been described in the preceding sections. No claim is made that these designs are optimum in any sense. Rather, they are representative for a specific spacecraft application (Titan III M/Mars orbiter), and provide bases from which quantitative evaluation of the following aspects of attitude control system performance can be made and compared.

- Stability and dynamic performance
- Steady-state attitude control accuracy
- Interaction with solar panel structure
- Control of disturbance torques
- Number of motor cycles required
- Reliability

Each of these items is discussed in the following paragraphs. Extrapolations of these results to other spacecraft applications (Jupiter flyby, out-of-ecliptic) are made where appropriate. A summary of the comparative data is presented in Table 5-19.

Stability and Dynamic Performance. Comparative data on stability and dynamic performance parameters for configurations A (closed-loop translator) and B (open-loop translator) are listed in Table 5-19. It can be seen that the values listed are satisfactory in each case. System damping is roughly the same in both cases. The only significant difference is that configuration B yields a low-gain divergent instability mode, whereas A does not. However, this is not considered important since an adequate gain margin of 5 exists.

Steady-State Attitude Error. Satisfactory in both cases. Configuration A yields less than 1 degree steady-state error, whereas B yields zero due to infinite dc gain of the open-loop translator. This is true for any disturbance torque level within the control capability of the ± 5 inch translator.

Interaction with Solar Panel Structure. Attitude control system interaction with solar panel structural resonance has long been recognized as a fundamental problem area in the solar-powered, electric propulsion class of spacecraft. With respect to this important comparative, configuration B enjoys a distinct advantage.

TABLE 5-19. SUMMARY OF COMPARATIVE DATA FOR TRANSLATOR ATTITUDE CONTROL ALTERNATIVES

Comparatives	Titan III M/Mars Orbiter Spacecraft		Jupiter Flyby Out-of-Ecliptic Vehicles
	Closed-Loop Translator A	Open-Loop Translator B	
Stability and dynamic performance			
Gain margin:			
Gain increase	25	8	8
Gain decrease	NA	5	5
Phase margin	40 degrees	40 degrees	40 degrees
Damping ratio	0.425	0.4	0.4
Steady-state attitude error	<1 degree	0	0
Control of disturbances:			
Control bandwidth	$\omega < 0.2$ rad/sec	$\omega < 0.13$ rad/sec	$\omega < 0.13$ rad/sec
Interaction with structure:			
Loop gain at 0.2 rad/sec	0.043	0.004	0.004
Steady-state limit cycle:			
Time between motor steps	43 seconds	170 seconds	170 seconds

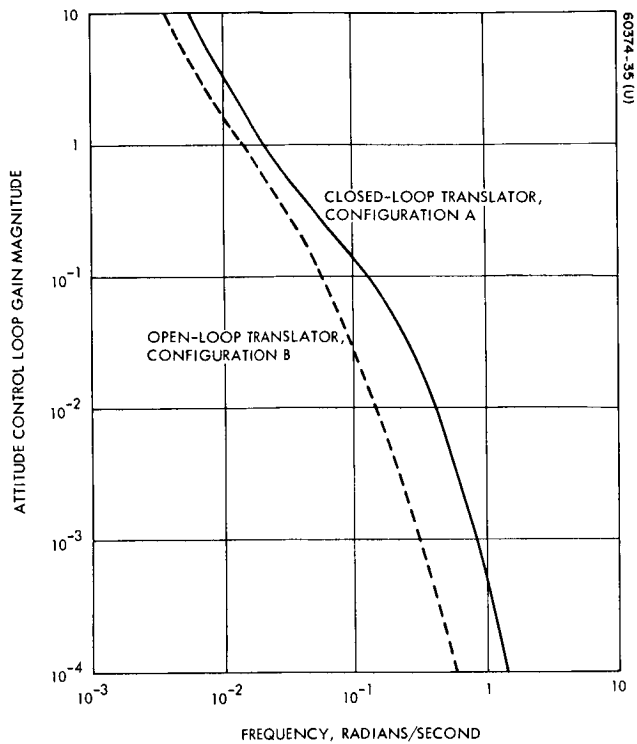


Figure 5-29. Attitude Control Loop Gain Magnitude Versus Frequency

Plots of system loop gain versus frequency for configurations A and B are shown in Figure 5-29. These curves are used to indicate the structural gain at resonance required to cause instability with each of the systems. Since present estimates indicate that solar panel structural resonance occurs in the frequency region 0.1 rad/sec to 1.0 rad/sec, a resonance frequency of 0.2 rad/sec will be assumed for comparison purposes. Thus, with configuration B, since the loop gain at 0.2 rad/sec is 0.004, a structural resonance gain of 250 or more is required to cause regenerative interaction. In the case of configuration A a structural gain of only 23 is required. Configuration B is therefore less susceptible to structural interaction at 0.2 rad/sec by a factor of roughly 10.

The question of whether or not structural resonance yields a gain in excess of the critical value (e. g., 250 at 0.2 rad/sec) cannot be conclusively answered without a detailed analysis of solar panel bending modes and coupling with the main body. However, it is felt that configuration B affords ample margin against the possibility of regenerative interaction. And, further parameter optimization can increase the safety margin over that indicated in Figure 5-29. The practical limit in this respect will be the size of the shaping lead terms.

Control of Disturbance Torques. This aspect of system performance is satisfactory in both cases. Attitude control system response to disturbance torque (L_d) is determined from the response function

$$\frac{\theta}{L_d} = \frac{1}{1 + Y(s)} \quad (s = j\omega)$$

where θ denotes attitude angle, and $Y(s)$ the loop transfer function, which is plotted versus frequency for both configurations in Figure 5-29. For $|Y(j\omega)| \ll 1$ the response becomes that of the body itself, as though no control system were present. As a rule of thumb, control is exercised over the

frequency range where $|Y(j\omega)| > 1$ holds. Thus, from Figure 5-29 control of disturbance torques is exercised for frequencies up to 0.2 rad/sec and 0.13 rad/sec, respectively, for configurations A and B. Since the principal disturbance torques, due to relative movement of cg and thrust vector, are expected to occur at frequencies substantially lower than this, either system is considered satisfactory from this standpoint.

Motor Limit-Cycling Rate. The time between successive motor steps during steady-state limit cycle operation is listed in Table 5-19 for each configuration. Although the values listed were determined analytically with no allowance made for the effect of system noise, effective filtering will reduce this to only a minor influence. The conclusion from Table 5-19 is that configuration A will yield a higher limit-cycle stepping rate (theoretically 4 times as high) than B. Since wear is a critical problem for long duration missions, configuration B is clearly preferable in this respect.

It is of interest to estimate the total number of motor steps required during the course of a mission. Since the system will operate in the limit cycle mode (one step per 170 seconds, theoretically) during most of the mission, the average cycling rate can conservatively be estimated as 1 step per 10 seconds. It can be noted that only 1700 steps are required for stop-to-stop translation (10 inches of travel). At this average rate, 5×10^6 steps are required for a mission of 500 days duration. This should pose no wear-out problem for the stepper motor since manufacturers' data indicates capability of 10^9 steps. Also, tests of Surveyor stepper motors have repeatedly demonstrated satisfactory performance for more than 10^7 total cycles.

Reliability. The reliability of the ion engine translation mechanism is recognized as a major factor in establishing overall spacecraft reliability, and an area where additional, more detailed studies seem warranted.

Selection of the stepper motor approach maximizes translator reliability with respect to other actuator alternatives. From the comparative analysis given above, configuration B maximizes reliability with respect to the two system alternates considered. This follows from two observations. First, fewer motor cycles (by a factor of 4) are required over the course of a mission, and second, a position indicating device is not required for system operation. (Such a device could be included for ground monitoring of translator position, but would not be used for onboard control purposes.)

The best data available on stepper motor reliability indicates a failure rate of 0.1 percent per 1000 hours of continuous operation. This yields a reliability factor 0.988 for continuous operation over a 500 day mission time. However, experience with stepper motors at Hughes indicates that the actual reliability may be somewhat higher than this.

Stepper motors used on the Surveyor program are very similar to the type proposed for this application. These are of the magnetic detent type, manufactured by IMC and Kearfott. Tests have repeatedly demonstrated satisfactory operation under Surveyor transit phase environmental conditions

(hard vacuum, -75°F to 200°F temperature) for greater than 10^7 motor cycles. Motors on Surveyor I operated satisfactorily for a period of 28 days on the moon, with the temperature between operating periods going as low as -250°F . In addition, Hughes' experience has been that not a single motor has failed once it has passed initial inspection.

Conclusions. It is concluded from this comparative analysis that configuration B, involving open-loop translator operation within the attitude control loop, is preferred over configuration A. The following major advantages of configuration B explain this preference.

- 1) Less susceptible, by a factor of approximately 10, to regenerative interaction with solar panel structure.
- 2) Fewer motor steps, by a factor of approximately 4, required during the course of a mission.
- 3) Greater reliability.
- 4) Zero steady-state attitude error.

It is further concluded from this analysis that the two-axis attitude control function is performed satisfactorily by ion engine translation alone. That is, control augmentation by other means (reaction jets, momentum storage wheels) is not required.

Preliminary Investigation of the Desirability of Momentum Exchange for Transit Phase Attitude Control

There is little doubt that the transit phase attitude control function, about any or all spacecraft axes, can be satisfactorily performed by means of a momentum storage device in conjunction with ion engine translation and reaction jet for momentum unloading. Furthermore, a variety of momentum storage devices exist from which to choose. These include

- Reaction wheels
- Reaction sphere
- Fluid circulators
- Control moment gyros

Each of these devices utilizes a rotating mass to store angular momentum. Of those listed, only reaction wheels and control moment gyros have reached the stage of space flight qualification and would be considered eligible for 1970 electric propulsion spacecraft applications.

However, given that a momentum exchange device could satisfactorily perform the control task, the question remains as to the necessity or even desirability of employing such a device. As an aid to establishing an answer to this question, Figure 5-30 shows a functional block diagram of a possible single-axis attitude control system involving momentum storage in conjunction with ion engine translation. A reaction wheel is chosen since it does not require consistent, high wheel speed as does the control moment gyro, nor are control torques applied through the wheel bearings. These considerations imply that a reaction wheel is preferred for long-duration missions. An ac servomotor is selected for its reliability advantage over dc motors, and a linear speed-torque motor characteristic is tentatively chosen, although a speed-insensitive torque characteristic could also be used.

As indicated in Figure 5-30, the attitude control system utilizes two parallel channels — one relatively fast-acting channel involving the reaction wheel, and one slower channel employing a stepper motor ion engine translation device. Thus, disturbance torques are continuously counteracted by the wheel, while the translation device continually acts to reduce wheel speed to near zero. The necessity of engine translation is again apparent with this system, since total momentum over a mission could not be stored with reasonable wheel size and speed, and wheel unloading via reaction jets would impose an equally prohibitive weight penalty.

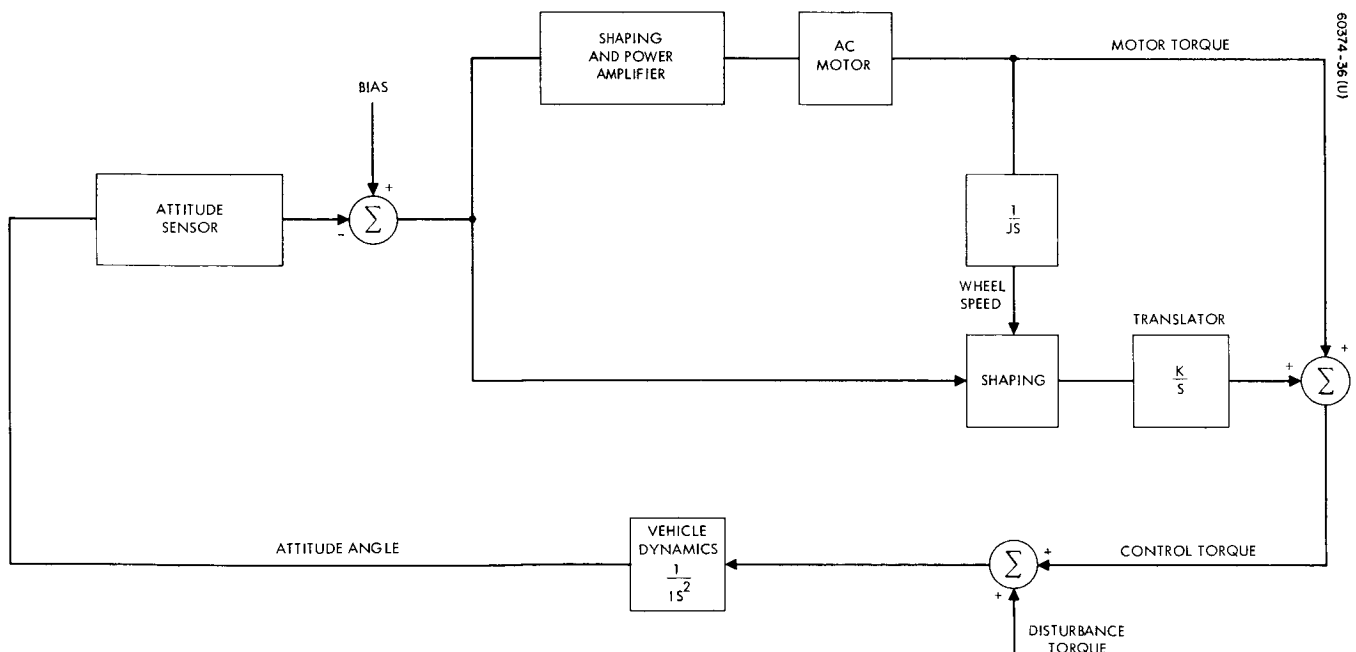


Figure 5-30. Attitude Control Configuration Utilizing Momentum Exchange

The question then arises, since continuously operating engine translation is required to unload the momentum storage channel, and since translation alone can satisfactorily handle the entire control function, what benefit is gained by adding momentum storage. It could be presumed that faster control response is obtainable with a reaction wheel. However, it was shown in Section 3 that adequate response is obtained with the proposed engine translator, and moreover, fast response is directly adverse to the requirement for minimum susceptibility to regenerative interaction with the solar panel structure. Thus it appears that no functional advantage is gained, and the added system complexity would degrade overall system reliability.

It is concluded, therefore, that momentum storage is neither necessary nor desirable for use in conjunction with ion engine translation. Application to control about the third axis (i. e., about the thrust vector) also seems neither necessary nor desirable since disturbance torques will generally be unidirectional and require that the full torque impulse ultimately be developed by the gas jet system.

TELECOMMUNICATIONS

Mission-Related Criteria

The antenna of an interplanetary probe must point to earth. The required angular coverage will determine the choice of a suitable antenna, control performance of the telecommunication link, and affect the vehicle configuration. It is therefore important to have knowledge of the expected earth track; the definition given in Figure 1A is in accord with JPL's usage (EPD-139, Volume III, Figure 5-31). Earth track expressed in cone/clock coordinates (conic projection) is convenient for an attitude-stabilized vehicle, and moreover gives directly the earth's angular proximity to or occultation by, the sun. Figure 5-32 extends the coordinate definition for the case where earth view occurs on the shadow side of the spacecraft.

For the various missions, different size boosters and solar arrays are appropriate, and it is a combination of these factors that determines the antenna size. Specific telecom performance has thus been tied to specific shroud diameters with the 20-foot diameter shroud being representative of the Saturn 1B/Centaur configuration (or one representing a similar shroud geometry), and the 10-foot diameter shroud representative of the Titan III M and Atlas (SLV-3C)/Centaur configurations.

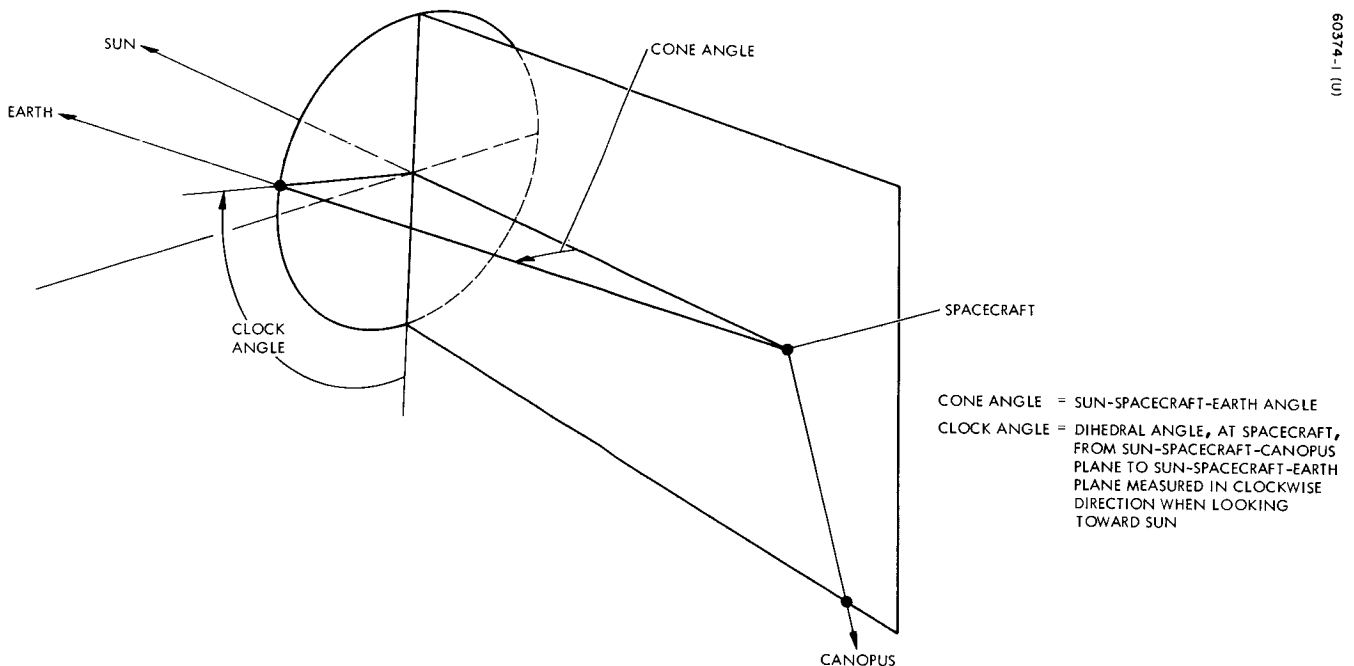
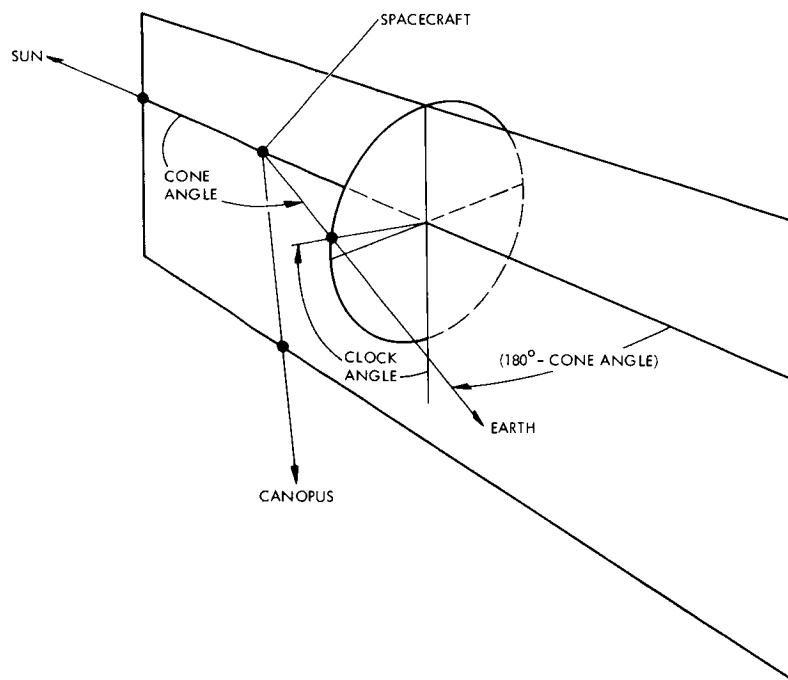


Figure 5-31. Earth View Coordinate System for Sun-Canopus Oriented Spacecraft, Sun Side



NOTE THAT CONE/CLOCK ANGLE DEFINITIONS REMAIN UNCHANGED, EXCEPT THAT POLAR COORDINATE PROJECTION NOW SHOWS COMPLEMENT OF CONE ANGLE ($180^\circ - \text{CONE ANGLE}$), WITH CONE ANGLE THEREFORE DEPICTED AS INCREASING TOWARDS ORIGIN (WHERE CONE ANGLE = 180°)

Figure 5-32. Earth View Coordinate System for Sun-Canopus Oriented Spacecraft, Shadow Side

Mars Missions

Earth View Angle. Typical earth tracks, computed from spacecraft ephemeris data, are given in Figures 5-33, 34, 35, 36 and 37. For the cases shown, it would be convenient to switch to the high-gain antenna about 150 days after leaving the earth's sphere of influence. From then on, the total pointing variation is less than 32 degrees in cone angle and less than 20 degrees in clock angle, indicating that only a limited pointing coverage is needed for the high-gain antenna.

Telecom Range. Plots of telecom range, computed from spacecraft ephemeris data, are given in Figures 5-38, 39, 40, 41, and 42.

Antenna Configuration. For a SEP spacecraft with a 21-foot diameter shroud, the antennas and their coverage are given in Figure 5-43.

Summary of Link Performance. Bit rates shown are for a maximum range of 350×10^6 km. Note that for some missions, encounter occurs at considerably less than maximum range; in those cases the bit rate at encounter could be correspondingly increased. For instance, for the 1971 Mars mission, encounter occurs at roughly half the maximum range; hence, the bit rate at encounter could be nearly quadrupled, provided that the added telecom mode is deemed worthwhile. Note also that the video function is needed only for the postencounter phase.

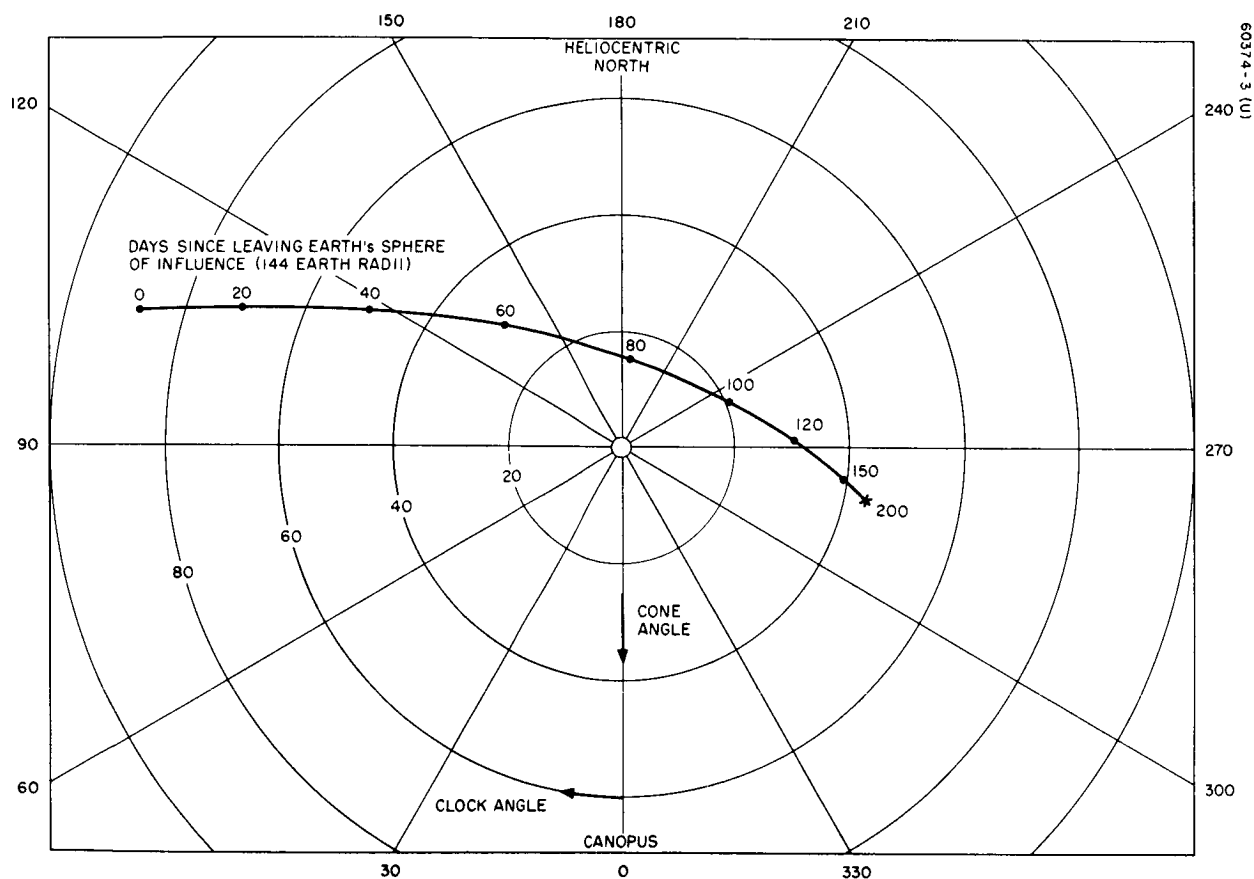


Figure 5-33. Cone/Clock Coordinates, as Viewed from Spacecraft,
for 1971 Launch to Mars, Encounter = $x = \text{JD } 41,310$
(4 January 1972) Typical Earth Track Versus Time

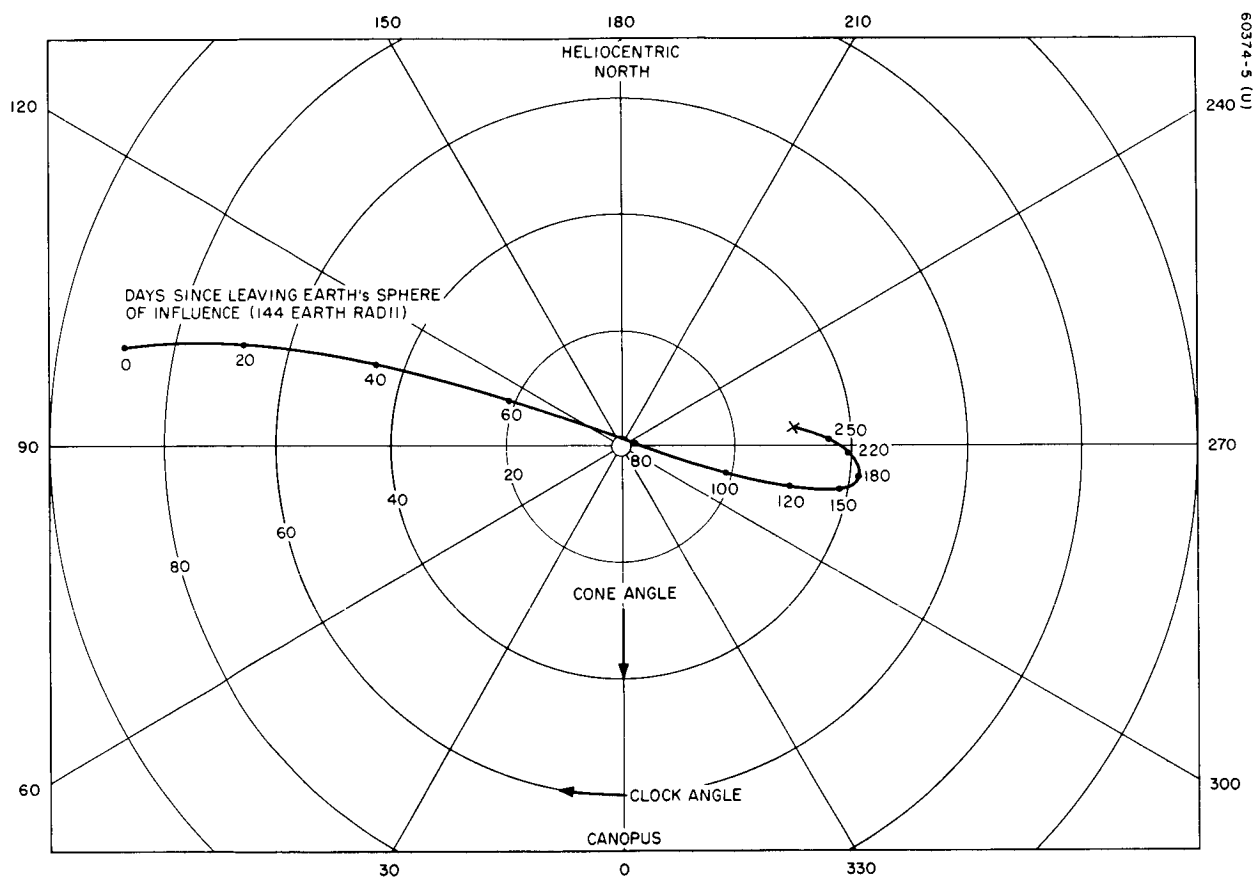


Figure 5-34. Cone/Clock Coordinates, as Viewed from Spacecraft,
for 1973 Launch to Mars, Encounter = x = JD 42, 172
(5 May 1974) Typical Earth Track Versus Time

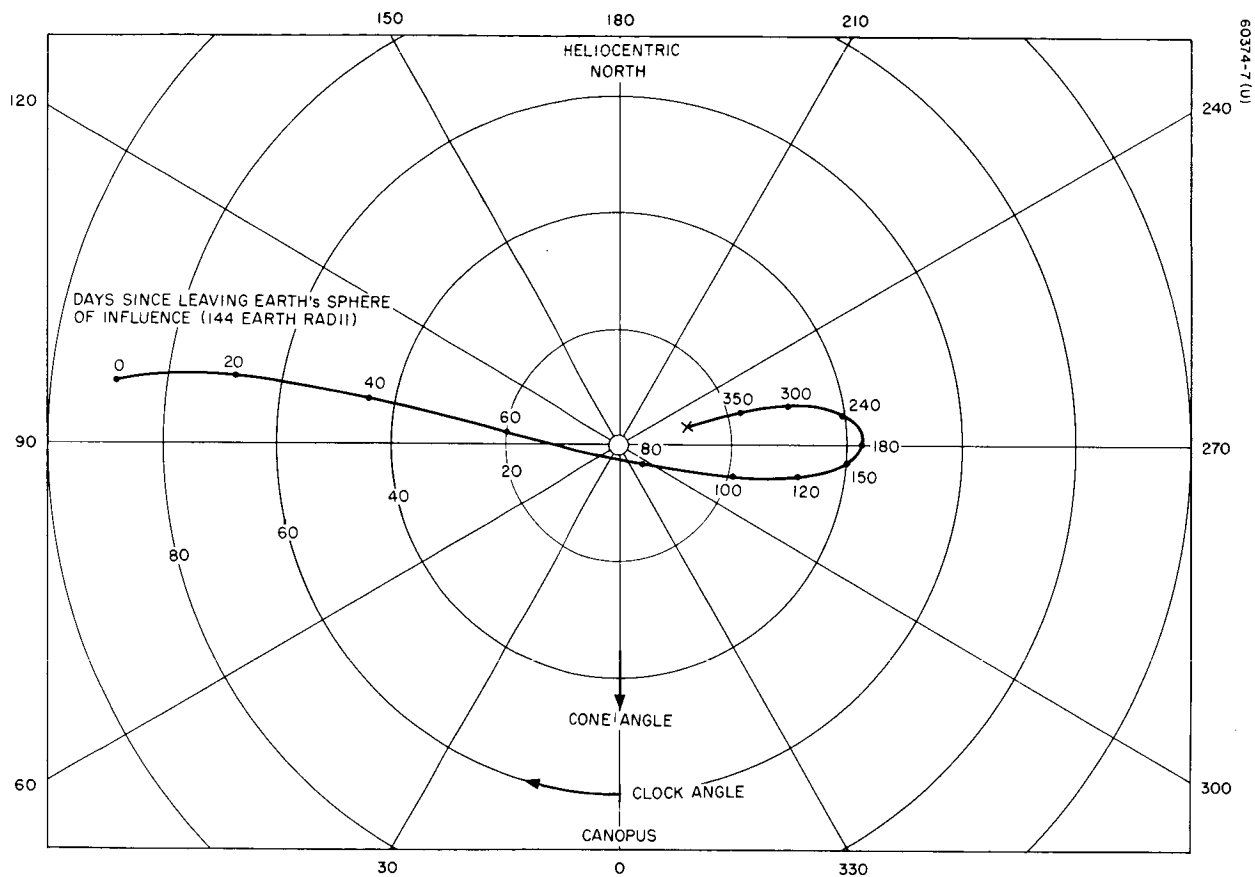


Figure 5-35. Cone/Clock Coordinates, as Viewed from Spacecraft,
for 1975 Launch to Mars, Encounter = $x = \text{JD } 43,046.7$
(25 September 1976) Typical Earth Track Versus Time

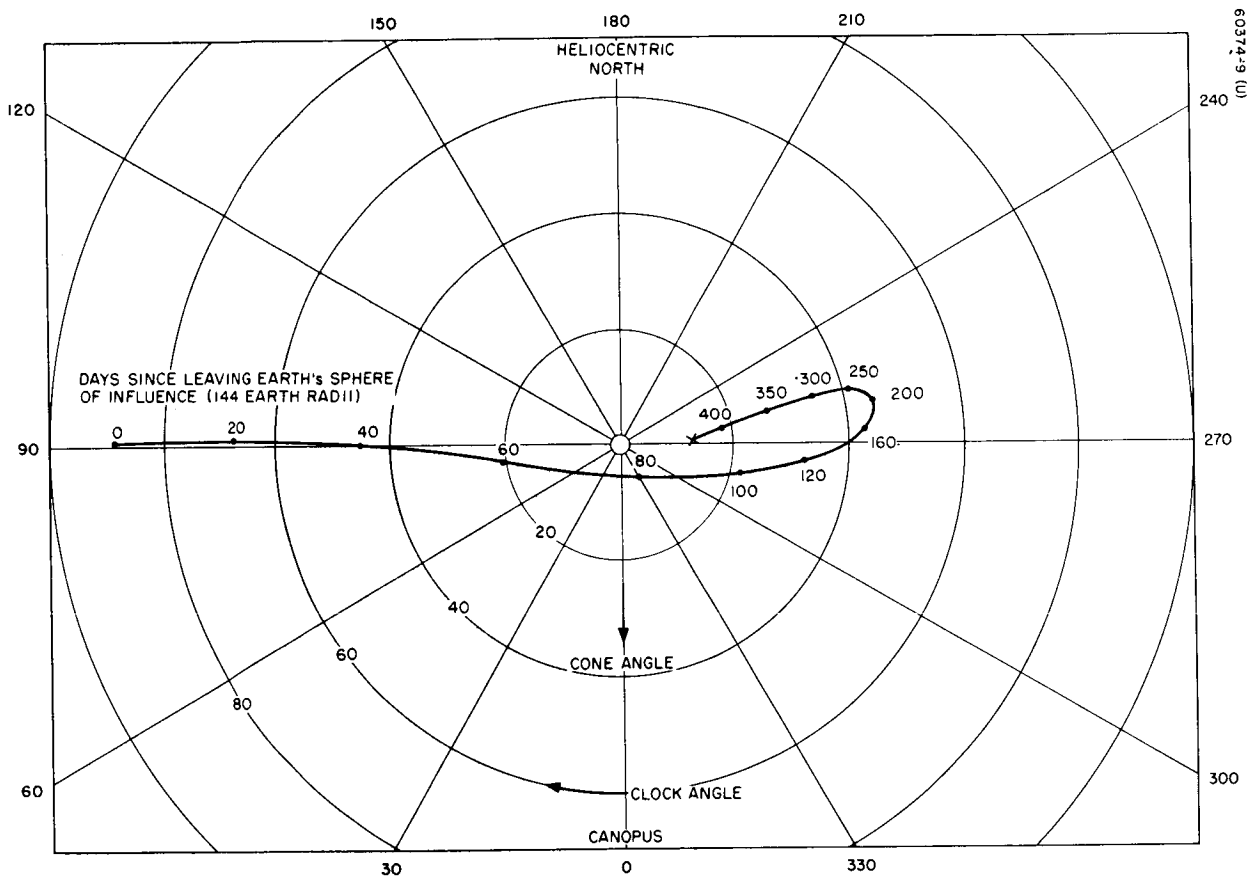


Figure 5-36. Cone/Clock Coordinates, as Viewed from Spacecraft,
for 1977 Launch to Mars, Encounter = x = JD 43,823.6
(11 November 1978) Typical Earth Track Versus Time

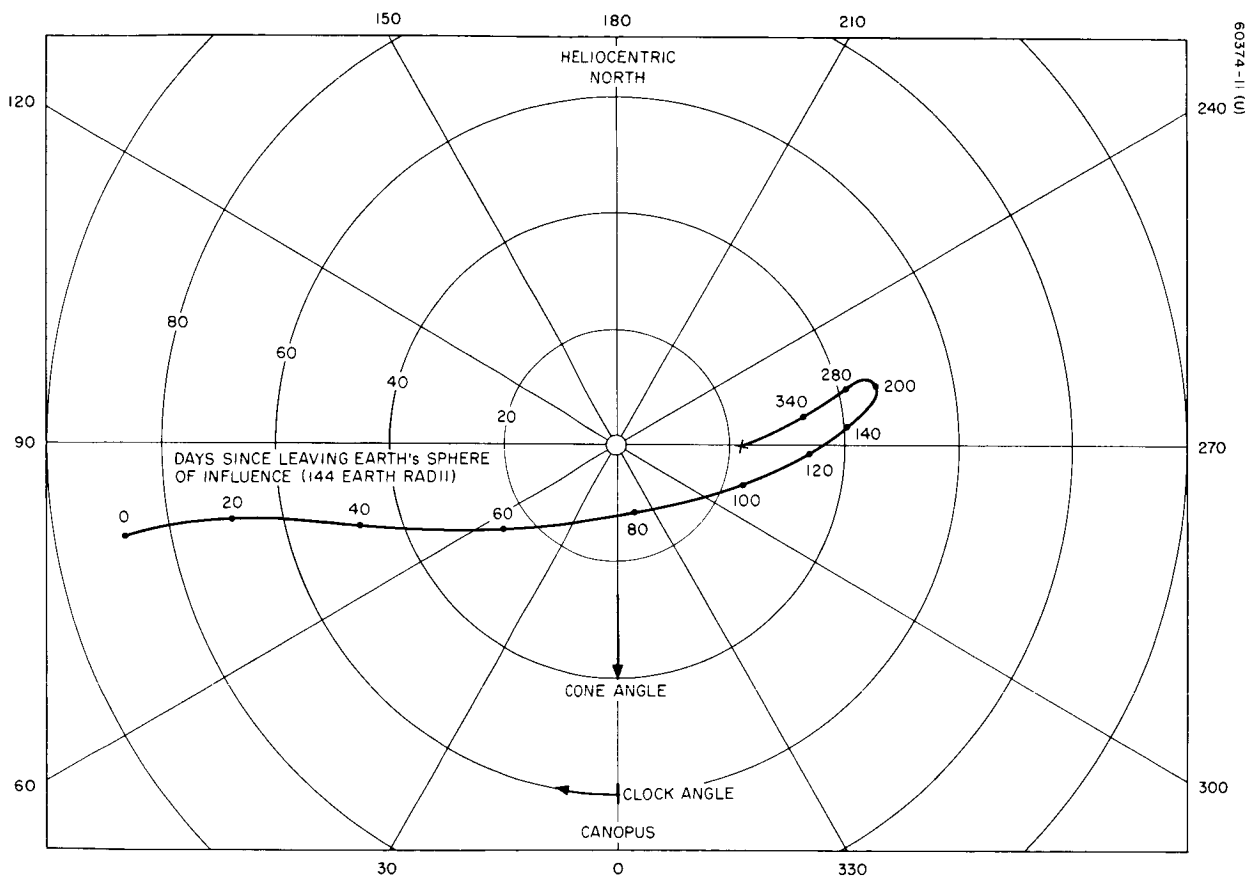


Figure 5-37. Cone/Clock Coordinates, as Viewed from Spacecraft,
for 1979 Launch to Mars, Encounter = x = JD 44,553.4
(10 November 1980) Typical Earth Track Versus Time

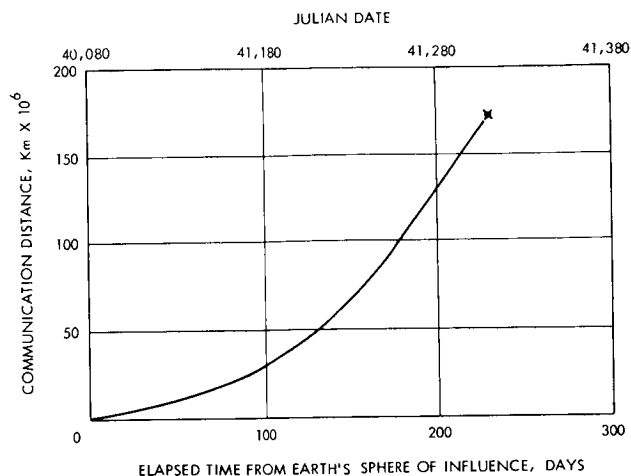


Figure 5-38. Communication Distance Versus Time, for 1971 Launch to Mars, Encounter = $x = \text{JD } 41,310.7$ (4 January 1972)

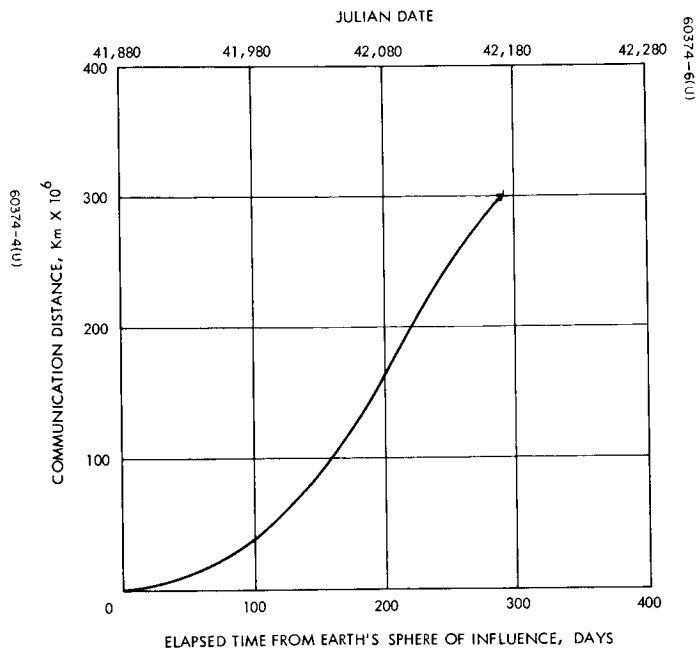


Figure 5-39. Communication Distance Versus Time, for 1973 Launch to Mars, Encounter = $x = \text{JD } 42,172$ (5 May 1974)

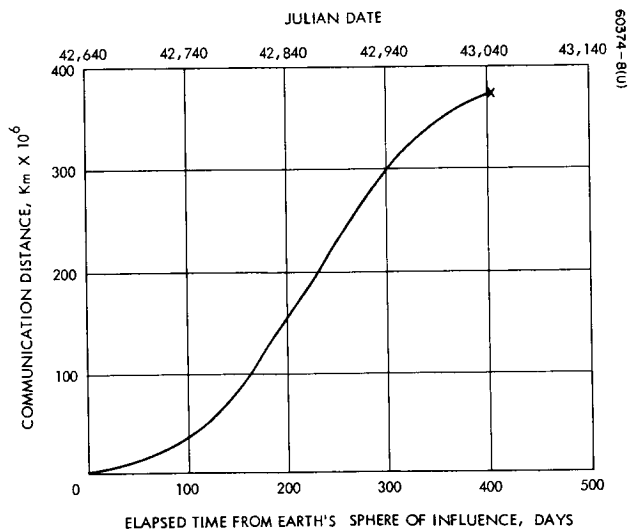


Figure 5-40. Communication Distance Versus Time, for 1975 Launch to Mars, Encounter = $x = \text{JD } 43,046$ (25 September 1976)

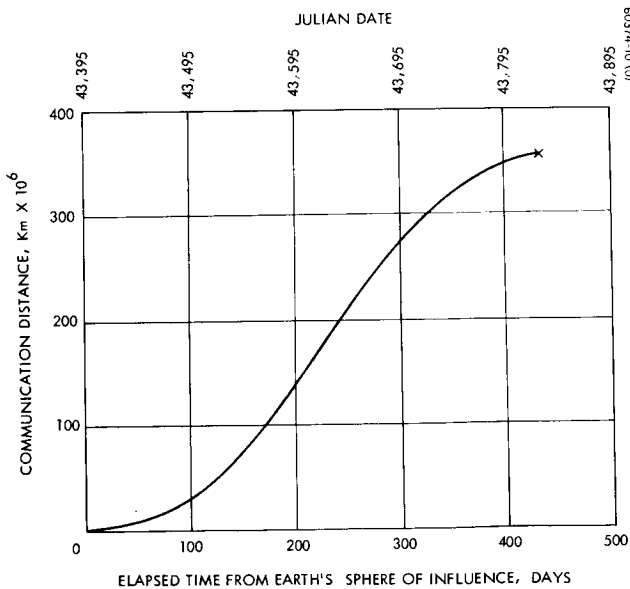


Figure 5-41. Communication Distance Versus Time, for 1977 Launch to Mars, Encounter = $x = \text{JD } 43,823.6$ (11 November 1978)

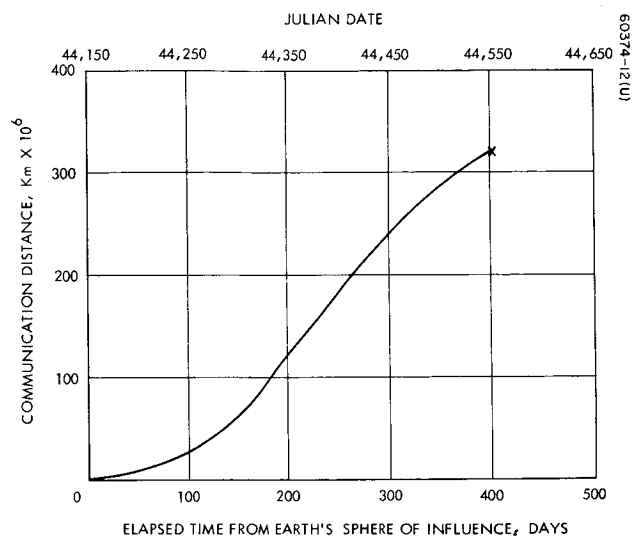


Figure 5-42. Communication Distance Versus Time, for 1979 Launch to Mars, Encounter = x = JD 44,554.4 (10 November 1980)

Downlink performance is summarized by the following parameters:

Frequency	2295 mc
Receiving antenna gain (210-foot antenna)	61 db
Effective temperature of receiving system	55° ± 10°K
Transmitter power	50 watts
Transmitting antenna gain (7 by 7 foot planar array)	33 db
Bit rate at maximum range (350 · 10 ⁶ km)	1500 bps
Transmission time for 500-line TV picture	≈8 min/picture

Bit rate at shorter ranges can be increased if desired.

Hemispherical coverage antenna cannot be used for downlink telemetry beyond 175 · 10⁶ km range (at 2 bps).

Data bursts occurring at rates above 1500 bps may be recorded and transmitted when convenient.

Video capability and tape storage for a Mars mission are discussed under "Subsystem Functional Description".

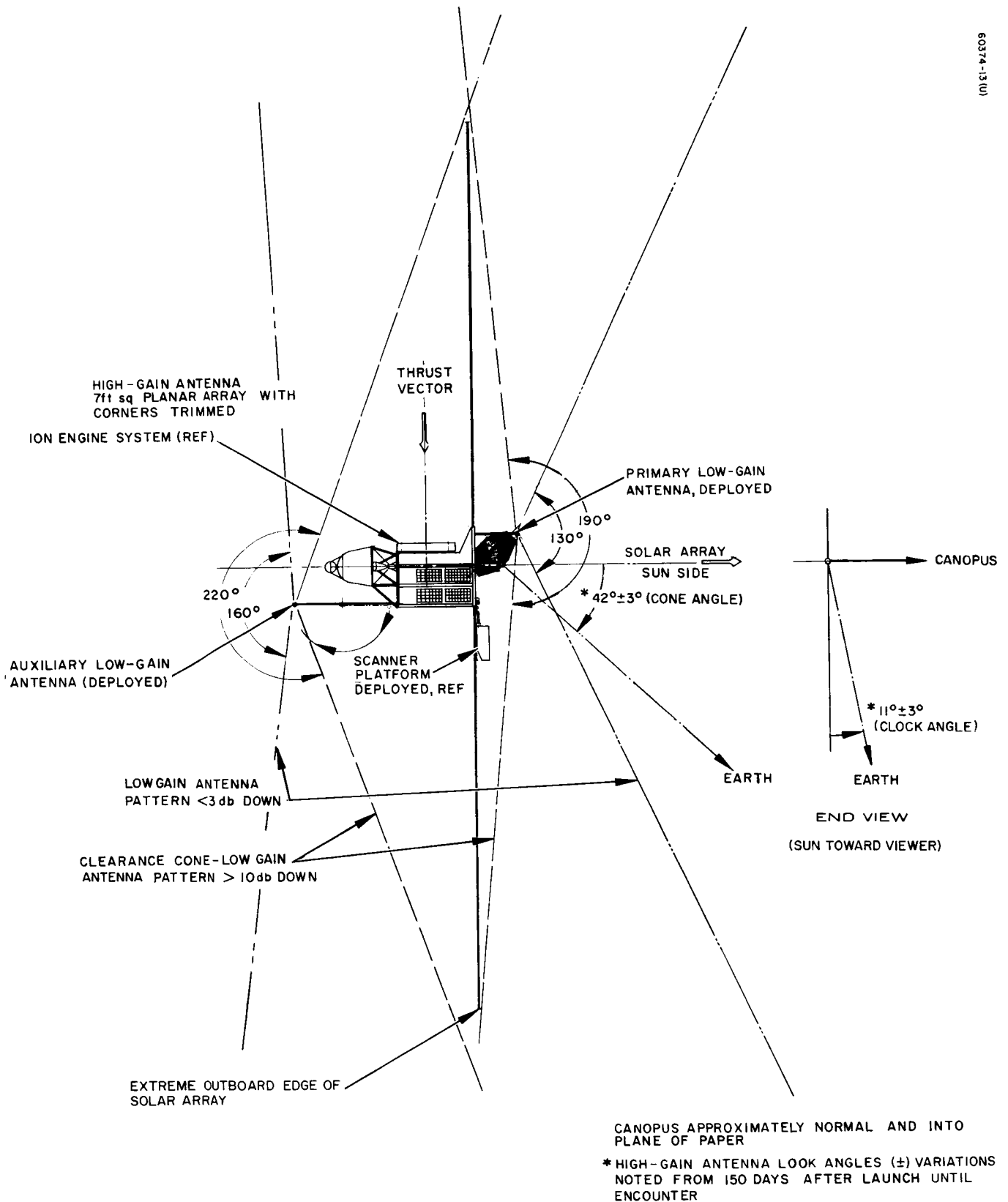


Figure 5-43. Saturn IB/Centaur SEP Spacecraft Antenna Coverage

Uplink performance is characterized by the following parameters:

Frequency	2115 mc
Receiving antenna gain (hemispherical coverage)	0 db
Effective temperature of receiving system (TDA)	950°K
Transmitter power	100 kw
Transmitting antenna gain (85-foot dish)	51 db
Bit rate	2 bps

Occultations

Solar Occultation at Planetary Opposition

Uplink. At Mars the sun subtends a view angle of about 1/3 degree. For a spacecraft antenna having a beamwidth of 4 degrees, the corresponding solid angle ratio is 144, which means that for a noncryogenic receiver at the spacecraft, the sun's intrusion into the spacecraft antenna beam will not degrade the uplink, unless there is true occultation.

Downlink. For the 210-foot dish (DSIF) having a beamwidth of about 0.14 degree, and using a cryogenic receiver, the sun's intrusion into the ground antenna beam will (briefly) disable the downlink.

Planetary Occultation. Will occur for a spacecraft orbiting around Mars.

Telecom Modifications for 10-foot Shroud SEP Configuration (Atlas (SLV-3C)/Centaur), Mars Mission

Available Power. Available raw power will be limited to about 475 watts. This will affect the telecom subsystem only slightly, but will force a cutback in power available for instrumentation.

High-Gain Antenna. A small parabolic reflector is proposed with characteristics as follows:

Size and type	3.5-foot diameter parabolic reflector
Polarization	RH circular
Gain	25.5 db
3-db beamwidth	8.5 degrees
Off-axis loss at ± 1.4 degrees	<0.2 db

Because of the ample stowage volume available, use of a parabolic reflector should present no space problems.

Downlink. The characteristics are the same as discussed except:

Bit rate at maximum range 400 bps
($3.50 \cdot 10^6$ km)

Transmission time for 500-line ≈ 31 minutes/picture
TV picture

Bit rate at shorter ranges can be
increased if desired

Data bursts occurring at rates
above 400 bps may be recorded,
and transmitted when convenient.

Jupiter Missions

Earth View Angle. The trajectory is first interior to that of the earth, then becomes exterior. Thus, initially the earth appears on the vehicle's shadow side (Figure 5-44), then after 127 days it moves to the vehicle's sun side (Figure 5-45) and remains there until encounter. After 200 days, the earth track becomes confined to a small line segment within which it oscillates with ± 27 degrees east-west amplitude, with the line segment roughly coincident with the earth's ecliptic plane.

Telecom Range. Communication distance versus time is given in Figure 5-46. It should be noted that the round-trip propagation time at maximum range will be 1 hour and 40 minutes; thus a considerable amount of on-board sequencing, computing, and decision-making may be necessary.

Antenna Configuration. For a SEP spacecraft with a 10-foot diameter shroud, the antennas and their coverage are shown in Figure 5-47. Selection and performance of these antennas for the Jupiter mission are discussed below.

Initially, omnidirectional coverage is obtained with two quasi-hemispherical coverage (low-gain) antennas, with switch-selection provided between antennas. This dual antenna arrangement will cover the initial near-earth phase (with the earth appearing on the vehicle's shadow side), as well as the transition of earth view to the vehicle's sun side.

After 200 days, coverage via the medium-gain antenna becomes practical, and after 275 days the low-gain antenna range for downlink telemetry is exceeded; thus antenna switchover should be accomplished in the 200 to 275-day interval.

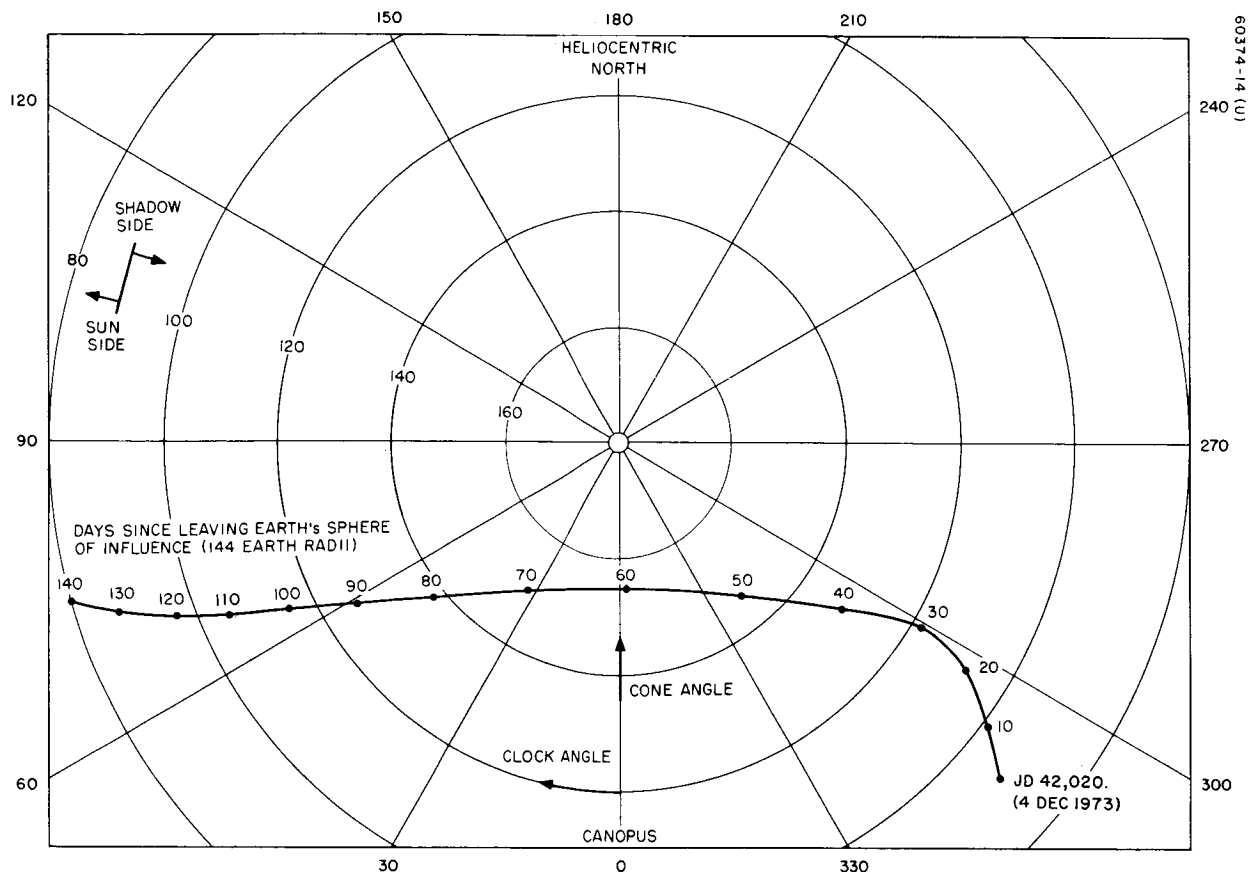


Figure 5-44. Typical Earth Track Versus Time, 900-Day Jupiter Flyby Mission, Cone/Clock Coordinates, as Viewed from Spacecraft, Shadow Side

The medium-gain antenna is a reticulated horn covering approximately 60 degrees beamwidth. While this is far from omnidirectional coverage, it requires only rather coarse pointing performance. A gain of 9 db provides range improvement by a factor of $\times 2.8$ over low-gain antenna coverage.

The medium-gain antenna is located on the back of the feed for the high-gain antenna, and is initially used with the latter's pointing mechanism centered (Figure 5-47).

After 450 days the medium-gain antenna becomes unusable for down-link telemetry, thus the downlink should be switched to the high-gain antenna before that time. The uplink command function continues via the medium-gain antenna until encounter; the command function is thus not critically dependent on accurate antenna pointing performance.

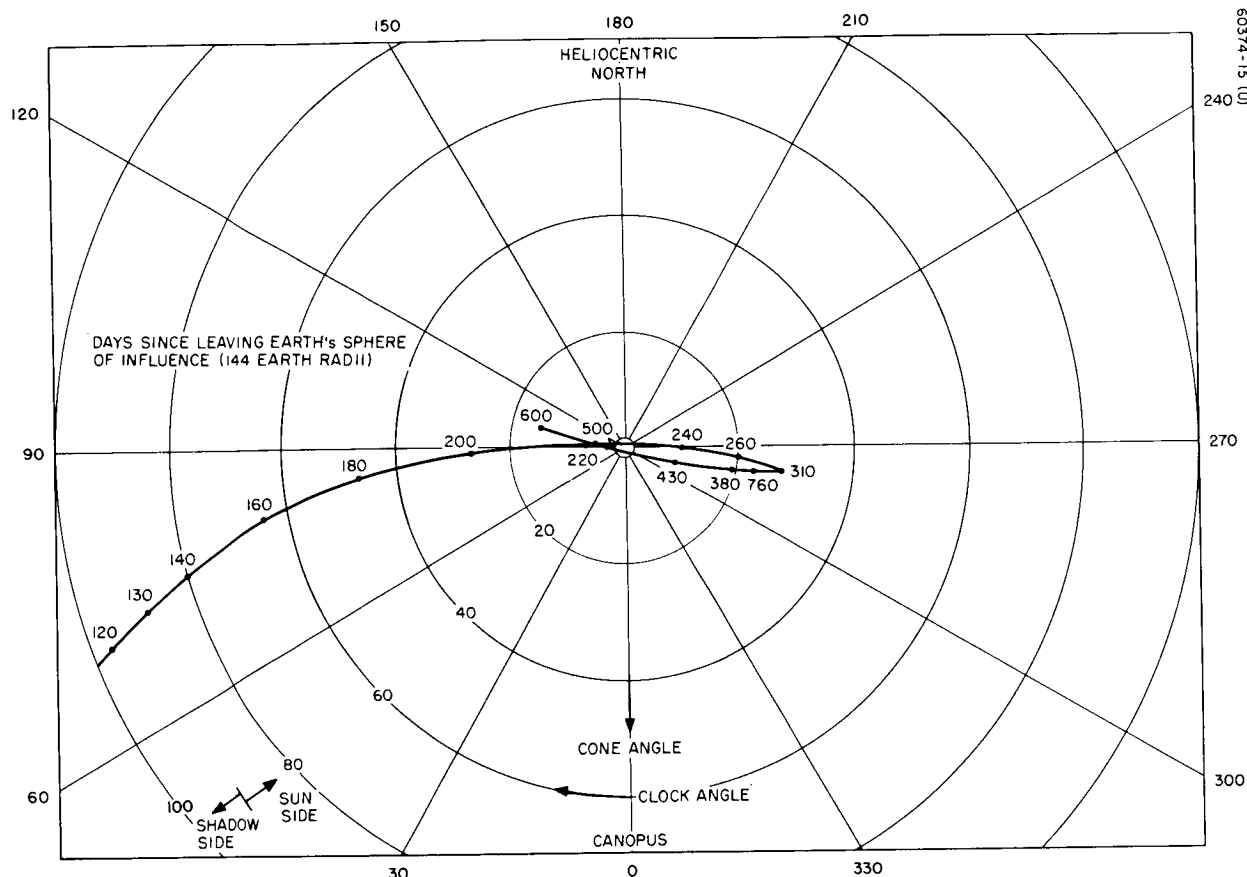


Figure 5-45. Typical Earth Track Versus Time, 900-Day Jupiter Flyby Mission, Cone/Clock Coordinates, as Viewed From Spacecraft, Sun Side, Encounter = $x = \text{JD } 42,921.5$ (23 May 1976)

The low-gain antenna (sun side) may intrude into the pattern of the high-gain antenna when the high-gain antenna is pointed to the right of the sunline. This inherent conflict in antenna location (Figure 5-47) can be resolved by redeploying or jettisoning the low-gain antenna (sun side) sometime prior to transitioning the downlink from the medium-gain antenna to the high-gain antenna.

The high-gain antenna is a 6 foot paraboloid, for which there is ample stowage volume on the spacecraft. Antenna gain is 30 db; half-power beamwidth is 5 degrees. The antenna can be pointed in two degrees of freedom over a ± 30 degree cone with respect to spacecraft centerline.

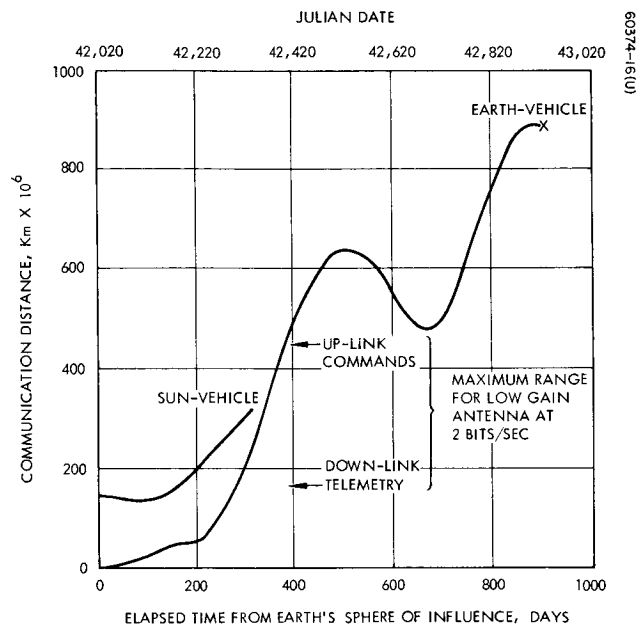


Figure 5-46. Communication Distance Versus Time, for 900-Day Jupiter Flyby, Encounter = $x = \text{JD } 42,921.5$ (23 May 1976)

Summary of Link Performance. Three antenna systems are indicated:

Low-gain, at 0 db	Isotropic coverage via two hemispherical-coverage antennas, with switch-selection between them
Medium-gain, at 9 db	60 degrees beamwidth
High-gain, at 30 db	5 degrees beamwidth

Downlink performance from Jupiter (path = 6 AU) is summarized by the following parameters:

Frequency	2295 mc
Receiving antenna gain (210-foot antenna)	61 db
Effective temperature of receiving system	$55^\circ \pm 10^\circ \text{K}$

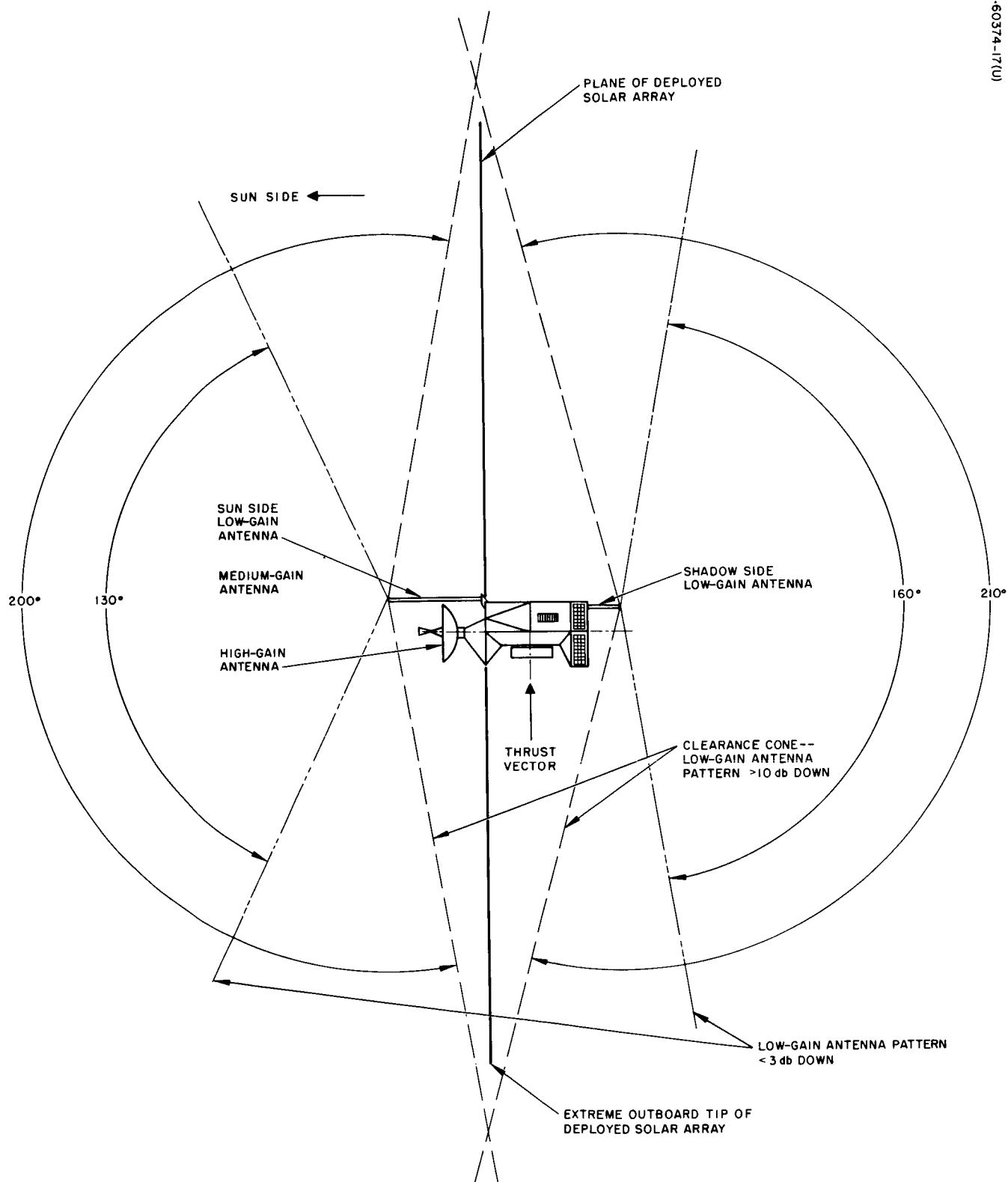


Figure 5-47. Atlas (SLV-3C)/Centaur SEP Spacecraft
Low Gain Antenna Coverage for Jupiter Mission

Transmitter power 50 watts

Transmitting antenna gain 30 db
(high-gain antenna)

Bit rate 100 bps

Uplink performance to Jupiter is characterized by the following parameters:

Frequency 2115 mc

Receiving antenna gain 9 db
(medium-gain antenna;
60 degree beamwidth)

Effective temperature of 950°K
receiving system (TDA)

Transmitter power 100 kw

Transmitting antenna gain 51 db
(85 foot dish)

Bit rate 2 bps

Video Capability and Tape Storage. For 500 x 500 line resolution, 250,000 picture elements will be needed. Gray-scale rendition subjectively equivalent to 64 brightness levels can be achieved with 3 bits/picture element, if means are provided to combat spurious contouring.* This results in a total of 750,000 bits/picture.

Nominal write-in rate = 12,000 bps at 12 ips tape speed; that is, 1 picture every 63 seconds.

Nominal read-out rate = 100 bps at 0.1 ips; that is, one picture may be transmitted every 2 hours. Tape speed ratio is then $120 \div 1$.

Occultations. As seen from earth, the spacecraft at encounter is only 3 degrees off the sunline, and is rapidly approaching solar occultation. Hence, not all data, TV-pictures, etc., may be brought back to earth before occultation, when a later re-play of the same data should be provided for.

Out-of-Ecliptic Probes (Figures 5-48 and 5-49)

Earth track is entirely confined within a cone angle range of 70 to 110 degrees and a clock angle oscillating between 15 and 160 degrees. The telecom range is close enough to permit operation of all links via a single

* L.G. Roberts, "Picture Coding Using Pseudo-Random Noise," IRE Transactions, IT, 8 February 1962, p. 145.

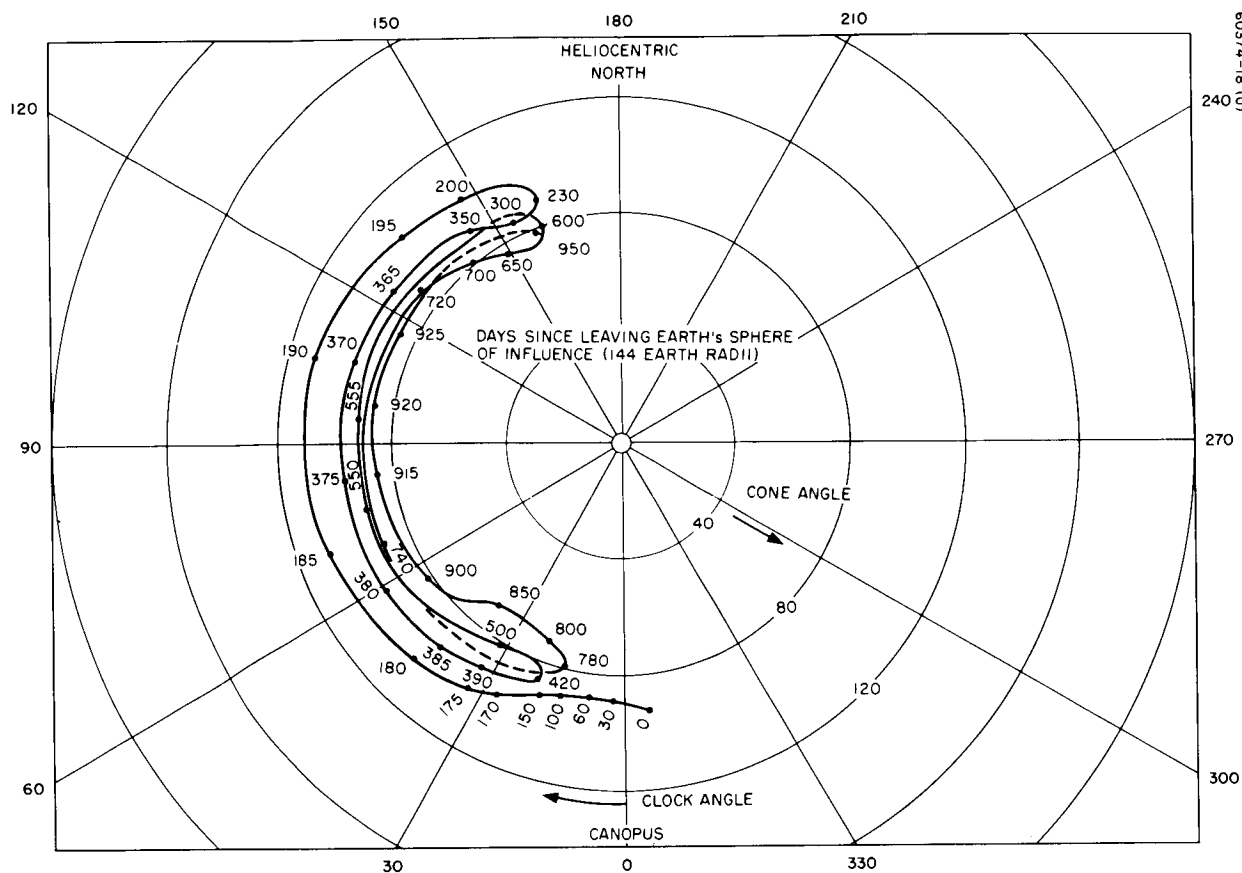


Figure 5-48. Typical Earth Track Versus Time,
Cone/Clock Coordinates, as Viewed from Spacecraft,
for Out-of-Ecliptic Probe
(Quasi-circular orbit, 1 AU radius, 1973 launch)

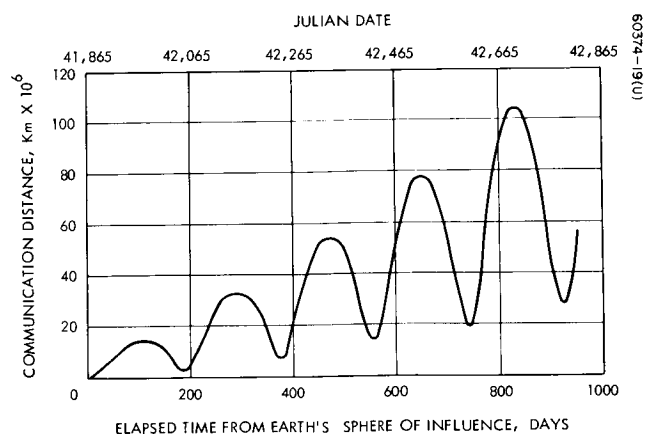


Figure 5-49. Communication Distance
Versus Time, for Out-of-Ecliptic Probe
(Quasi-circular orbit, 1 AU radius
1973 launch)

hemispherical-coverage antenna. (A second such antenna to cover the vehicle's shadow side is provided, but would be used only in case of temporary loss of attitude control.)

Venus Missions (Figures 5-50, 51, and 52)

Earth track is entirely confined to the vehicle's shadow side. The telecom range is close enough to permit operation of all links via a single hemispherical-coverage antenna.

Mercury Missions (Figures 5-53, 54 and 55)

Initially, the earth appears on the vehicle's shadow side. After 190 days the earth track moves to the sun side, and continues along its apparent great-circle path, until after 435 days it is back on the shadow

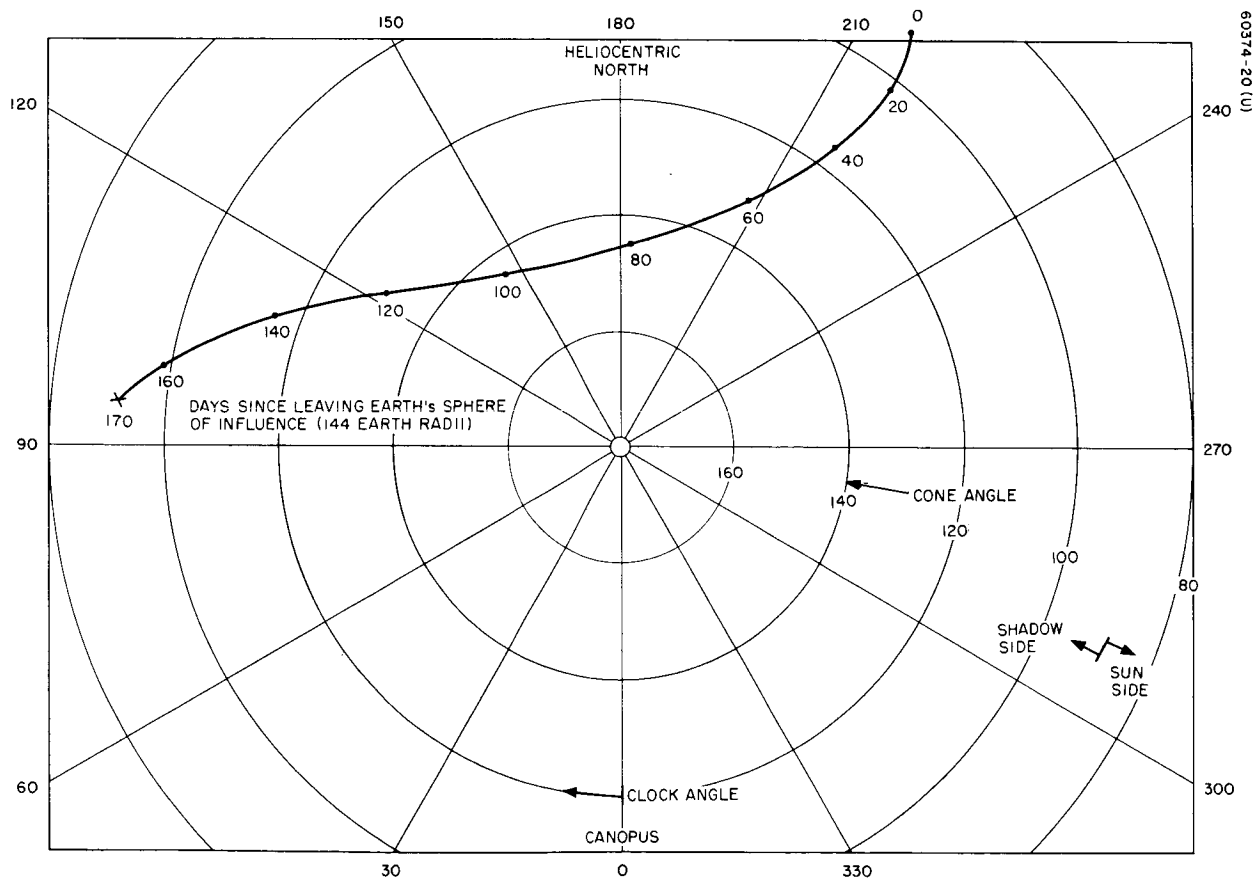


Figure 5-50. Typical Earth Track Versus Time,
Cone/Clock Coordinates, as Viewed from Spacecraft,
for 1975 Launch to Venus
Encounter = x = JD 42, 719.8
(3 November 1975)

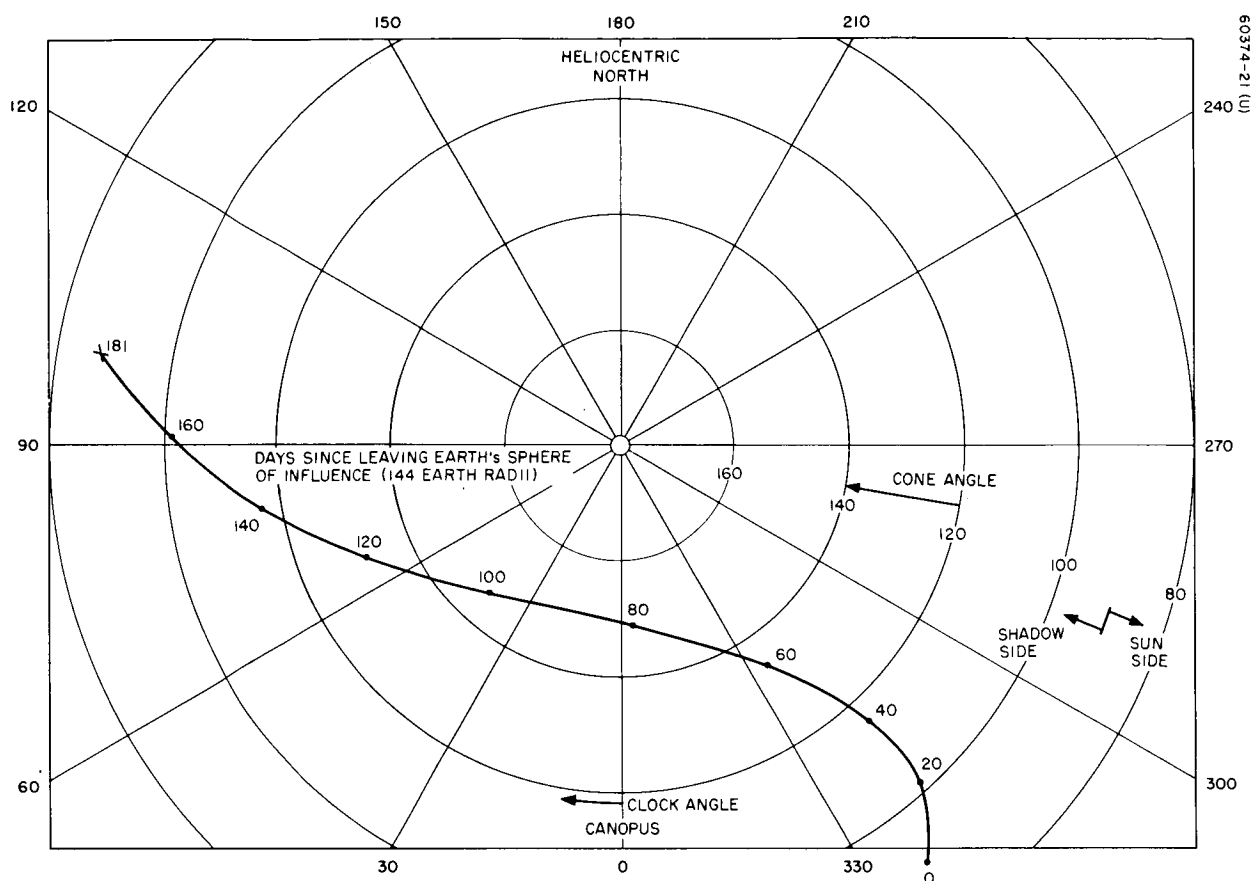


Figure 5-51. Typical Earth Track Versus Time,
Cone/Clock Coordinates, as Viewed from Spacecraft,
for 1976 Launch to Venus Shadow Side,
Encounter = $x = \text{JD } 43,314.6$
(20 June 1977)

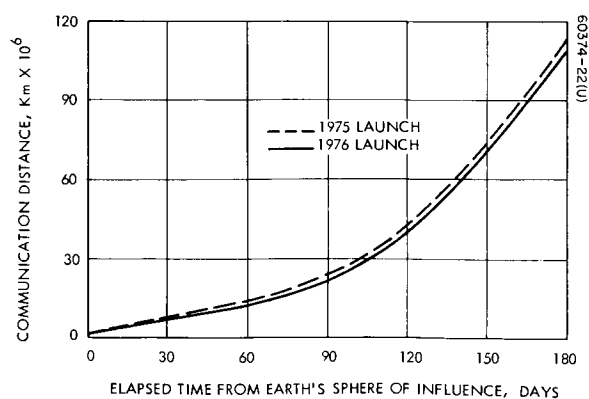


Figure 5-52. Communication Distance
Versus Time, for 1975 and 1976
Launch to Venus

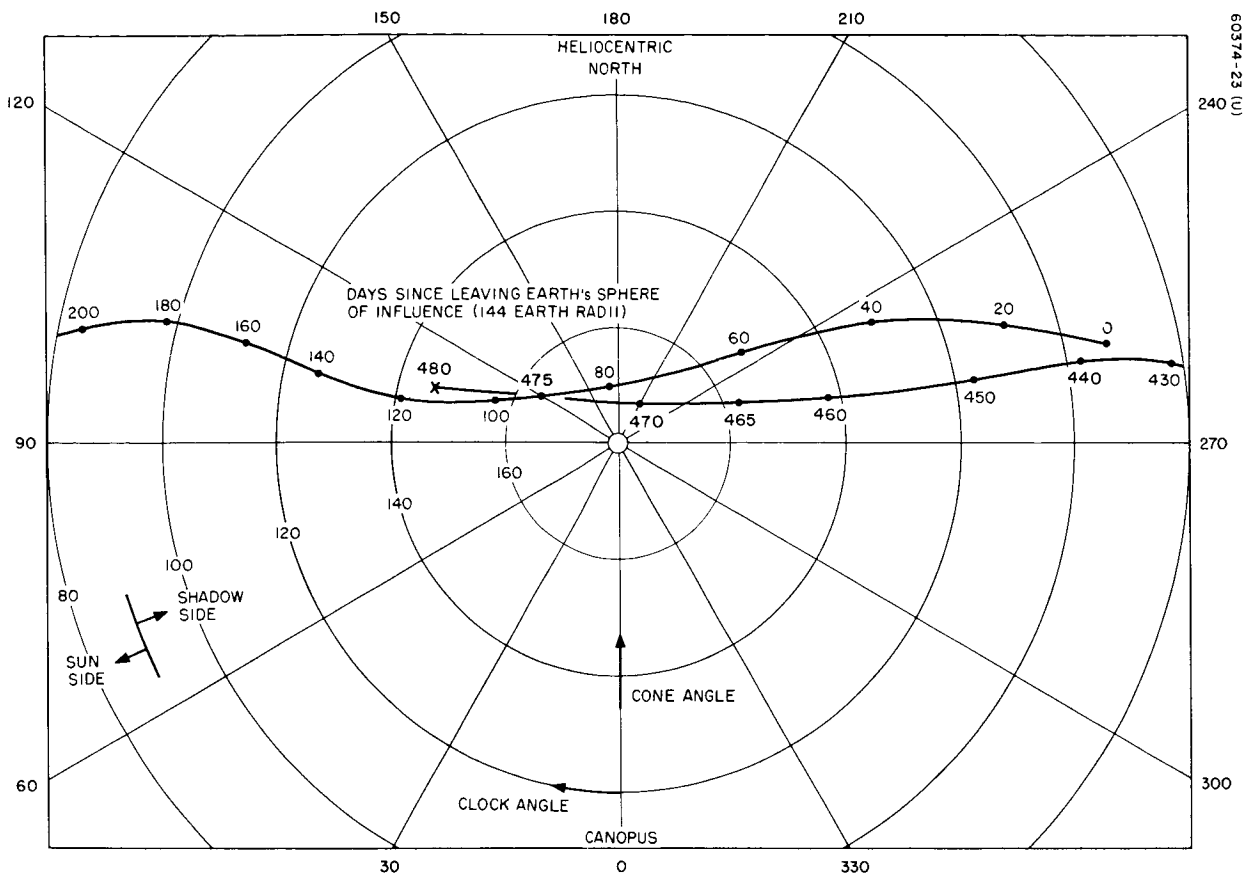


Figure 5-53. Typical Earth Track Versus Time,
Cone/Clock Coordinates as Viewed from Spacecraft,
for 1974 Launch to Mercury Shadow Side,
Encounter = x = JD 42,584.4
(21 June 1975)

side, where it remains until encounter. The telecom range is short at the start and at encounter, but between 220 and 420 days it is great enough to exceed the range of a hemispherical-coverage antenna for downlink telemetry. Thus the most appropriate antenna type for this mission is a medium-gain antenna having a 60-degree pancake pattern (and providing a gain of about 3 db). A hemispherical-coverage antenna may be additionally provided on the shadow side, for backup early and late in the mission, if desired.

Subsystem Design Criteria

- Telecommunications between earth and spacecraft are to be compatible with DSIF capability as programmed.
- RF power source is to be limited to 50 watts.

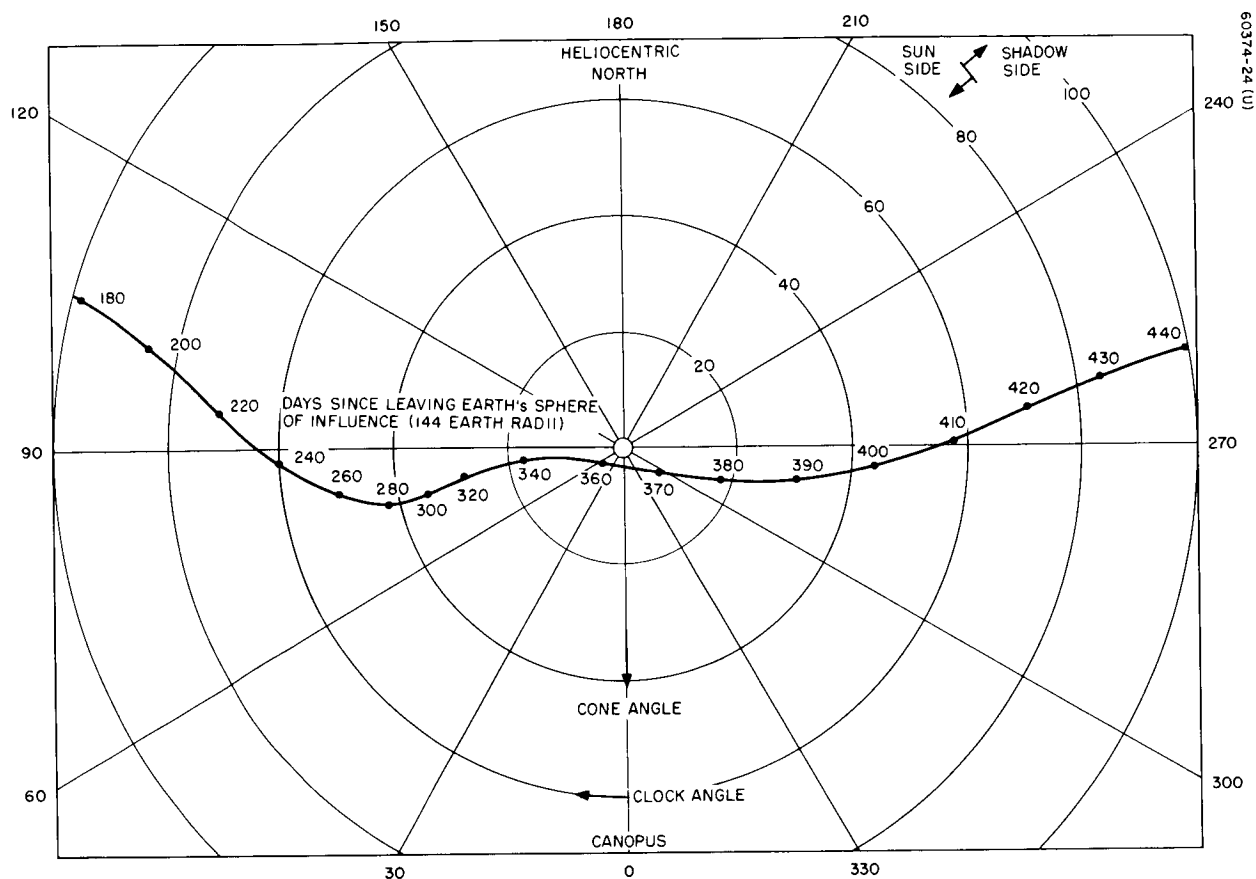


Figure 5-54. Typical Earth Track Versus Time, Cone/Clock Coordinates, as Viewed from Spacecraft, for 1974 Launch to Mercury Sun Side

- Spacecraft receiver to be noncryogenic.
- Only fixed (i. e., no segmented or unfurling) antenna types are to be considered.
- Duty of telecom links is to be continuous from launch, except for solar and planetary occultations.
- Data storage capacity 10^8 bits. Data bursts may be stored and transmitted when convenient.
- Video capability (slo-scan) desired after rendezvous. 1967 technology to be applicable for telecom link.

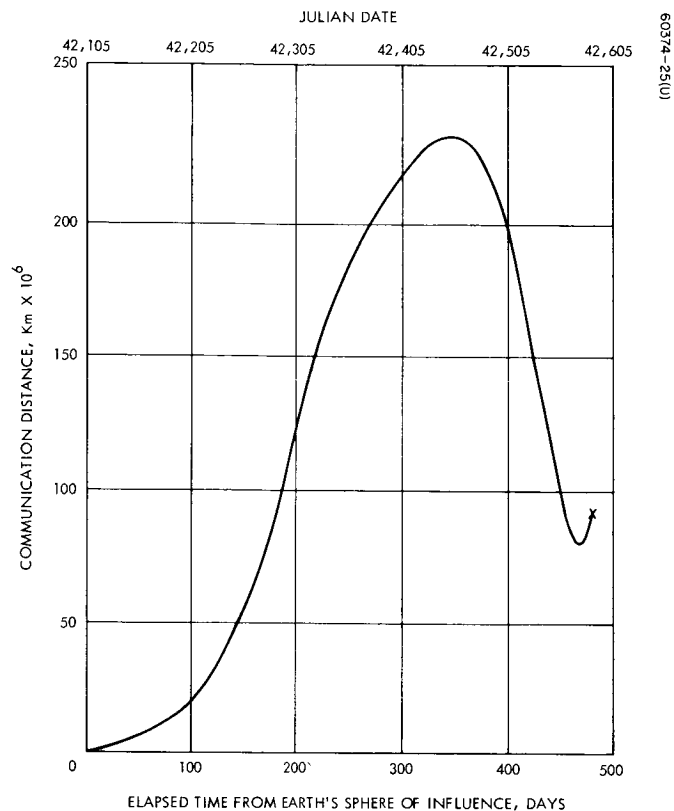


Figure 5-55. Communication Distance
Versus Time, for 1974 Launch
to Mercury,
Encounter = x = JD 42,584.4
(21 June 1975)

Subsystem Characteristics SEP Mission to Mars, with a 20-foot Diameter Shroud

Antennas (Figure 5-43)

High-Gain Antenna. A 7-foot array appears to offer the most favorable tradeoff between available volume, desired gain and expected off-axis losses. The antenna may be characterized as follows:

Size and type	7 by 7 foot planar array
Polarization	RH circular
Gain	32.7 db
3-db beamwidth	3.8 degrees
Off-axis loss at ± 1.4 degrees	1.6 db

Figure 5-56 shows the 7 by 7 foot planar array as roughly equivalent to an 8 foot diameter parabolic antenna (even though the latter offers a larger aperture area). This results from the higher illumination efficiency attainable with the array. The array is also advantageous in that, being flat instead of curved, and with no feed protruding for illumination, it offers a smaller stowage volume.

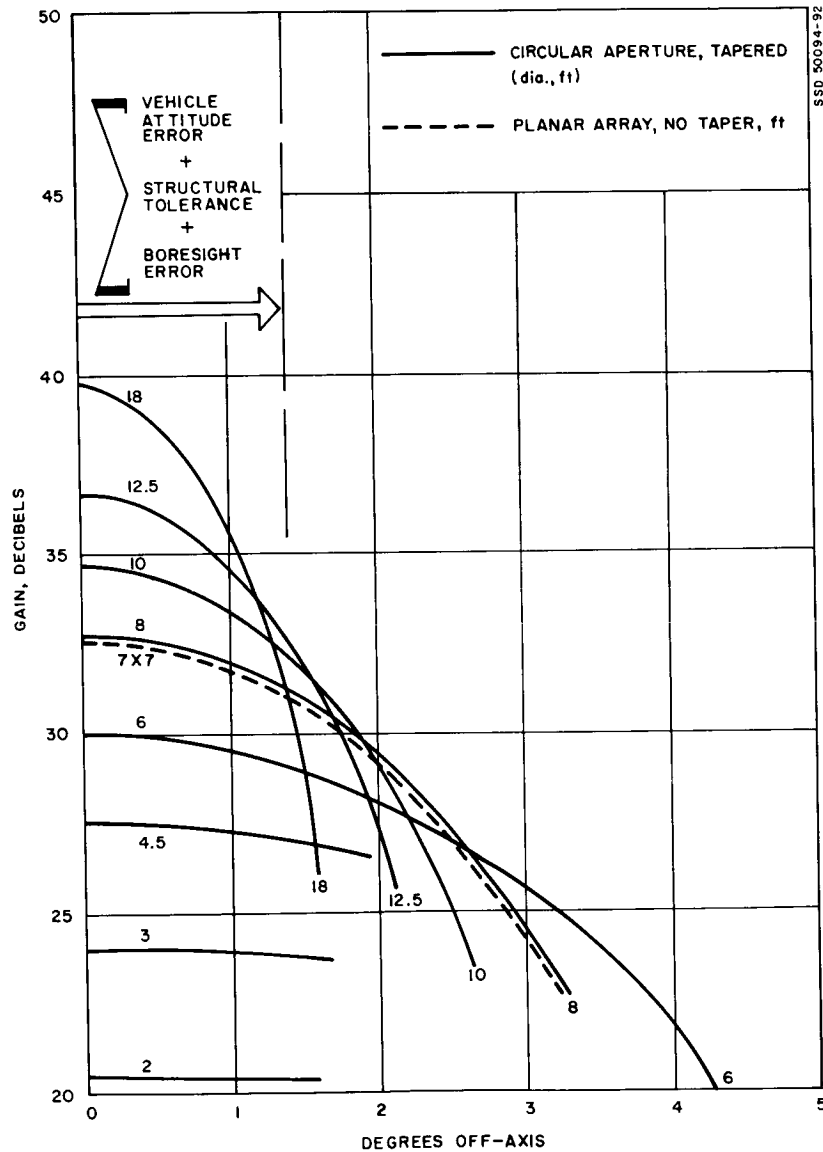


Figure 5-56. Antenna Gain Versus Off-Axis Pointing at 2300 mc

Figure 5-56 also shows that for an antenna larger than this, the narrower beamwidth would require that

$$\begin{aligned} &\text{Pointing tolerance be less} \\ &\text{than } \Sigma (\text{vehicle attitude error} \\ &+ \text{structural tolerance} + \text{boresight error}) = \pm 1.4 \text{ degrees} \end{aligned}$$

Such fine-control pointing would place additional dynamic load and stability demands on the vehicle's attitude control system (unless the antenna angular momentum is fully compensated, or unless a two-degree-of-freedom phased array were used for inertia-less pointing). Furthermore, the higher directivity of the antenna would make the acquisition task more difficult, and would complicate the problem of obtaining sufficient structural rigidity during launch. Finally, the antenna specified will fit inside the 21-foot shroud without resorting to segmented or unfurling antenna configurations.

For the size contemplated, the weight of such an array is comparable to that of a parabolic reflector. The proposed array is similar to one used on Surveyor.

Low-Gain Antennas. The major problem of the low-gain antenna system is the sheer size of the solar array. No single antenna with practical support structure can provide adequate coverage (of say, not less than 3π steradians), unless perhaps it were mounted on the tip of a deployed solar panel. But that would entail long feeders (with big losses); furthermore, the low-gain antenna system must be usable prior to deployment of the solar panels; also, it is helpful not to mount antennas on any component that is to be jettisoned at encounter. It follows that for adequate spatial coverage, and for acceptable location on the spacecraft, a two-antenna configuration is called for, with switch selection between them (as well as between both low-gain antennas and the high-gain antenna). The primary low-gain antenna is located on the sun side of the solar array, and an auxiliary low-gain antenna fills in the shadow side of the solar array.

The primary low-gain antenna is stowed beside the high-gain antenna. When both (high-gain and primary low-gain) antennas are deployed, neither will intrude into the other's pattern. Low-gain antenna coverage as shown is intended to be optimized for the sun side, since maximum communication distances occur there. Both low-gain antennas must have a well-defined shadow cone (Figure 5-43) to avoid spurious reflections from portions of the spacecraft or solar array. Experience with the Surveyor has shown that spacecraft structure detail sized on the order of a wavelength, even if only weakly illuminated, may still degrade the desired pattern substantially. Definitive pattern information can be obtained only by measurement on an actual or model spacecraft.

The low-gain antennas are circularly polarized in the directions normal to the solar array, and linearly polarized in directions parallel to the solar array.

RF Power Source

For a deep space mission departing today, a maximum of 20 watts would be available, using an Apollo-type TWT. A mission departing in 1971 (based on 1967 technology), could be provided with at least 50 watts of RF power, either by improved performance of a single tube (TWT or klystron amplifier), or by the use of parallel tubes.

Present beam efficiency for the Apollo TWT is 35 percent, including heater power. Power conditioning losses would bring this down to about 30 percent, and other circuitry for frequency control, drive, and modulation would bring the overall efficiency of the transmitter to about 27 percent.

Alternate configurations offering higher power and/or higher efficiency are possible, but the selection of a fully space qualified RF power source by 1967 limits the choice.

Receiver Front End

Either a parametric amplifier or a tunnel diode amplifier (TDA) could be used as a noncryogenic front end for the spacecraft receiver. A paramp would require substantial pump power at K-band, however, and would yield a noise figure only about 2 db better than a TDA. The latter is therefore recommended. Either type of preamplifier tends to fail safely, that is, safe failure of a preamplifier (if followed by an adequate and working postamplifier with a-g-c-) would increase the noise figure by perhaps 6 db — a noncatastrophic event. The TDA is therefore not provided redundantly.

Size and Weight

Weight of vehicle telecommunication subsystem (includes antenna and tape recorder)	165 pounds
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Volume of vehicle telecommunication subsystem	3500 cubic inches
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Special Interfaces

Deep Space Network

Telecommunications between earth and spacecraft are to be compatible with the capability of the Deep Space Network as programmed. For details refer to:

- 1) "System Capabilities and Development Schedule of the Deep Space Instrumentation Facility, 1964-1968," Tech. Memo 33-83, Revision 1; JPL, April 1964.
- 2) "Planned Capabilities of the DSN for Voyager," EPD-283; JPL, 15 September 1965.

- 3) "Description of the Deep Space Network Operational Capabilities as of January 1, 1966," Tech. Memo 33-255; JPL, July 1966.

Vehicle Guidance and Attitude Control

It might be conceivable to use a body-fixed high-gain antenna, provided that the resulting constraints on the thrust angle turned out to be acceptable. The potential simplification resulting from deletion of the antenna pointing task seems sufficiently important to justify some exploration of the trajectory constraints that such an approach would impose. In particular, it would have to be resolved if a thrust angle constant versus earth (rather than versus sun) would significantly reduce either payload or flexibility, and if course corrections (by means of thruster pointing) will have been brought down to residual levels by the time the high-gain antenna is switched into service.

While elimination of a separate antenna pointing mechanism seems an intriguing possibility in the case of the Mars missions, it appears quite attractive in the context of a Jupiter mission, in the course of which the solar-electric propulsion is shut down after 550 days.

Possible Telecom Limitations

Possible degradation of telecom performance by an electric propulsion system has not been thoroughly studied (note that any such degradation would be incurred only in the transfer trajectory and not after encounter). Possible interfering mechanisms are:

- 1) Effects of non-neutralized propellant residue such as:
 - a) Specular reflection and/or scattering of signal energy (uplink and downlink)
 - b) Absorption of signal energy (uplink and downlink)
 - c) Transmitter power restrictions arising from breakdown and mismatch at antenna.
- 2) Effects of noise power emission by electric propulsion system, appearing as radio frequency interference in the spacecraft receiver (uplink only).

Items a and b are unlikely to be troublesome at maximum range, since the high-gain antenna is then used (with orientation away from the thruster). With a low-gain antenna at intermediate range (say, $30 \cdot 10^6$ km), however, adequate telecom margins are no longer quite so certain. As to item 2, the available evidence suggests caution and further work.

Electric Propulsion for Heliocentric Transfer Phase. A contracted study conducted at Hughes in 1961 - 1962* indicated little probability of degrading the downlink since the transmitter aboard the spacecraft would overpower all incidental noise; there would probably be no difficulty with the uplink if a contact-ionization type thruster were used; and there was likely to be some constraint on the uplink with an electron-bombardment type thruster.

In particular, expected and actual noise power spectra up to 1 GHz were reported, which indicated high noise generation at VHF (and hence in the IRIG VHF telemetry band).

The 1965 SERT I spacecraft** which used IRIG VHF telemetry frequencies reported no degradation of the downlink, but noted a somewhat degraded uplink which, however, in the context of a near-earth telecom system, was tolerable. The SERT data as reported are not sufficiently detailed to permit a quantitative determination of actual noise power (although this could presumably be reconstructed with access to NASA and RCA/Astroelectronics notebooks and calibration records); however, such information would cover only a single frequency, and one which probably was not an optimum choice at that. NASA/Lewis had proposed that a wide-band microwave radiometer be flown aboard SERT I to make a sensitive spectrum analysis in situ covering the frequency range to 1 GHz. To this end, an in-house development was in fact initiated at Lewis. Unfortunately, subsequent weight and flight schedule problems forced progressive cutbacks in the experiment until only the VHF command receiver remained, resulting in meager information derived from its (telemetered) AGC-bus voltage. On the basis of present data and in the context of a Mars mission, it appears that further investigation is warranted to assure that the uplink will remain operational to the desired range.

It is the present consensus of opinion that it is not possible to meaningfully simulate the noise mechanisms in a ground environment (such as a combination vacuum chamber plus shielded room), since ion thrusters in general and plasma oscillation frequencies in particular are highly sensitive to environment factors. Only a microwave radiometer experiment flown aboard an ion-propelled spacecraft would provide the needed answers. Thus, on the basis of known data and in the context of a Mars mission, it seems dangerous to assume that the uplink will remain operational to the desired range without further experimental verification.

* R. D. Wanselow and H. L. Wiser, "Investigation of RF Noise Generation from Space Vehicles," final report Contract NAS 8-862.

** NASA Technical Memorandum, NASA TM X-1077, page 10.

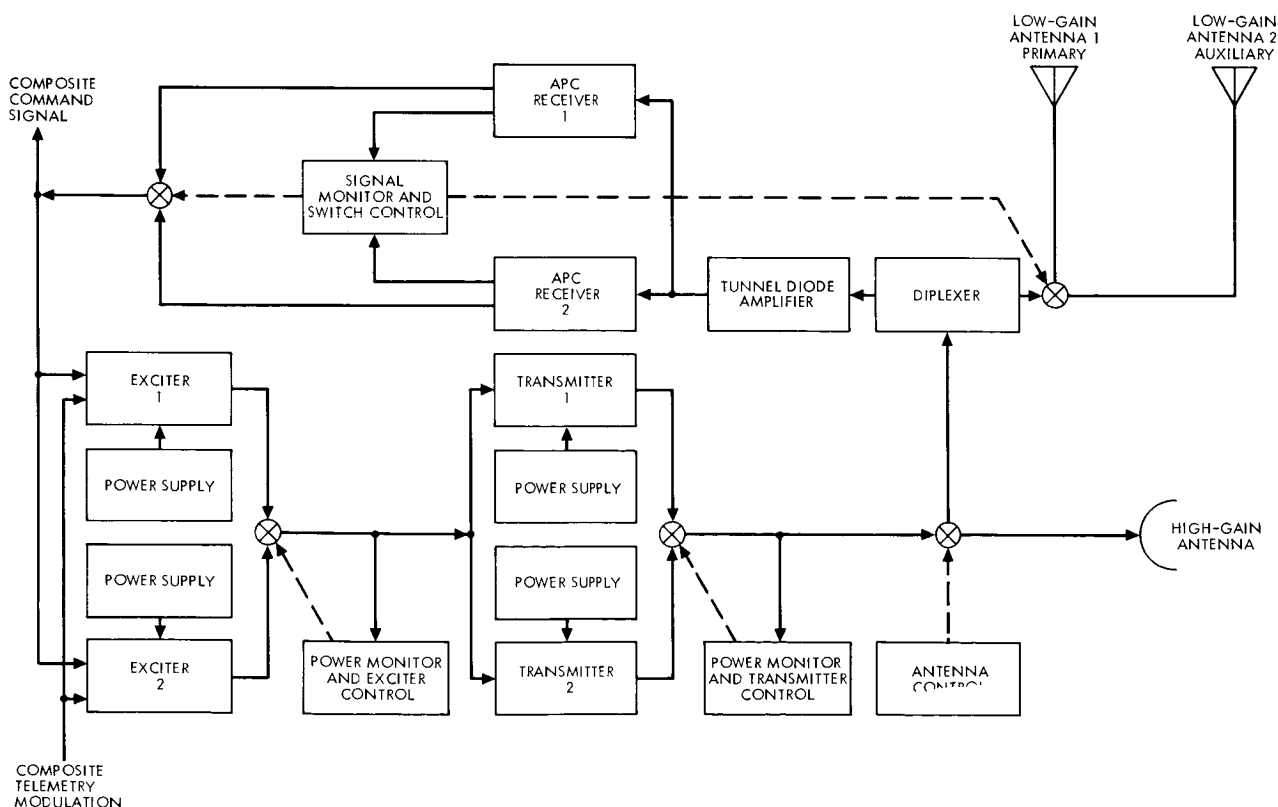
Subsystem Functional Description

Radio Subsystem (Figure 5-57)

The telecom system is required to perform three functions: 1) track position, range and range rate of the spacecraft, 2) telemetry engineering and scientific data from the spacecraft, and 3) transmit commands to the spacecraft.

The radio subsystem is required to receive a modulated RF carrier from stations of the Deep Space Network, demodulate command and ranging signals, coherently translate the frequency and phase of the RF carrier by a fixed ratio, modulate the carrier with telemetry and ranging signals, and retransmit it back to earth.

When a signal is not being transmitted to the spacecraft, transmitter frequency control is provided by an auxiliary crystal oscillator. This non-coherent mode of operation permits one-way doppler tracking, angular position tracking, and telemetry reception by the ground stations.



SSD 50094-93

Figure 5-57. Spacecraft Radio Subsystem

The methods and rationale for failure detection and switching are similar to those of Mariner. The control of failure switching between redundant RF elements is provided by either ground command or on-board failure detection. In the case of ground control, receipt of the appropriate command causes transfer of dc power from the active to the standby element, and reversal of the circulator switch.

The signal monitor cycles both receivers as well as both low-gain antennas. Cycling is initiated 1) whenever the received signal falls below usable level, and 2) periodically, to ensure that the best available signal is being used. Status of the signal monitor, as well as of the other failure monitors, is telemetered to earth.

Telemetry Subsystem

The telemetry tasks are summarized in Figure 5-58. Note that the video, science and engineering data are all multiplexed in the high-rate telemetry mode. Only the engineering data can be accommodated in the low-rate telemetry mode. The telemetry and data automation techniques used are adapted from those of the Mariner. PCM/PSK/PM modulation-demodulation, in combination with PN synchronization codes, is employed. Since the rate at which video data is gathered exceeds the capability of the telemetry channel, storage and playback are provided by a tape recorder.

Tape Recorder (Mars Missions)

A capacity of 10^8 bits is desired for the tape recorder. For 500-line TV pictures, on the order of 100 frames could be stored (Roberts' modulation), which would leave ample space for science and engineering data. On the basis of a packing density of 1000 bps it will take a playback speed of 3 ips to accommodate 1500 bps (in Manchester code). For a record speed of 30 ips, this would yield a tape speed ratio of 10. Eight tracks on 2100 feet of tape, with the tracks sequentially scanned, would provide the desired storage capacity, and would allow some flexibility in access.

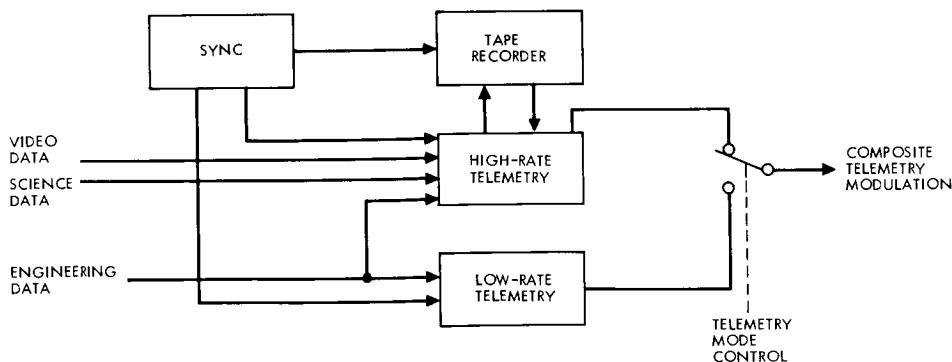


Figure 5-58. Spacecraft Telemetry Flow Diagram

Ground Command Versus Spacecraft Sequencing and Computing

The command system provides for the transmission of both discrete and quantitative commands to the spacecraft. At the maximum range of $350 \cdot 10^6$ km (for a Mars mission) a round-trip propagation time must be contended with of 33 minutes. It follows that for each spacecraft control function, the simplicity and reliability of ground control must be weighed against the resulting reaction time and telecom load. A central computer and sequencer is provided, with ground command backup of all vital functions.

Other Telecom Functions

It may be desirable to use the spacecraft as a two-way telecom relay to another spacecraft, to an atmospheric probe, or to a lander vehicle. No such telecom need has been explicitly spelled out here, but the possibility of such a requirement should be noted, and appropriate weight and power allowances must be set aside for such use.

Summary and Conclusions

No special telecom difficulties are expected, except possibly in the area of noise power emission by the electric propulsion system, appearing as radio frequency interference in the spacecraft receiver (uplink only, as discussed under "Possible Telecom Limitations"). The task of resolving this interface represents a major pacing item.

Recommended Action

It is recommended that an effort be made to conduct a meaningful flight experiment aboard an earth-orbital vehicle to resolve uncertainties as to any interaction between the electric propulsion and the telecom systems. A spacecraft carrying such an experiment should have the same type of thruster that is to be used for solar electric propulsion on a larger vehicle. Experiment instrumentation should consist, as a minimum, of a sensitive microwave radiometer to explore the power and spectral composition of local emissions in the 1750 to 1850 MHz and the 2100 MHz bands. These tests should include a hemispherical-coverage antenna and a high-gain antenna with a programmed conical scan. Then, by turning the thruster on and off (ground command), changes in the signal as received at the spacecraft could be readily noted and telemetered. Additionally, it would be of engineering interest to note if the antennas, when used in the 2300 MHz transmit function, see a change in conductance, susceptance, and linearity of the surrounding propagation medium.

It is fully recognized that the experiment suggested calls for a major well-coordinated, technical and administrative effort on the part of all concerned.

RETRO PROPULSION

All spacecraft configurations designated for orbiter missions presented herein assume the utilization of a liquid bipropellant retro-rocket subsystem. The systems have been sized for a specific impulse of 315 seconds for a liquid propellant utilizing MMH fuel and N_2O_4 oxidizer.

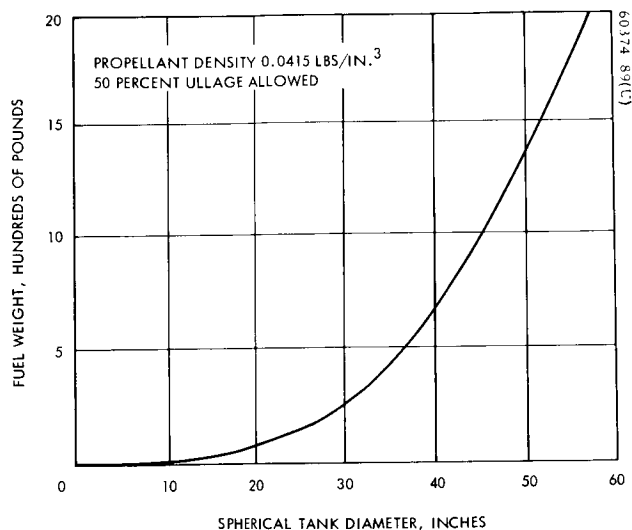


Figure 5-59. Fuel Weight Versus Spherical Tank Diameter

Figure 5-59 relates spherical tank size versus propellant weight. This retro-rocket system sizing procedure is consistent with the retro-rocket performance curve used in the mission analysis (see Figure A-2 of Appendix A).

STRUCTURAL CONSIDERATIONS

Based on the findings and conclusions of the structural and dynamics analyses performed during the Phase I effort of the SEP system study, certain baseline structural considerations were adhered to in the conceptual design of spacecraft within the scope of the follow-on phase of the SEP system study. The ground rules adhered to for the spacecraft configurations presented herein are:

- The spacecraft-launch vehicle structural interface, or field joint, will be a separate joint from the spacecraft launch vehicle separation plane.
- The separation joint may be located at any station upstream of the permanent field joint with sufficient clearance allowed for the stowed solar array panels.
- The electric engine thruster array will be externally mounted to the prime spacecraft structure so that large structural cutouts with associated peripheral reinforcement will not be required; also, no explosive, jettisonable, or hinge opening structural hatches are to be considered to permit clearances for the electric thrusters exhaust beam from within the spacecraft bus.
- For adequate engine array the spacecraft structure clearance should exist to permit translation of the engine in any direction normal to the thrust vector for spacecraft center of gravity - thrust vector alignment and two-axis attitude control purposes.

- Power conditioning panel modules are to be mounted externally on the spacecraft bus structure and are not to serve as prime load carrying members in themselves. The design of the power conditioning modules support racks are therefore not constrained by spacecraft structural environment and their location on the bus.
- Only the individual in-board solar panels are to be retained for the orbiter spacecraft retro-maneuver and the cantilevered panels may be additionally supported other than at their primary hinge line on the spacecraft bus by a tension or compression strut.
- The entire volume within the stowed solar panel stacks may be occupied as required by spacecraft subsystems and/or structure. The bus is not limited to a circular volume inscribed within the square envelope provided by the stowed solar panels. The spacecraft structural rigidity is enhanced by relaxing the bus envelope constraint in this manner.

During Phase I of the SEP system study, a detailed analysis of the dynamical behavior and internal forces for the Saturn IB/Centaur SEP spacecraft during the boost phase was conducted. Results of this study were presented in the Phase I Final Report in Section 6-87. The spacecraft conceptual designs developed during the follow-on study phase were primarily to show feasibility of adopting a basic spacecraft design which would be suitable for a variety of missions. No detailed dynamics analyses were conducted for each of the configurations presented herein since the designs are basically similar in concept to the configuration analyzed in detail during Phase I.

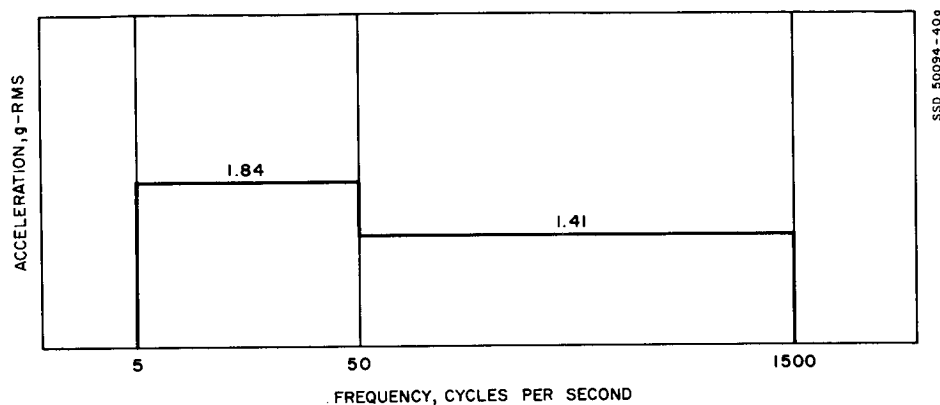
During Phase I of the study the structural arrangement of the SEP spacecraft was primarily determined by vehicle dynamics during the boost phase. Three basic structural arrangements were analyzed utilizing a standard Hughes computer program. The program calculates the following data from a vibratory input:

- 1) Vibration modes and frequencies of the vehicle structure
- 2) Accelerations, velocities, and displacements for 25 members of the structure
- 3) Loads in these members

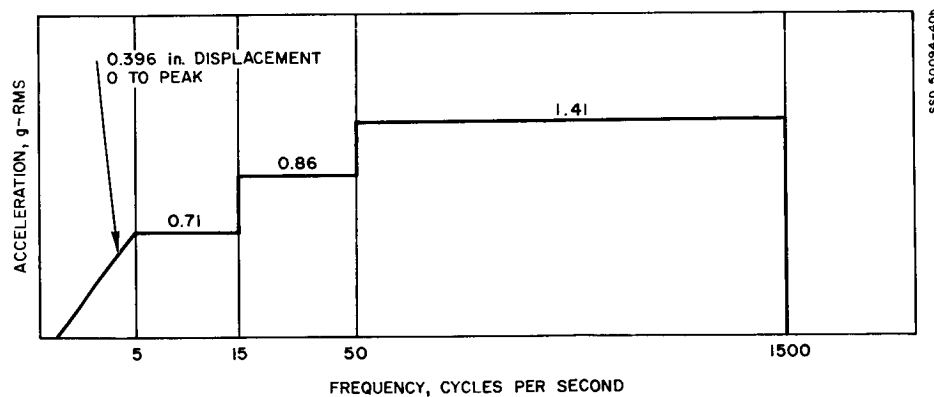
Dynamic and quasi-dynamic environment caused by the launch vehicles were estimated and used as inputs to the program. These conditions are listed in Table 5-20, where Z and x-y refer respectively to launch vehicle longitudinal and lateral directions, and VFSW = variable frequency sine wave. Loads due to stage separation, deployment of any subsystem equipment, electric engine operation, and jettison of equipment were all assumed to be negligible.

Figure 5-60 shows the input acceleration as a function of input frequency. Yield strength of the structural members was used as the criterion for design.

Three structural configurations were considered during Phase I of the study and for detailed discussion and results reference may be made to Section 6 page 6-87 of the Phase I Final Report.



a) Thrust Axis (Z-Direction)



b) Lateral Axis (Any X-Y Direction)

Figure 5-60. Variable Frequency Sine Wave

TABLE 5-20. DEFINITION OF MECHANICAL ENVIRONMENT

Phase	Quasi-Static, g^6 (Z-direction)	Quasi-Static, g (X-Y direction)	Sine (VFSW), g-rms (Z direction) Frequency Range, Hz	Sine (VFSW), g-rms (X-Y direction); Frequency Range, Hz	Sine (VFSW) Torsion About Z axis rad/sec ²	Random (g^2 /cps) White Noise Frequency Band, Hz ³	Endurance Time of Random Vibration, seconds	Description of Shock and Transient Loads
Liftoff ¹	-2.0	2.0			60	0.04 (100 to 1500)	30	Comprehended in sine and random environment
Maximum q^1	-3.0	1.0			60	0.04 (100 to 1500)	50	Comprehended in sine and random environment
First stage burnout ¹	-6.5	0.5	50 percent of Figure 5-60	50 percent of Figure 5-60	0	0	0	Comprehended in sine and random environment
Engine ¹ shutdown	0	0			60	0.04 (100 to 1500)	60	6.5 g quasi-static, decreasing instan- taneously to zero
Retro-engine fire ²	+4.0	0.5	0.5 (5 to 1500)	0.3 (5 to 1500)	0	0.002 (5 to 1500)	100	Comprehended in random environment
Retro-engine shutdown ²	0	0	0.5 (5 to 1500)	0.3 (5 to 1500)	0	0.002 (5 to 1500)	0	4 g quasi-static, decreasing instan- taneously to zero

¹ Loads applied at Centaur-spacecraft interface, i. e., field joint.

² Loads applied at retro-engine attach points.

³ To be applied in Z and any two orthogonal X-Y directions simultaneously.

⁴ Applied at lander/bus interface.

⁵ Environment during second (S-IV-B) and third (Centaur) stage boost comprehended.

⁶ Angular acceleration to be applied about roll, pitch, and yaw axes.

⁷ Directed from 0 to 45 degrees to the plane of the array.

Note: Pressure transients from all pressure induced forces except acoustical loads shall be less than 0.015 psia on any portion of folded array.

6. PERFORMANCE COMPARISON

To more vividly demonstrate the flexibility of the modular electric-propulsion system design, a summary table of the Mars orbiter/lander mission capability of a solar-powered electric-propulsion spacecraft design scaled to a variety of launch vehicles (the Titan III M, Saturn IB/Centaur, and Saturn IB/Zero Stage) — all utilizing the same size module electric engine system — is presented in Table 6-1. This performance comparison is based on the mission payload — defined as the lander capsule, orbital scientific instrumentation, telecommunications equipment, and power generation equipment — that can be placed in a 5000 km circular Martian orbit during the 1970 to 1980 launch opportunities. It is evident from these results that this same basic electric propulsion system design concept is compatible with spacecraft designs ranging in Mars mission payload capabilities from a minimum of 1897 pounds (for a 1971 Titan III M launched spacecraft) to a maximum of 9422 pounds (for a 1977 Saturn IB/Zero Stage launched spacecraft).

TABLE 6-1. LAUNCH VEHICLE COMPARISON
MARS ORBITER AND LANDER MISSIONS

(5000 x 5000 km orbit)

Trajectory Characteristics				Titan III M 9.6 kw at 1.0 AU		Saturn IB/Centaur 14.4 kw at 1.0 AU		Saturn IB/Zero Stage 41.6 kw at 1.0 AU	
Year	C ₃ , (km/sec) ²	Approach Velocity, km/sec	Flight Time, days	Injected Weight	Mission* Payload	Injected Weight	Mission* Payload	Injected Weight	Mission* Payload
1971	4.23	1.96	231	5554	1897	8320	3060	23,300	8127
1973	6.93	1.14	293	5013	1960	7520	3130	21,700	8782
1975	5.49	0.90	404	5301	2140	7950	3422	22,500	9392
1977	4.07	1.10	432	5586	2215	8380	3539	23,400	9422
1979	3.07	1.38	407	5786	2217	8680	3534	24,000	9387

*Mission payload includes scientific instrumentation, telecommunications equipment, and power generation equipment.

To further illustrate the advantages of the solar-powered electric-propulsion spacecraft concept, a performance comparison of electric-propulsion spacecraft with all-chemical propulsion spacecraft for the Mars orbiter/lander mission has been made.

The performance comparison for the Mars mission is presented for both the 1973 and 1979 launch opportunities. For the Mars mission comparison, the performance capability using electric propulsion is based on the Titan III M launch vehicle, with the spacecraft previously defined. The required all-chemical spacecraft injected weight is determined that would in effect provide the identical mission payload capability. Tables 6-2 and 6-3 indicate weight breakdowns for a 1973 all-chemical propulsion spacecraft, respectively.

TABLE 6-2. TITAN III M SEP SPACECRAFT WEIGHT BREAKDOWN
1973 MARS ORBITER AND LANDER MISSION
(10 kilowatts power level)

Items		Weight, pounds	
		Injected	At Approach
Mission payload			
Science payload	1510*	1960	1960
Telecommunications	180		
Power	270**		
Solar array (portion jettisoned)		400	---
Electric engine system		250	---
Thrustor system	66		
Propellant tankage	24		
Power conditioning and controls	94		
Structure and mechanisms	34		
Feed system	32		
Propellant, Hg		273	---
Guidance and control		95	90
Structure		455	405
Thermal control		40	40
Electrical harness		60	54
Retro system		1480	1480
Totals		5013	4029

* Lander capsule weight is included as part of science payload weight.

** 100 pounds of power associated with the mission payload is that portion of the solar array subsystem retained with the orbiter.

The retro system weight is based on an approach velocity of 1.14 km/sec, propellant I_{sp} = 315 seconds, and mass fraction of 0.90.

TABLE 6-3. ALL-CHEMICAL SPACECRAFT
1973 MARS ORBITER AND LANDER MISSION
(5000 x 5000 km orbit)

Items	Weight Injected, pounds	
Mission payload		1960
Science payload	1510	
Telecommunications	180	
Power	270	
Guidance and control		95
Structure		670
Thermal control		40
Electrical harness		<u>60</u>
Total		2825

To approximate the total injected weight, the following relationship was used:

$$\begin{aligned}
 \text{Total injected weight} &= W_i = 2825 + (\text{retro system}) \\
 &\quad + (\text{midcourse correction propulsion}) \\
 &= \left[2825 + 1.00 - \frac{W_{\text{orb}}}{W_{\text{app}}} (W_i) \right] 1.02
 \end{aligned}$$

where

$$\frac{W_{\text{orb}}}{W_{\text{app}}} = 0.39 \text{ for } 3.3 \text{ km/sec } V_{\text{approach}} \text{ for } 5000 \times 5000 \text{ km orbit.}$$

The 1.02 accounts for 2 percent of the injected weight for the midcourse propulsion system

$$W_i = [2825 + (1.00 - 0.39) W_i] 1.02$$

$$W_i = 2881 + 0.622 W_i$$

$$W_i = \frac{2881}{0.378} = 7625 \text{ pounds } \left(C_3 = 15.5 \frac{\text{km}^2}{\text{sec}^2} \right)$$

Tables 6-4 and 6-5 indicate weight breakdowns for a 1979 Titan III M solar-powered electric-propulsion spacecraft and a 1979 all-chemical propulsion spacecraft, respectively. The required injected weights for the all-chemical spacecraft that would, in effect, provide the identical mission payload capability for the 1973 and 1979 launch opportunities, are shown in Table 6-6 in comparison to the solar-electric spacecraft capabilities.

TABLE 6-4. TITAN III M SEP SPACECRAFT WEIGHT BREAKDOWN
1979 MARS ORBITER AND LANDER MISSION
(9.6-kw power level)

Items	Weight, pounds	
	Injected	At Approach
Mission payload	2217	2217
Science payload		
Telecommunications	1767*	
Power	180	
Solar array (portion jettisoned)	270	
Electric engine system	400	---
Thrustor system	250	---
Propellant tankage		
Power conditioning and controls	66	
Structure and mechanisms	24	
Feed system	94	
Propellant, Hg	34	
Guidance and control	394	---
Structure	95	90
Thermal control	520	465
Electrical harness	40	40
Retro system	60	54
	1810	1810
Totals	5786	4676

* Lander capsule weight is included as part of science payload weight.

** 100 pounds of power associated with the mission payload is that portion of the solar array subsystem retained with the orbiter.

The retro system weight is based on an approach velocity of 1.14 km/sec, propellant I_{sp} = 315 seconds, and mass fraction of 0.90.

TABLE 6-5. ALL-CHEMICAL SPACECRAFT
1979 MARS ORBITER AND LANDER MISSION
(5000 x 5000 km orbit)

Items	Weight Injected, pounds	
Mission payload		2217
Science payload	1767	
Telecommunications	180	
Power	270	
Guidance and control		95
Structure		1600
Thermal control		40
Electrical harness		<u>60</u>
Total		4012

To approximate the total injected weight, the following relationship was used:

$$\begin{aligned}
 \text{Total injected weight} &= W_i = 4012 + (\text{retro system}) \\
 &\quad + (\text{midcourse correction propulsion}) \\
 &= \left[4012 + \left(1.00 - \frac{W_{\text{orb}}}{W_{\text{app}}} \right) (W_i) \right] 1.02
 \end{aligned}$$

where

$$\frac{W_{\text{orb}}}{W_{\text{app}}} = 0.24 \text{ for } 4.5 \text{ km/sec } V_{\text{app}} \text{ for } 5000 \times 5000 \text{ km orbit.}$$

The 1.02 accounts for 2 percent of the injected weight for the midcourse propulsion system.

$$W_i = [4012 + (1.00 - 0.24) W_i] 1.02$$

$$W_i = 4092 + 0.775 W_i$$

$$W_i = \frac{4092}{0.225} = 18,200 \text{ pounds} \quad \left(C_3 = 14.0 \frac{\text{km}^2}{\text{sec}^2} \right)$$

TABLE 6-6. PERFORMANCE COMPARISON OF
ELECTRIC-PROPULSION SPACECRAFT
WITH ALL-CHEMICAL SPACECRAFT

	Solar-Powered Electric-Propulsion Spacecraft		Equivalent All-Chemical Spacecraft	
	1973	1979	1973	1979
Required C_3 , km/sec ²	6.93	3.07	15.5	14.0
Approach velocity, km/sec	1.14	1.38	3.3	4.5
Mission payload, pounds*	1960	2217	1960	2217
Required injected weight, pounds	5013	5786	7625	18,200
Required launch vehicle	Titan III M	Titan III M	Saturn IB/ Centaur	Saturn IB/ Zero Stage

* For 5000 kilometer circular Martian orbit.

The 1979 all-chemical spacecraft in this comparison is approximately the size currently under study in the Voyager program, to be launched by the Saturn V launch vehicle. The 1973 all-chemical spacecraft is approximately the same as that previously studied for the Saturn IB/Centaur launch vehicle, which is no longer being considered for development.

Within the capabilities of the family of launch vehicles considered here, it would require a Saturn IB/Zero Stage booster, now only in the preliminary study phase, to accomplish these Mars missions with a comparable mission payload to that attainable with a Titan III M launched solar-powered electric-propulsion spacecraft. Estimated launch vehicle costs (including launch operations) for the Saturn IB/Zero Stage booster are approximately four times those for the Titan III M.

APPENDIX A. TRAJECTORY INPUT DATA

Figures A-1 through A-4 show the input data that was used for performance computation. The trajectory program used this data in tabular form linearly interpolated (Tables A-1 through A-3).

Figures A-1 and A-2 and Table A-1 show injected weight capability of the design boosters and retro rocket performance used for orbiter missions. This data was provided by Jet Propulsion Laboratories. Figure A-3 and Table A-2 show propulsion system efficiency (power converter and engine system) as a function of specific impulse. This data was generated by Hughes Aircraft Company.

Previous trajectory studies utilized a power curve that varied as the inverse 1.7 power of heliocentric radius. This approximation was used because the information required to compute the power profile more accurately was not available. A relative power curve for use in trajectory studies on the present program was generated by Hughes from the information available at the time the final simulation studies were initiated. This power profile is given in Figure A-4 and Table A-3. More recent information indicates that while this power profile is an improvement over that which was used previously, it should be modified to conform to the relationships discussed in Appendix D.

The Mars orbiter and Jupiter flyby missions have been simulated using the power curve given in Appendix E. These simulations indicate that the power level at 1 AU should be reduced about 10 percent on each of these missions; performance improvements of about 1.5 percent in net orbit weight at Mars and about 15 percent of net approach weight for the Jupiter flyby occur as a result. The Venus orbiter mission (which was not simulated with this power curve) will show a decrease in net orbit weight, which may be expected to be of the order of 2 to 3 percent. These performance changes are large with respect to the variations that have been shown to exist for other vehicle and propulsion system parameters (e.g., specific impulse, thrust attitude), although for the inner planets the changes are fairly small in an absolute sense. It does appear that the Jupiter flyby design may be conservative.

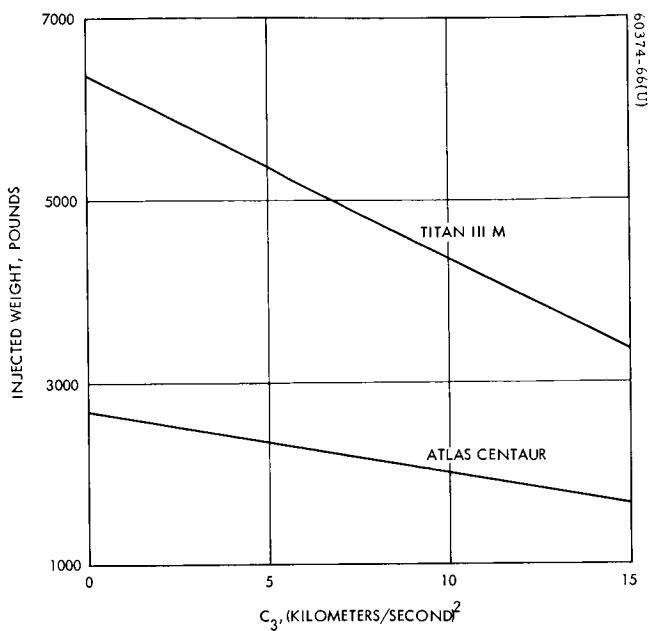


Figure A-1. Injected Weight (pounds) versus C_3 (km/sec)² for Atlas/Centaur and Titan III M

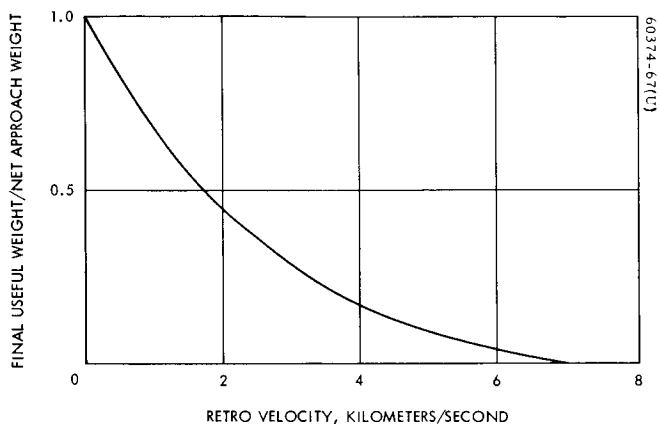


Figure A-2. Final Useful Weight/Net Approach Weight versus Retro Velocity, km/sec

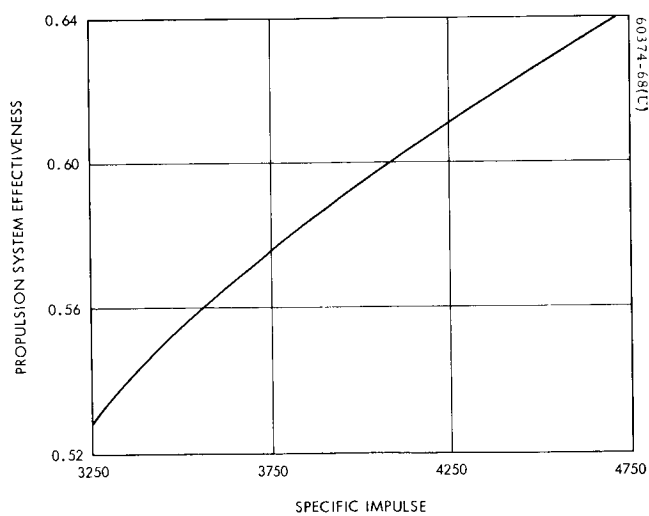


Figure A-3. Propulsion System Effectiveness versus Specific Impulse

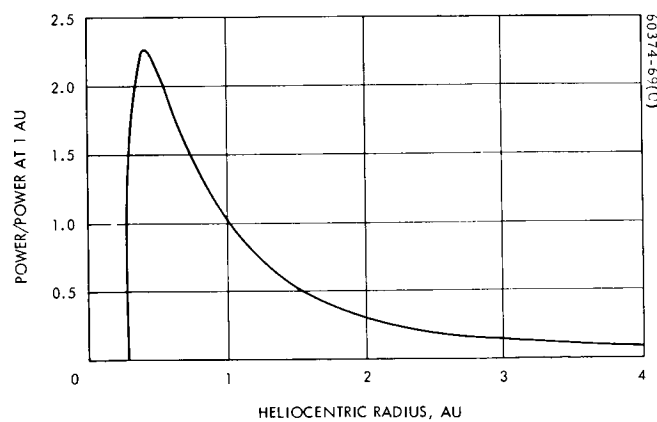


Figure A-4. Power/Power at 1 AU versus Heliocentric Radius, AU

TABLE A-1. INJECTED WEIGHT CAPABILITY

Specific Impulse	Propulsion System Efficiency	C_3 -(km/sec) ²	Injected Weight, pounds	
			Titan III M	Atlas Centaur
3250	0.528			
3500	0.554	0	6400	2700
3750	0.575	5	5400	2350
4000	0.595	10	4400	2000
4250	0.611	15	3400	1650
4500	0.627			
4750	0.642			
5000	0.653			

TABLE A-2. PROPULSION SYSTEM EFFICIENCY

Retro Velocity, km/sec	$\frac{\text{Final Useful Weight}}{\text{Net Approach Weight}}$
0	1.0
0.5	0.81
1.0	0.66
1.5	0.545
2.0	0.445
2.5	0.36
3.0	0.285
3.5	0.225
4.0	0.175
4.5	0.13
5.0	0.095
5.5	0.07
6.0	0.045
7.0	0

TABLE A-3. POWER PROFILE

Heliocentric Radius, AU	<u>Power</u> Power at 1 AU	Heliocentric Radius, AU	<u>Power</u> Power at 1 AU
0	0	1.35	0.631
0.3	0	1.4	0.590
0.35	1.937	1.45	0.553
0.4	2.241	1.5	0.519
0.45	2.241	1.55	0.488
0.5	2.142	1.6	0.460
0.55	2.008	1.7	0.411
0.6	1.863	1.8	0.369
0.65	1.722	1.9	0.333
0.7	1.588	2.0	0.303
0.75	1.465	2.1	0.276
0.8	1.353	2.2	0.253
0.85	1.251	2.4	0.214
0.9	1.158	2.6	0.184
0.95	1.075	2.8	0.160
1.0	1.000	3.0	0.140
1.05	0.932	3.5	0.1042
1.1	0.870	4.0	0.0806
1.15	0.814	5.0	0.0516
1.2	0.763	6.0	0.0358
1.25	0.717	8.0	0.0202
1.3	0.675	10.0	0.0129

APPENDIX B. DESCRIPTION OF CONTINUOUS THRUST THREE DIMENSIONAL TRAJECTORY PROGRAM

The trajectory program described in this appendix has been developed to facilitate mission analysis tasks associated with the selection of design parameters for a solar powered electric propulsion spacecraft.

BASIC ASSUMPTIONS

Only the heliocentric phase of an interplanetary flight is integrated in the present program. However, the boundary conditions at the start of the heliocentric phase are determined as follows:

- 1) The vehicle is assumed to be initially placed at some geocentric radius (e.g., the earth's sphere of influence) with the departure time corresponding to this position assigned as the time when the low thrust propulsion system is activated. In short, the position of the vehicle at power-on time is specified instead of the actual launch time from an earth-bound station. This procedure has been shown to be an excellent approximation (Appendix C).
- 2) The orientation of the geocentric vehicle velocity at departure may be specified at will. This angle is determined from the sun-earth line at departure time in the plane of an initial parking orbit.
- 3) Orientation of the initial parking orbit is consistent with the initial inclination angle required for the heliocentric phase and a specified value of the departure energy parameter C_3 . The altitude of the initial parking orbit may be arbitrarily assigned.
- 4) Position of the vehicle at departure time in the heliocentric ecliptical reference frame of 1950.0 is adjusted to correspond to requirements set forth in 2 and 3 above.

Planetary positions are obtained by direct interpolation of coordinate information published by the HMS Stationery Office, using a five-point interpolation routine and a 10-day interval.

Boundary conditions at the termination of the heliocentric phase are computed from the amount of velocity decrement needed for pericenter insertion into a planetary orbit of specified dimensions. To avoid the effect of singularity in planetary perturbation at small planetary distances, computation of perturbations due to the destination planet is suppressed within the appropriate sphere of influence. Vehicle velocity relative to the destination planet at planet encounter thus represents the hyperbolic excess approach velocity.

Planetary encounter is considered to be achieved when all three components (radial, longitude, and latitude) of position deviation between the vehicle and destination planet are smaller than some specified amount (e. g., one planetary radius).

PROGRAM DESCRIPTION AND INFORMATION FLOW

The program consists of 17 subroutines listed in Table B-1, each with a brief remark regarding its functional characteristics, description of input data; their format is given in Table B-2.

TABLE B-1. LIST OF ROUTINES IN THE SEARCH PROGRAM

<u>Routine</u>	<u>Deck Name</u>	<u>Purpose of Routine</u>
MAIN	MAIN	Controls program flow
DATAIN	INPUT	Data input, initialization of control settings
RENITL	RENITL	Set up problem parameters (p , i , δ , T_L)
CINCON	C	Computes initial conditions
ETERAT	ETERAT	Controls integration of differential equations, search for zero radial relative velocity components, detailed trajectory output
BV	B	Computes position and velocity for elliptical motion
DE	DE	Computes perturbations for all planets, thrust, solar pressure force, and force summation
FDIFEQ	F	Computes differential equation set

Table B-1 (continued)

<u>Routine</u>	<u>Deck Name</u>	<u>Purpose of Routine</u>
DERIV	FAUX	Auxiliary subroutine for "F"
GRESET	G	Resets variables
TERP	INTERP	Computes planetary positions and generates interpolated quantities corresponding to zero radial relative velocity
PLANPV	PLANET	Computes planetary positions and velocities in cartesian coordinates
ROTMAT	ROTMAT	Performs coordinate rotation transformations
R2RMIN	R2RMIN	Determines trial solution for zero radial velocity component and generates correction to T_L for convergent solution in longitude
THREED	THREED	Generates corrections to T_L , i , and δ for convergent solution in latitude
TWOD	TWOD	Generates corrections to T_L and P for convergent solution in radial distance
PRNT	PRNT	Computes output quantities and controls output format

TABLE B-2. DESCRIPTION OF PROGRAM INPUT

General Information

- 1) Definition: Table refers to any subscripted variable
- 2) Input flow:
 - a) Data cards in Section I are fixed for a given run
 - b) Data cards in Section II are fixed for N-cases of data cards in Section III (see card 7, field 4)
 - c) Input of Section II with accompanying cases of input from Section III may be stacked without limit, but input from Section I must be sufficient to cover all cases including excursions in ranges of dates during search

Inputs

Section I - Basic Tables

- 1) Notes on card group numbers 2, 3, 4, and 5
 - a) Values of the quantities in field 1 must be nondecreasing
 - b) A minimum of three entries is required, the last card being an end-of-table marker with -1 in field 1
- 2) Notes on card group 6:
 - a) Julian date table, JD(j) of each planet must be non-decreasing and have equally spaced intervals
 - b) Order of inputs:
 - First set of tables must be for starting planet
 - Second set of tables must be for destination planet
 - Order of remaining planetary data tables is unimportant
 - c) Range of JD(j) must include at least three entries before minimum JD and at least two entries beyond maximum JD of interest

Table B-2 (continued)

Card No. 1 Format (10I4)

- Field 1: $N(\eta)$ = Number of entries in (I_{sp}, η) table, (maximum 20)
- Field 2: $N(W_o)$ = Number of entries in (C_3, W_o) table, (maximum 20)
- Field 3: $N(\Sigma)$ = Number of entries in ($\Delta V, \Sigma$) table, (maximum 20)
- Field 4: $N(K_r)$ = Number of entries in (r_v, K_r) table (maximum 100)
- Field 5: $N(JD)$ = Number of entries in Julian date tables (maximum 500)
- Field 6: $N(P1)$ = Number of planets (maximum 4, minimum 2)

Card group Nos. 2, 3, 4, and 5 Format (2E12.8)

- Card No. 2, Field 1: $I_{sp}(j)$ = specific impulse, seconds
- Field 2: $\eta(j)$ = power conversion efficiency
- Card No. 3, Field 1: $C_3(j)$ = injection energy parameter in $(\text{km/sec})^2$
- Field 2: $W_o(j)$ = injected weight, pounds
- Card No. 4, Field 1: $\Delta V(j)$ = retro-velocity increment, km/sec
- Field 2: $\Sigma(j)$ = weight ratio for orbit insertion
- Card No. 5, Field 1: $r_v(j)$ = heliocentric distance, AU
- Field 2: $X_r(j)$ = power coefficient

Table B-2 (continued)

Card group No. 6 Format (F8.1, F10.3, F8.4, F8.6, 2F14.7)

Field 1: JD(j) = Julian Date - 2,400,000.0

Field 2: l(j) = celestial longitude, degrees

Field 3: b(j) = celestial latitude, degrees

Field 4: r(j) = heliocentric distance, AU

Field 5: CD(j) = calendar date, year-month-day

Field 6: m(j) = ratio of sun's mass and planetary mass

Field 7: $\rho(j)$ = planetary radius, km

Section II - Flight Parameters

- 1) Notes on card group Nos. 8, 9, and 10
 - a) Arguments of these tables are assigned to field 2, and their values must be nondecreasing
 - b) A maximum of 100 entries is permitted in each table
 - c) A minimum of three entries is required, the last card being an end-of-table marker with -1 in field 2

Card No. 7 Format (10I4)

Field 1: N(γ) = number of entries in γ table

Field 2: N(δ) = number of entries in δ table

Field 3: N(K_T) = number of entries in K_T table

Field 4: N = control number: signifying the number of Section III cards to be run with Section II cards

Field 5: DIMEN = control for dimensionality: two for two-dimensional problems (no searches on i and δ); three for three-dimensional problems.

Table B-2 (continued)

Field 6: SMTRJ = control for output: one for summary output during search followed by a detail output for the final trajectory; two for trajectory output without search

Field 7: OPTPO = print out option: 0 for summary output of only runs that have converged in longitude; one for summary outputs of all runs

Field 8: INDEL = δ selection control, used only when SMTRJ = 2: one for $\pm\delta_3$ controlled by the position of vehicle relative to line of node; two for values specified in the δ -table; three for δ_3

Card group Nos. 8, 9, and 10 Format (2E12.8)

Card No. 8, Field 1: $\gamma(j)$ = thrust angle; in-plane, degrees

Field 2: $(T-T_L)_{\gamma}(j)$ = elapsed time; days

Card No. 9, Field 1: $\delta(j)$ = thrust angle; out-of-plane, degrees

Field 2: $(T-T_L)_{\delta}(j)$ = elapsed time, days

Card No. 10, Field 1: $K_T(j)$ = power control; 1.0 for on, 0.0 for off

Field 2: $(T-T_L)_T(j)$ = elapsed time, days

Card No. 11 Format (6E12.8)

Field 1: n_H = initial planetocentric distance, planetary radius units

Field 2: γ_H = angle measured from the sun-planet line to the planetocentric velocity vector

Field 3: ρ_{PO} = radius of parking orbit around the starting planet, km

Table B-2 (continued)

Field 4:	A	= effective solar array area coefficient, ft^2/kw at 1 AU
Field 5:	α_2	= control for suppressing computation of perturbation due to the destination planet (i. e., for $d_2 \leq \alpha_2 p_2$)
Field 6:	B^2	= square of threshold limit; B in planetary radius units

Card No. 12 Format (6E12.8)

Field 1:	I_{sp}	= specific impulse, seconds
Field 2:	n_p	= exponent for power degradation
Field 3:	α	= specific weight of propulsive system, lb/kw
Field 4:	h_A	= apocenter altitude of planetary orbit, km
Field 5:	h_p	= pericenter altitude of planetary orbit, km
Field 6:	δ_3	= out-of-plane thrust angle (used either as a trial value during search or for a specific trajectory)

Card No. 13 Format (6E12.8)

Field 1:	ΔT	= interval of integration, days
Field 2:	T_F	= maximum flight time, days
Field 3:	ΔT_{PO}	= print-out interval, days

Table B-2 (continued)

Section III - Run Specifications

Card No. 14 Format (6E12.8)

Field 1:	C_3	= injection energy parameter, (km/sec) ²
Field 2:	P_o	= total power, kw
Field 3:	k_p	= multiplying factor used for modifying P during search, a number slightly larger than 1.0
Field 4:	T_o	= power-on Julian date
Field 5:	i_{123}	= orbital inclination angle, used for specific trajectory run only
Field 6:	k_I	= limiting control for maximum orbital inclination angle, $0 \leq k_I \leq 1.0$

Computation within the program may be divided into three main phases:

- 1) Problem setup - For each set of in-plane thrust angles (δ) and injection energy parameter (C_3) inputs, a trajectory can be generated when the following four parameters are specified:
 - a) P_o = power level, at 1 AU
 - b) T_L = departure time
 - c) i = initial inclination angle between the planetary orbit plane and the trajectory plane
 - d) δ = out-of-plane thrust angle (represented by the angle of roll about the sun-vehicle line)

Initial conditions corresponding to these parameters are set up in the first block.

- 2) Trajectory computation - This is done in the second block. Monitoring of the difference between the heliocentric distance of the vehicle and the destination planet takes place after an elapsed time test. When a minimum is reached in this

difference, any difference in vehicle longitude and planet longitude will be corrected by applying a correction to the departure time T_L . Iteration in T_L is continued until the longitude difference becomes less than the specified limit (B). If no minimum is found at the end of the specified flight time, computation will commence with the next set of C_3 .

- 3) Parameter search phase - Corrections to both δ and i are generated to drive latitude differences and corrections to P are generated to drive radial differences separately to within the specified limit in this phase. Refinement in departure time is carried out simultaneously with these corrections. When all differences become less than the specified limit, a new control on flight time (T_{STOP}) is generated, and the printout control is set to 2. These controls, together with the current set of P_0 , T_L , i , and δ are fed back to the first phase to compute a final trajectory for detail printout. At the completion of this trajectory output, computation will restart with the next set of C_3 from the first phase.

Power profile as a function of heliocentric distance (r) is approximated by the expression

$$P(r) = P_0 K_r / r_v^n P$$

where r_v is in AU, K_r is a set of tabular inputs, and n_p is an input quantity.

OUTPUT FORMAT

Two output formats are provided: one gives a summary of results of consecutive iterations on trajectories and one gives a detailed trajectory printout at uniform intervals. These formats are displayed in Figure B-1.

In the summary format, run parameters are listed in a two-line heading. Tabular printout in this format consists of 14 quantities as follows:

T_L	= departure time
$T - T_L$	= flight time to reach the condition of zero radial relative velocity
W	= vehicle weight
WP	= nonpropulsive portion of vehicle weight
$VH2$	= hyperbolic excess velocity

SUMMARY

M1 = X.XXXXXXXXXX	M2 = X.XXXXXXXXXX	M3 = X.XXXXXXXXXX	M4 = X.XXXXXXXXXX		
NH = XXX.X	GH = XX.X	RPO = XXX.X	ALP = XX.X	ISP = XXXX.X	ETA = X.XXX
C3 = X.XXX	GA = XX.X	WO = XXXXX.X	NP = X.XXX	NA = XX.X	RP = XXXX.X
TL DATE	T-TL DAYS	W LBS	WP LBS	VH2 KM-S	DR
XXXXX.XXX	XXX.XXX	XXXXX.X	XXXXX.X	XX.XXX	XXXX.X
				DL	DB
				XXXX.X	XXXX.X
				IL DEGS	IE DEGS
				X.XXX	X.XXX
				DELTA DEGS	P KW
				XX.XXX	XXX.XX
					WORB LBS
					XXXXX.X
					ZETA DEG
					XXX.XX

TRAJECTORY

M1 = X.XXXXXXXXXX	M2 = X.XXXXXXXXXX	M3 = X.XXXXXXXXXX	M4 = X.XXXXXXXXXX		
NH = XXX.X	GH = XX.X	RPO = XXX.X	ALP = XX.X	ISP = XXXX.X	ETA = X.XXX
C3 = X.XXX	GA = XX.X	WO = XXXXX.X	NP = X.XXX	NA = XX.X	RP = XXXX.X
P = XXX.XX	IL = XX.X	DEL = XX.X			
T DATE	T-TO DAYS	W LBS	WP LBS	VH2 KM-S	R MKM
XXXXX.XXX	XXX.XX	XXXXX.X	XXXXX.X	XX.XXX	XXX.XXX
				L DEG	B DEG
				XXX.XXX	XX.XXX
				CL-AN DEG	EVS DEG
				XXX.XX	XX.XXX
				D1 MKM	D2 MKM
				XXX.XXX	XXX.XXX
					ZETA DEG
					XXX.XXX
					WORB LBS
					XXXXX.XX

Figure B-1. Printout Format

DR, DL, DB = components of vehicle-planet distance in the radial, longitude, and latitude direction in planetary radii units

IL = initial inclination angle from the planetary orbit plane to the trajectory plane

IE = inclination of the initial parking orbit to the planetary orbit plane

DELTA = out-of-plane thrust angle

P = power level required (at 1 AU)

WORB = weight in orbit

ZETA = angle between the sun-planet line and the asymptotic approach direction at encounter

The quantities printed out correspond to the time when the condition of zero radial relative velocity is satisfied. The last two quantities are printed out only when the longitude difference (DL) is less than the specified limit.

In the trajectory format, a third line is added to the heading print to display the values of power level, initial inclination angle, and out-of-plane thrust angle associated with the final trajectory. The tabular format again displays 14 quantities as follows:

T = time

T-T₀ = elapsed time from departure time

W	= vehicle weight
WP	= nonpropulsive portion of vehicle weight
VH2	= vehicle velocity relative to the destination planet
R	= heliocentric distance of the vehicle
L	= vehicle longitude
B	= vehicle latitude
CL-AN	= clock angle (Canopus-earth system)
EVS	= earth-vehicle-sun angle
D1	= distance from earth to vehicle
D2	= distance from vehicle to the destination planet
ZETA	= angle between the sun-planet line and the planetocentric vehicle velocity
WORB	= weight in orbit

APPENDIX C. EFFECTS OF TIME OF INITIATION OF LOW THRUST ACCELERATION

The initiation of the integration of the trajectory in heliocentric coordinates can be at any selected distance from the earth (see Appendix B). If a very small value of geocentric radius were chosen as the initiation point of the integration, the earth perturbation terms would be large, and it would therefore be necessary to use a very small step size during the initial part of the integration in order to maintain accuracy. For this reason the computation time required for an earth-Mars trajectory approximately doubles when the initiation of the integration of the trajectory is changed from 144 to 36 earth radii. For the simulation results shown in this report, the start of the integration (and low thrust propulsion system initiation) has been fixed at 144 earth radii as a compromise between trajectory accuracy and required computational time. The effects of this assumption are shown in this appendix.

Simulations in which the trajectory integration is started at 36 earth radii have been made for both Mars and Jupiter trajectories. Coast periods have been used at the start of these test trajectories to simulate engine ignition at geocentric radii from 36 to 144 earth radii. These results are shown in Figures C-1 and C-2 for Mars and Jupiter, respectively, and indicate a linear variation in net weight with radius at low thrust initiation. Since initiation of the integration inside 36 earth radii is unwieldy computationally, weight changes in this region have been extrapolated. The following analysis demonstrates that a linear extrapolation is valid.

Assume a body is moving on a hyperbolic trajectory in an inverse square field, it can easily be shown that if constant acceleration is directed along the instantaneous velocity vector

$$\frac{dE}{dR} = R_p a \sqrt{\frac{R}{R-1}} \sqrt{\frac{1}{1 + \frac{1}{R + (2/\epsilon - 1)}}} \quad (1)$$

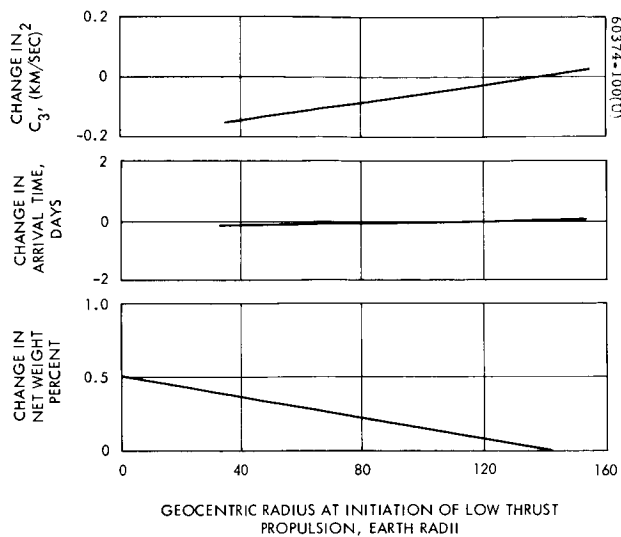


Figure C-1. 1971 Mars Orbiter, Change in C_3 , Change in Arrival Time, and Change in Net Weight versus Geocentric Radius at Initiation of Low Thrust Propulsion

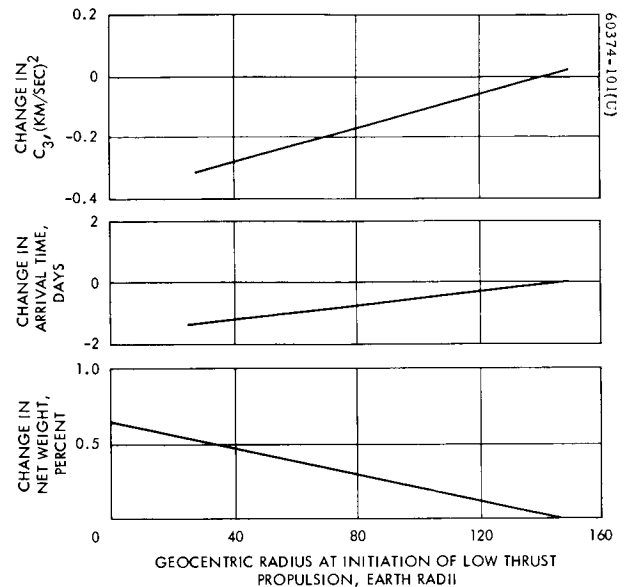


Figure C-2. Jupiter Flyby, Change in C_3 , Change in Arrival Time, and Change in Net Weight versus Geocentric Radius at Initiation of Low Thrust Propulsion

where

E = specific orbital energy, $(\text{km/sec})^2$

R_p = radius of perifocus, km

a = acceleration, km/sec^2

ϵ = instantaneous orbital eccentricity

R = radius in units of perifocus radius

The energy change due to a continuous acceleration may then be found by integrating Equation 1. If acceleration is initiated at perifocus the energy is

$$E = R_p a \int_1^R \sqrt{\frac{R}{R-1}} \sqrt{\frac{1}{1 + \frac{1}{R + (2/\epsilon - 1)}}} \quad (2)$$

Equation 2 can be integrated in closed form for the limiting cases of

$$\epsilon = 1 \quad (3)$$

$$\epsilon = \infty$$

Since the limits of Equation 3 give limiting values of Equation 1, the integration of these limiting values (Equation 2) will give the maximum variation in the energy addition. The limits of Equation 2 are plotted in Figure C-3, and the energy addition is not only shown to be a remarkably linear function of geocentric radius, but also essentially independent of orbital eccentricity.

For the low thrust trajectories under consideration, the increase in net weight with earlier low thrust propulsion system initiation is directly due to the increase in energy with this change, and the linear extrapolation of net orbit weight is therefore valid. Since the above analysis is completely general and valid near any attracting body, the demonstrated linear variation of performance with radius at which propulsion is initiated also applies to low thrust termination at the target planet.

On an actual trajectory it will be necessary to delay the initiation of low thrust propulsion system operation for some period of time. Reasons for this include not only possible earth eclipse, mechanical limitations, and residual atmosphere effects, but also a possible requirement for a built-in coast period to accommodate off-nominal launch vehicle performance (see guidance and control discussion). The required delay is probably at least 1 day, which places the thrust initiation outside 36 earth radii.

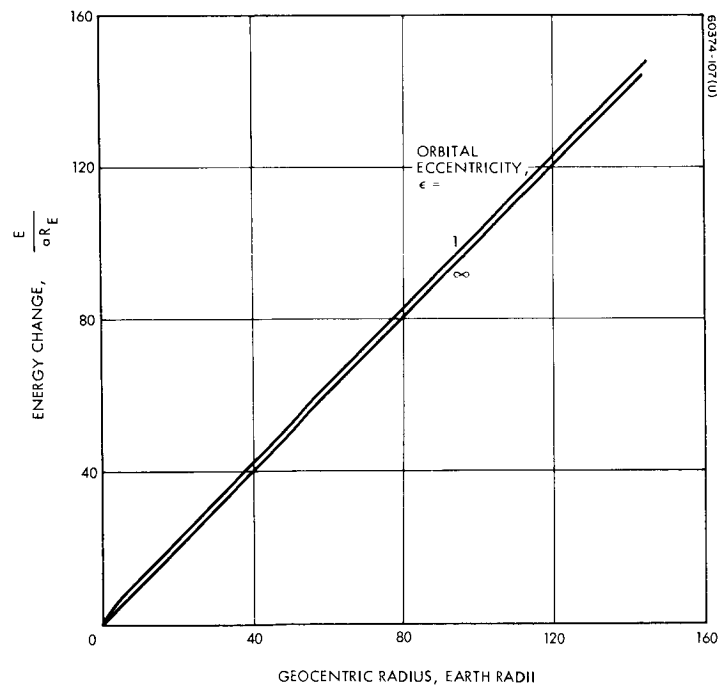


Figure C-3. Energy Change, E/aR_E , versus Geocentric Radius, Earth Radii

Figures C-1 and C-2 show that the total effect of changing the low thrust initiation time is very small, which demonstrates that the reduction of simulation cost by starting the integration at 144 earth radii has not caused any significant inaccuracy in the results. Although the performance numbers obtained with low thrust ignition at 144 earth radii could be increased to correspond to an earlier initiation time, the size of this change is so small that it would perhaps be better to consider the capability of thrust initiation inside 144 earth radii as a slight conservatism in the computation. A similar detailed analysis at the target planet would demonstrate an analogous performance insensitivity because the kinematic problem is similar, and these conclusions therefore also apply to thrust termination at the target planet.

The analysis discussed above shows that the geocentric and planetocentric phases of a low thrust interplanetary mission can be approximated with closed form expressions to an accuracy that is more than adequate for performance studies. This demonstrates conclusively that any low thrust interplanetary trajectory optimization problem can be validly analyzed by considering the geocentric and planetocentric effects as algebraic boundary conditions in a detailed simulation of the heliocentric phase.

APPENDIX D. SOLAR PANEL POWER VERSUS DISTANCE FROM SUN

INTRODUCTION AND SUMMARY

A solar electric propelled vehicle requires a very careful estimate of solar power available from the solar panels for purposes of trajectory analysis and engine system design. Accordingly, this analysis was prepared.

The results show that solar power available from silicon solar cells as a function of solar distance peaks in the neighborhood of 0.7 AU. The curve is nearly linear between 0.7 and 1.4 AU. The ratio of power at 0.7 AU to power at 1.0 AU is approximately 1.2.

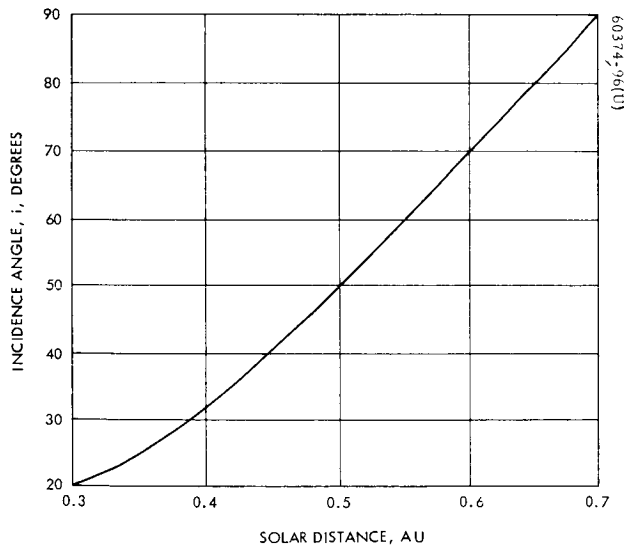


Figure D-1. Angle of Incidence for Maximum Power versus Solar Distance

Power can be maintained constant from 0.7 AU to nearly 0.3 AU by reducing the angle of solar incidence according to the curve of Figure D-1.

Derivation of the Temperature Equation

Power output of a solar panel depends on the panel temperature and on solar cell power versus temperature characteristics. Therefore, it is necessary to first find the relation between the temperature of the solar panel and distance from the sun. To do so, assume the temperature gradient across the panel is small. Let:

α_c = absorptivity of a solar cell

ϵ_c = emissivity of a solar cell

e_p = emissivity of the dark side of the panel

t = temperature, °K

σ = Stefan-Boltzmann constant = 5.735×10^{-12} watts/cm²/1°K⁴

i = angle of solar panel to sunline

w = energy, watts

η = solar cell efficiency

r = heliocentric radius, AU

By the inverse square law, the incoming radiation is

$$w_i = \frac{w_{\odot}}{r^2} \sin i \quad (1)$$

The energy absorbed by the panel and converted to heat is the total energy absorbed minus that converted to electrical power, i. e.,

$$\begin{aligned} w_a &= w_i \alpha_c (1 - \eta) \\ &= (1 - \eta) \frac{w_{\odot}}{r^2} \alpha_c \sin i \end{aligned} \quad (2)$$

The energy radiated from the panel is

$$w_{\text{rad}} = \epsilon_p \sigma T^4 + \epsilon_c \sigma T^4 \quad (3)$$

(dark side) (Sun side)

Balancing radiations and solving for panel temperature

$$T = \left[\frac{(1 - \eta) w_{\odot}}{\sigma r^2} \frac{\alpha_c}{\epsilon_p + \epsilon_c} \sin i \right]^{1/4} \quad (4)$$

Discussion of the Constants

Since maximum power output is rather sensitive to the absorptivity and emissivity constants and to reflectivity changes at small angles of incidence, comments on selection of values for the constants of the equation are in order.

The emissivity of the dark side of the panel, ϵ_p , quoted by various authorities is between 0.90 and 0.96; $\epsilon_p = 0.95$, is used here.

The absorptivity of a cell, α_c , varies from 0.80 to 0.94, depending mainly on passband and amount of filter in the cover glass. Although α_c of 0.7 can be obtained for high temperature (small heliocentric radius) applications with some degradation of efficiency, the value $\alpha_c = 0.82$ is used here mainly because of proton and electron damage. This damage varies with the inverse square law, which, in combination with the long exposure times associated with high temperature missions, will cause a power degradation and probable cancellation of the extra power available from the use of the special filters. The emissivity of the cell, ϵ_c , also varies with the filter, but not so widely, and $0.82 < \epsilon_c < 0.87$ appears to hold. The upper value is used for the calculations. Thus, the important constant is

$$\frac{\alpha_c}{\epsilon_p + \epsilon_c} = 0.45$$

The temperature of a panel then is from Equation 4,

$$T = \frac{326[(1 - \eta)(\sin i)]^{1/4}}{\sqrt{r}} \quad (5)$$

In the case of a mission to $r < 0.7$ AU the angle of incidence, i , can be decreased to maintain a temperature such that constant power output is available. However, reflectivity changes with i causing a decrease in power output of 2 percent at $i = 30$ degrees to 20 percent at $i = 10$ degrees. It may be that α_c decreases sufficiently due to changing reflectivity characteristics in this region to permit maximum power output down to $i \approx 10$ degrees. Data is not yet available to indicate this.

It should be noted that for small angles of incidence, parts of any protrusions in a panel such as spars are likely to be much hotter, so that solar cells in the neighborhood of these hot spots will be inoperative.

Also, it should be noted that from a vehicle control viewpoint, it is not feasible to try to control panel attitude within the tolerances implied at closer solar range.

Cell power output is then obtained by iterative use of Figure D-2 with Equation 5. The curve of Figure D-2 is a typical relation of maximum power output versus temperature for silicon solar cells, and the efficiency factor, η , can be obtained from it. At both high and low temperature extremes η must be modified by the power versus illumination intensity relation shown (typically) in Figure D-3.

Results

Figure D-2 shows the relation between power output and temperature of silicon solar cells, based on data obtained from Hoffman Electronics Corporation. The curve is based on a 100 percent output at 25 to 28°C. The

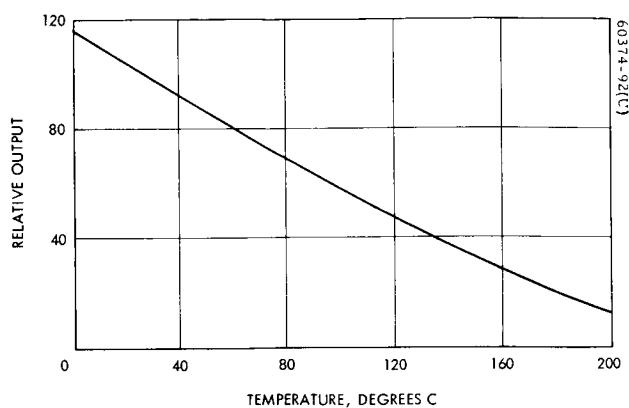


Figure D-2. Temperature Characteristics of N/P Solderless Sintered Ti-Ag Solar Cells

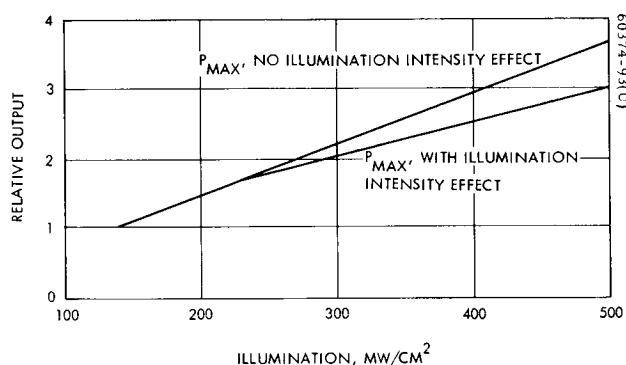


Figure D-3. Solar Cell Output versus Illumination Intensity

assumed. The curve is normalized to unity power output at 1 AU. The power peak at 0.7 AU and rapid power reduction at closer sun distances should be noted. This curve should be compared with the power curve actually used in the trajectory analysis shown in Figure A-4 of Appendix A.

Figure D-1 shows the incident angle i related to distance from the sun for maximum power. The curve includes the effects of illumination intensity of Figure D-3, but does not include the change in reflectivity with angle of incidence.

output can be improved perhaps by 10 percent at the higher temperatures by use of low series resistance cells.

Figure D-3 is a relation between cell output and illumination intensity at constant temperature. This data also was obtained from Hoffman Electronics Corporation.

Figure D-4 is a graph of panel temperature versus distance from the sun for $i = \pi/2$ as computed from Equation 5. The dashed branch of the curve is for the condition of maintaining the panel temperature, by variation of angle of incidence, that is necessary to hold constant power. Cell output data is not available at temperatures corresponding to $r > 1.5$ AU; hence, the curve is considerably in doubt beyond 2.0 AU. The changes in trajectory energy in this region have small effect on payload on a Jupiter mission, however.

Figure D-5 combines the data of Figures D-2 and D-4 to show solar panel power output variation with solar distance for $\theta = 90$ degrees. No cell degradation due to the intense radiation at the small solar distances is

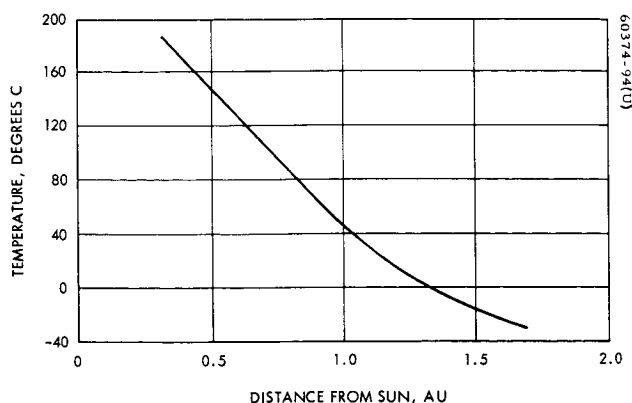


Figure D-4. Solar Panel Temperature versus Solar Distance

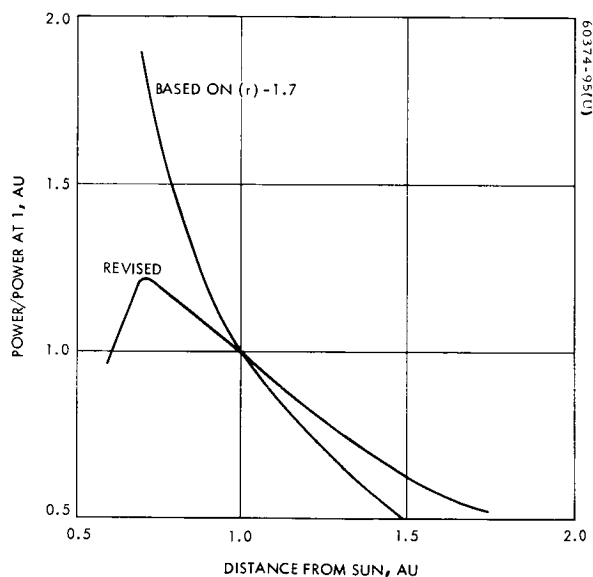


Figure D-5. Power versus Solar Distance

Conclusions

A linear relation of the form $k_1 - k_2 r$ holds for solar cell power between 0.7 and 1.4 AU. For $r < 0.7$, power falls off rapidly unless the solar panel plane is programmed to rotate according to the astronomical unit angle of incidence relationship approximating Figure D-1.

Power can be maintained at 1.2 times power at 1 AU from 0.7 AU into approximately 0.3 AU, including the effects of variation of reflectivity at small angles of incidence.

The $(r)^{1.7}$ relation used in previous studies should be replaced by the linear relation shown above. This has the net effect of some 10 percent for a Mars trajectory. The effect on the Jupiter mission has not been studied, but the integrated energy differential between the 1.7 and the linear relation should show a significant improvement in payload resulting from use of the new data.

The minimum solar range at which solar electric propulsion can be employed is in the neighborhood of 0.3 AU.

APPENDIX E. LIQUID MERCURY CATHODE ENGINE DESIGN

A major thermal consideration in the design of systems for space vehicles is the ability to withstand the temperature rise associated with exposure to direct solar incidence. If a system design is such that it cannot tolerate unlimited solar exposure, a constraint is said to exist. An analysis of the degree to which this constraint exists for a liquid mercury cathode engine as well as its affect on engine design will be presented in this appendix.

Assume that the cathode heat sink is a surface of low solar absorptivity and high thermal emissivity. Such a surface has a space temperature T_s of 12°C , in direct sunlight at 1 AU. (Space temperature is defined as that temperature which exists in the absence of any other heat loads.) The increase in operating temperature (ΔT) due to exposure to the sun is found by

$$(T + \Delta T)^4 = T^4 + T_s^4 \quad (1)$$

where T is the "shaded" operating temperature. This relation is plotted in Figure E-1, for $T_s = 12^\circ\text{C}$. Figure E-1 clearly demonstrates that the higher the operating temperature of the heat rejection surface the less the sensitivity to the additional heat load due to the sun. If, for a specific mission, the closest the vehicle comes to the sun is 1 AU, then these results indicate the maximum temperature increase to be experienced.

Consider now the design of a thruster that employs a high temperature liquid mercury cathode and eliminates the auxiliary heat rejection system associated with the low temperature cathodes discussed in the 1965 Final Report.* Figure E-2 shows that the proposed design for the high temperature cathode is a simple modification of the original design presented in Reference 1, Section 4. The changes consist of: 1) heat shielding over the entire internal surface area of the thruster, and 2) an engine shell fabricated from aluminum. (The original design had shielding only between anode and shell and the shell was fabricated entirely of magnetic material.)

*"Solar Powered Electric Propulsion Spacecraft Study," Final Report, prepared for JPL by Hughes under Contract No. 951144, December 1965.

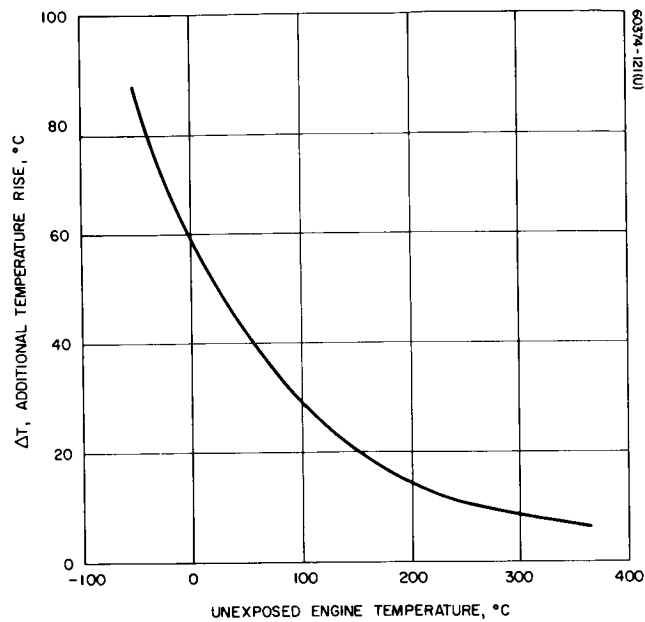


Figure E-1. Liquid Mercury Cathode Temperature Increase due to Solar Exposure at 1 AU

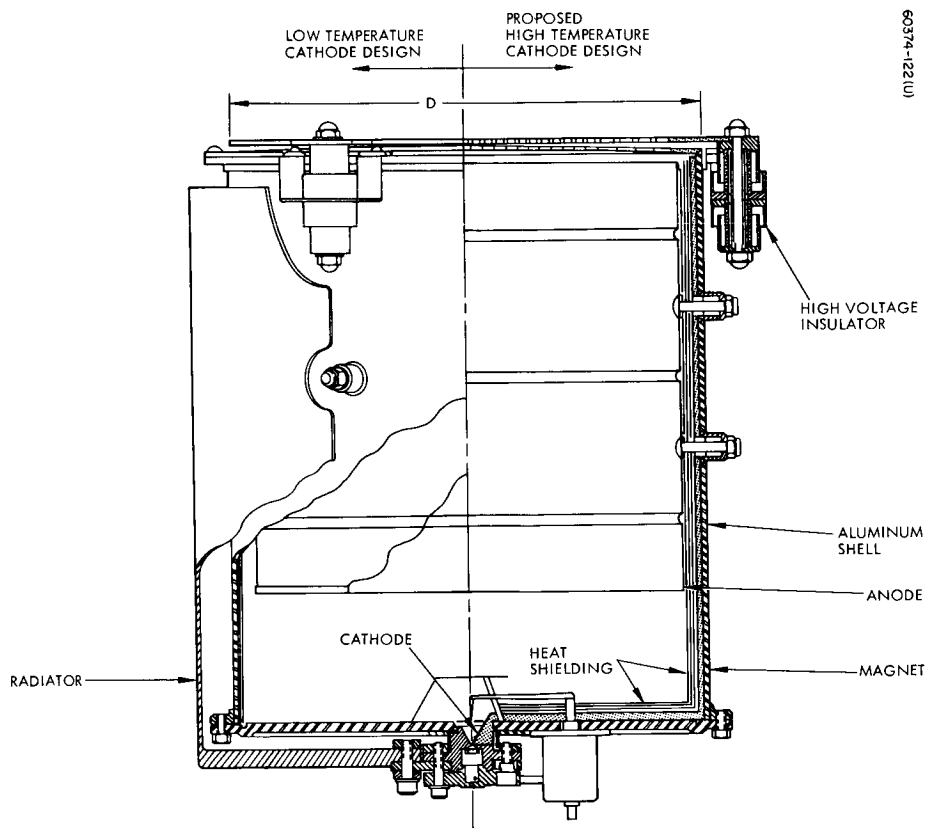


Figure E-2. Proposed Thruster Design to Utilize Height-Temperature Liquid-Mercury Cathode

Since the proposed design depends on the aluminum shell for the cathode heat sink, internal heat shielding is imperative. Without heat shielding, thruster temperatures are as shown in Figure E-3. A freely radiating thruster runs at 300°C , while a spread-out type array may raise this to 400°C . (A close-packed array results in intolerable temperatures, if only from the standpoint of the permanent magnet, and hence will not be considered.) Thus, in order for the aluminum shell to provide a sink temperature at something lower than 300°C , shielding is required. Thruster temperature versus shielding effectiveness is shown in Figure E-4. The

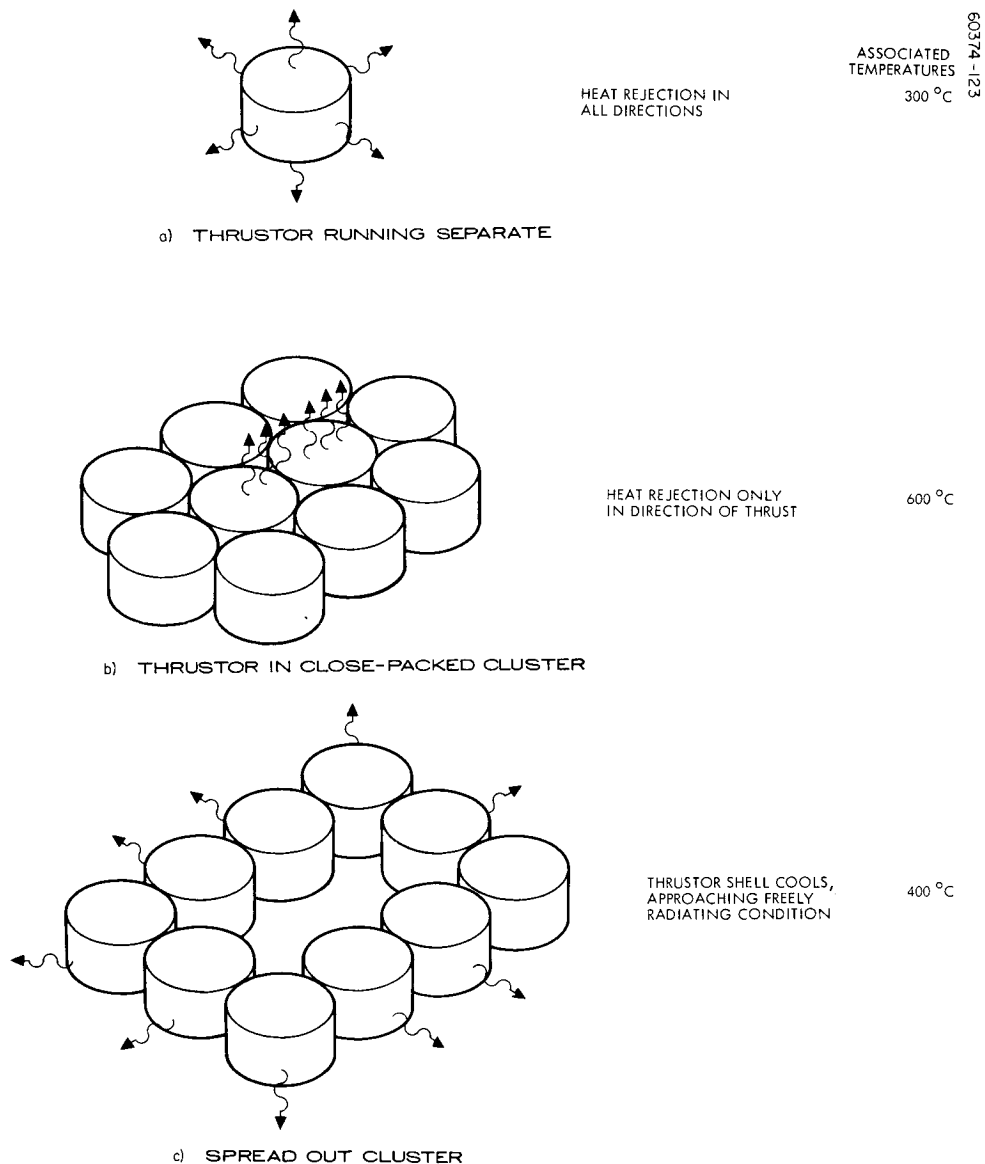


Figure E-3. Thermal Considerations in Clustering of Thruster Modules

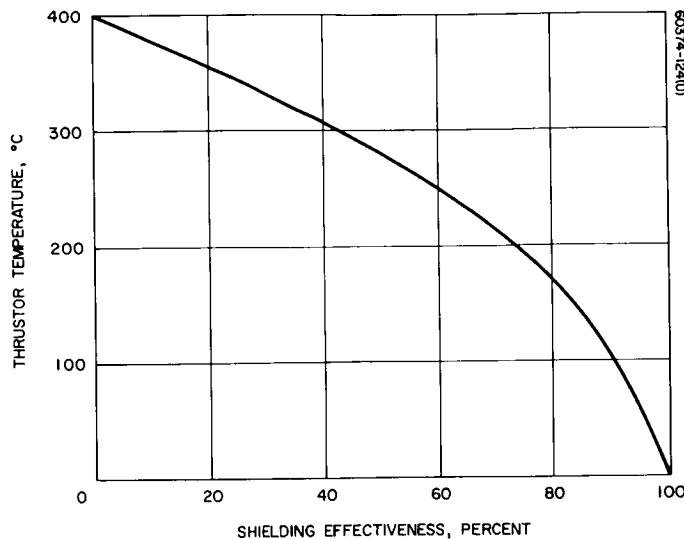


Figure E-4. Thrustor Temperature versus Shielding Effectiveness

results are based on an unshielded temperature of 400°C. From practical considerations (weight and volume of shielding), it is unlikely that above 90 percent effective shielding can be expected. Thus, it must be assumed on this basis alone that cathode sink temperature will be above 100°C. Moreover, it has already been established that high sink temperatures (as close to 300°C as possible) are desirable from the standpoint of insensitivity to solar radiation. Therefore, these two considerations (shielding ability and thermal constraint) do not conflict, they both indicate the requirement for high sink temperatures.

One final consideration remains — engine weight. The higher the sink temperature, the less temperature drop can exist between cathode and sink. To minimize the temperature drop, more aluminum is required as a heat conduction path. Figure E-5 shows the weight of aluminum required to provide the corresponding sink temperatures for a 20-cm engine. For example, in order to raise the sink temperature from 250° to 275°C requires a 100 percent increase in aluminum weight. Thus, although a high temperature sink minimizes the thermal constraints (i.e., solar incidence effects), it also increases the engine weight.

The choice of sink temperature becomes more clear-cut when a second curve is superimposed on Figure E-5. This curve shows the added weight (estimated) for the degree of shielding required to produce the indicated engine temperatures. Because of this shielding weight requirement, there appears to be no weight advantage in operating the sink temperature below 200°C. Figure E-5 also shows that the penalty becomes prohibitive above 250°C. Hence, the range of selection for engine (sink) temperature is narrowed to between 200° and 250°C.

The weight increase associated with these temperatures (see Figure E-5) is approximately 2 pounds. The nominal weight of a 20-cm engine is 8 pounds (with a permanent magnet shell). To conclude that the final engine weight is now 10 pounds is probably conservative. Engine redesign, utilizing the modifications of the aluminum shell, etc., could possibly eliminate some of the weight of the old permanent magnet shell. Latest design studies (August 1966) indicate that engine weight (for a 20-cm, liquid-Hg cathode thruster may be on the order of 7 pounds (including aluminum shell, magnets, and shielding).

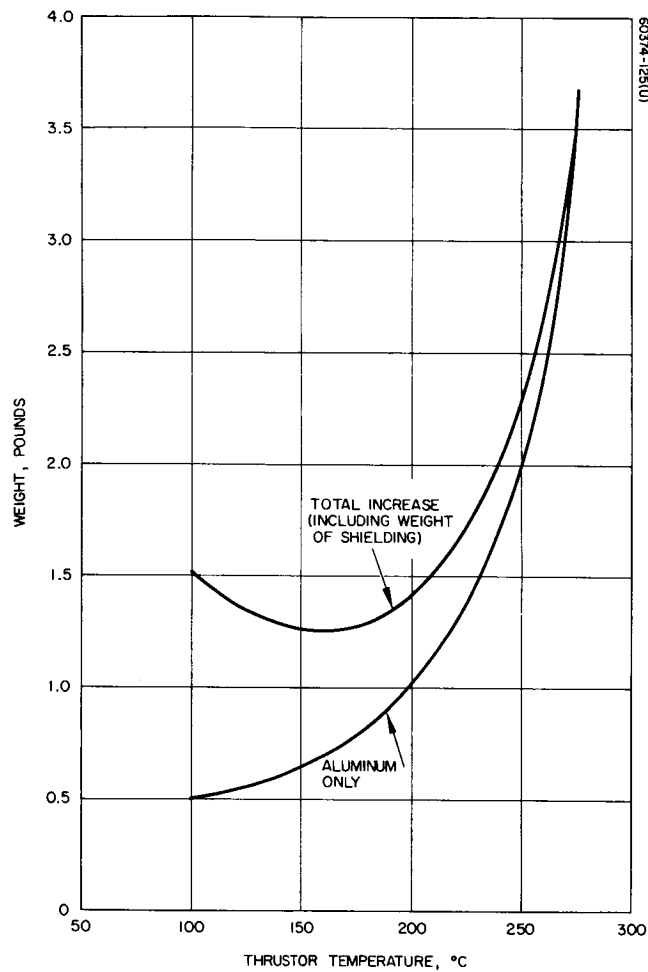


Figure E-5. Thrustor Weight Increase
versus Thrustor Temperature

To satisfy the requirements of this study, the above analysis for the 20-cm engine was generalized for engines of any size. It was found that the upper limit of engine temperature (250°C) is valid for engines of any size. The corresponding aluminum engine shell weight, for thrusters up to 40 cm diameter, is shown in Figure E-6.

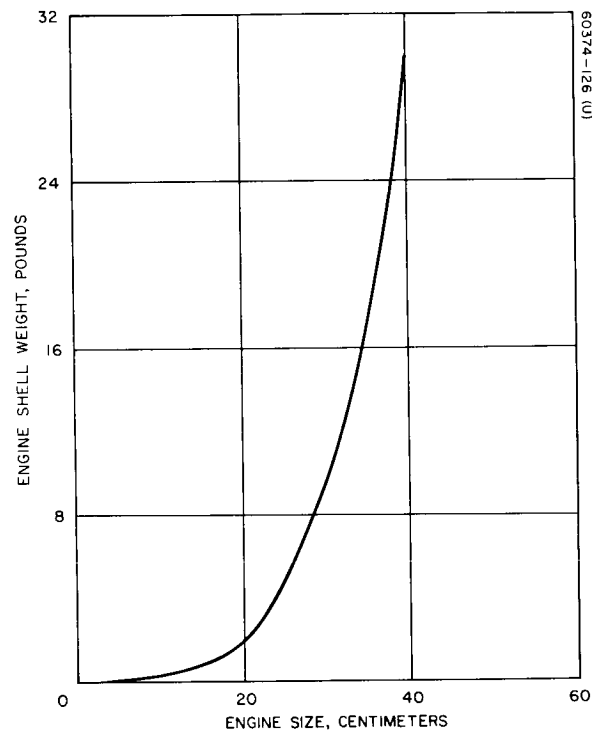


Figure E-6. Liquid Mercury
Cathode Engine Shell Weight
versus Engine Size

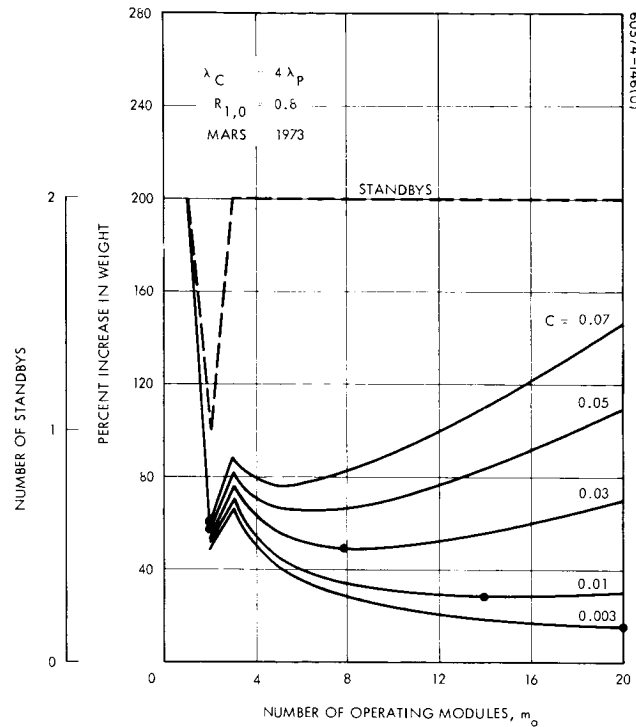


Figure F-2. Weight Penalty to Obtain 0.99 Reliability

The optimum number of operating and standby modules (for 1973 Mars mission) are shown in Figure F-1 as a function of the reliability of an unmodularized system. These curves are for $\lambda_C = 4\lambda_P$ and various values of C . Notice that the optimum number of operating modules is quite sensitive to the parameter C , and generally increases with decreasing C . Because modules are counted as standbys when they are turned off, the optimum number of standbys is relatively insensitive to the number operating and to $R_{1,0}$.

The effects of C , $R_{1,0}$, and λ_C/λ_P on the percent increase in weight, G_{m_o, n_o} , are shown in Figures F-2 through F-4, respectively. Figure F-2 shows that after an initial drop the weight penalty remains approximately constant for small values of C . Therefore, for systems where C is small, little penalty is paid when large numbers of modules are employed. As C increases, however, the curvature at the minimum increases and a more serious weight penalty is incurred if the optimum module size is not employed. Figure F-3 presents the weight penalty for increasing the system reliability to at least 0.99 as a function of the initial number of operating modules for various values of $R_{1,0}$. A general conclusion indicated by these curves is that their minima, i.e., the optimum number of modules operating, is relatively insensitive to $R_{1,0}$. The sensitivity of weight penalty on the ratio λ_C/λ_P is shown in Figure F-4. Although the weight penalty is dependent on λ_C/λ_P , variation of this parameter has almost no effect on the location of the optimum m_o point.

APPENDIX F. SYSTEM RELIABILITY - WEIGHT STUDY

The results of a computer study to determine the optimum thruster module from a weight-reliability standpoint are given in this appendix. Although the study included all the possible combinations of values of the parameters $R_{1,0}$, λ_c/λ_p , C , and $\bar{R}$, only a portion of the results will be presented. These results are summarized in Figures F-1 through F-4 for the 1973 Mars mission, Figures F-5 through F-8 for the 1977 Mars mission, and Figures F-9 through F-12 for the Jupiter mission.

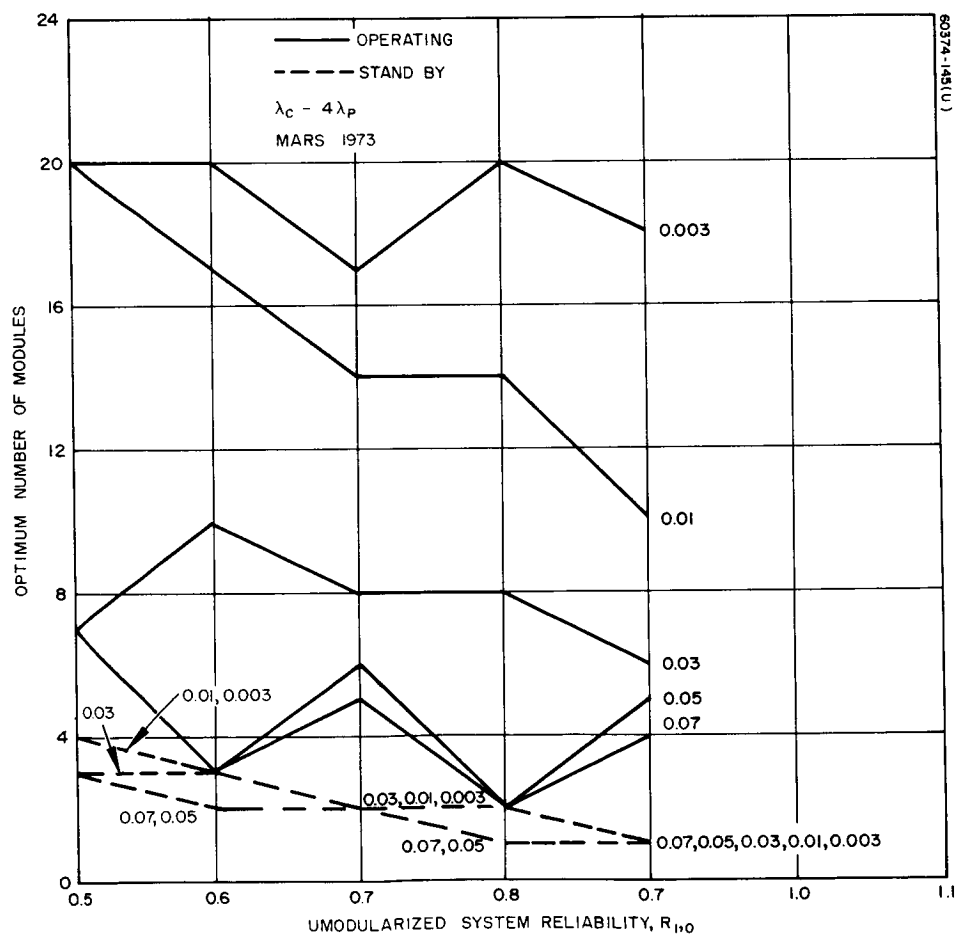


Figure F-1. Optimum Number of Modules to Obtain 0.99 System Reliability

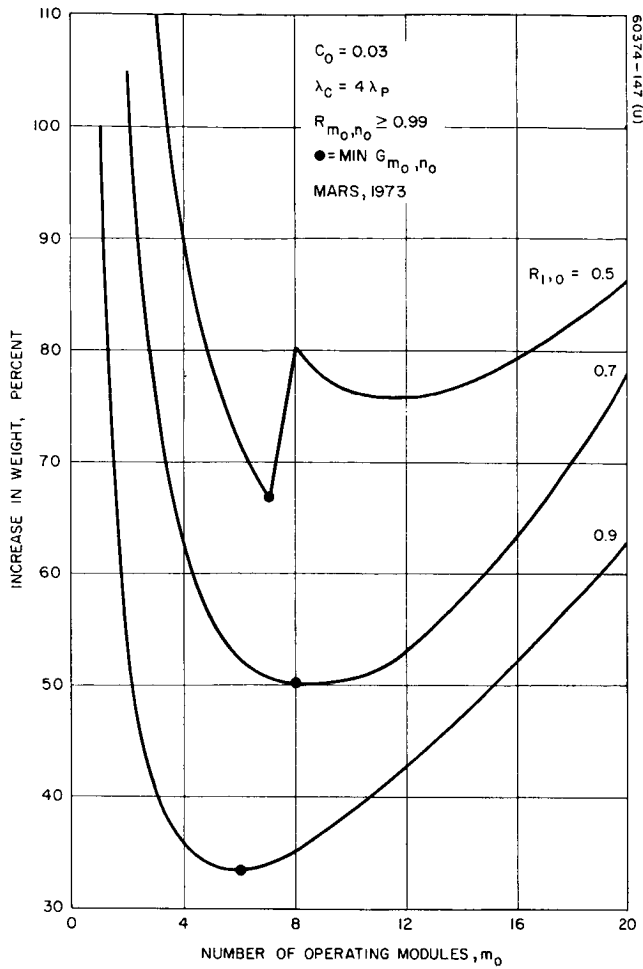


Figure F-3. Effect of $R_{1,0}$ on Weight Penalty

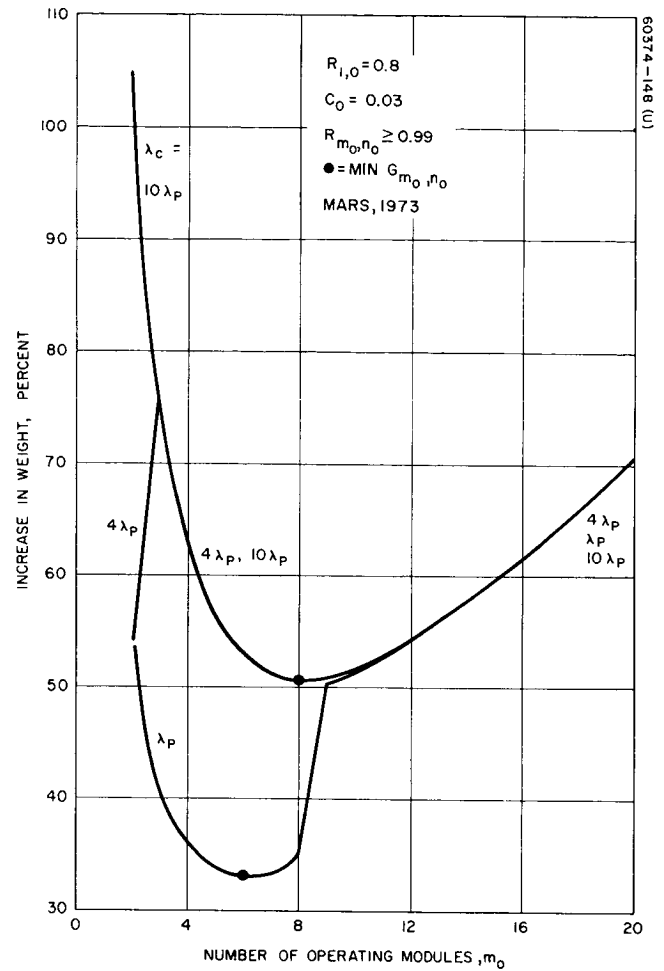


Figure F-4. Effect of λ_c / λ_p on Weight Penalty

In summary, from Figures F-1 through F-12 it can be concluded that for all three model missions the optimum number of modules operating has a significant dependence on the parameter, C , and has almost no dependence on the parameters λ_c / λ_p and $R_{1,0}$.

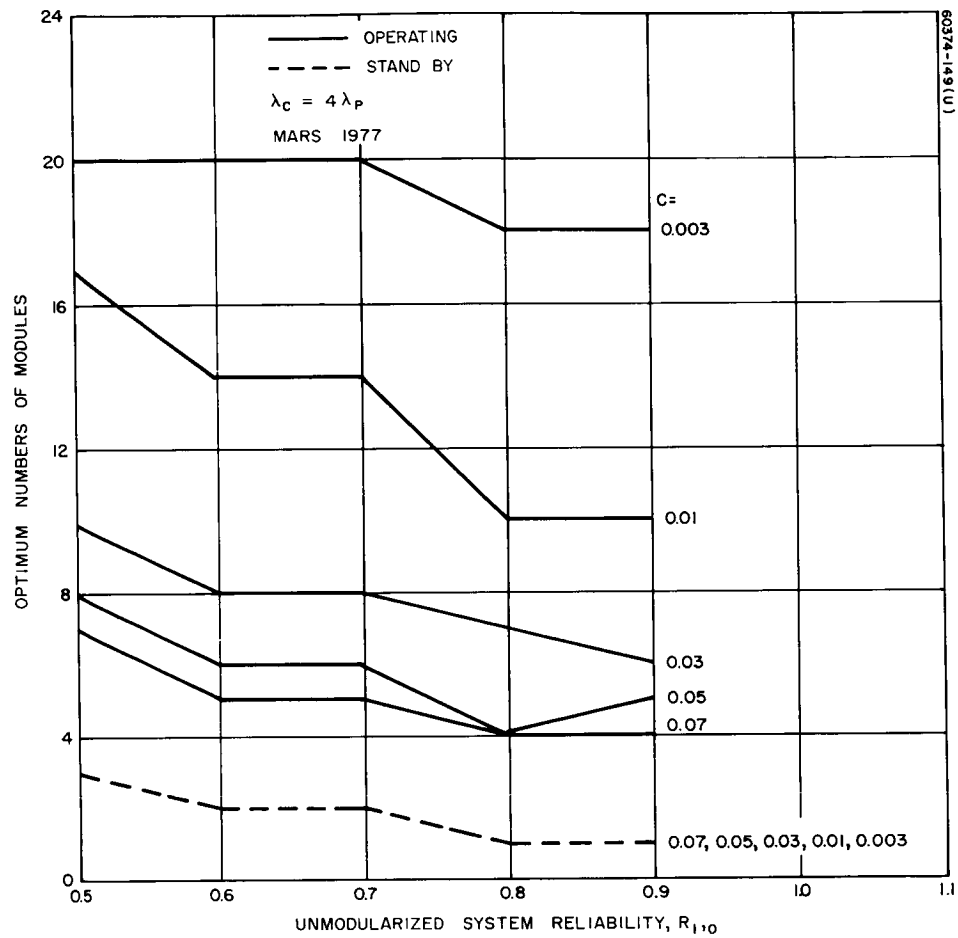


Figure F-5. Optimum Number of Modules to Obtain 0.99 System Reliability

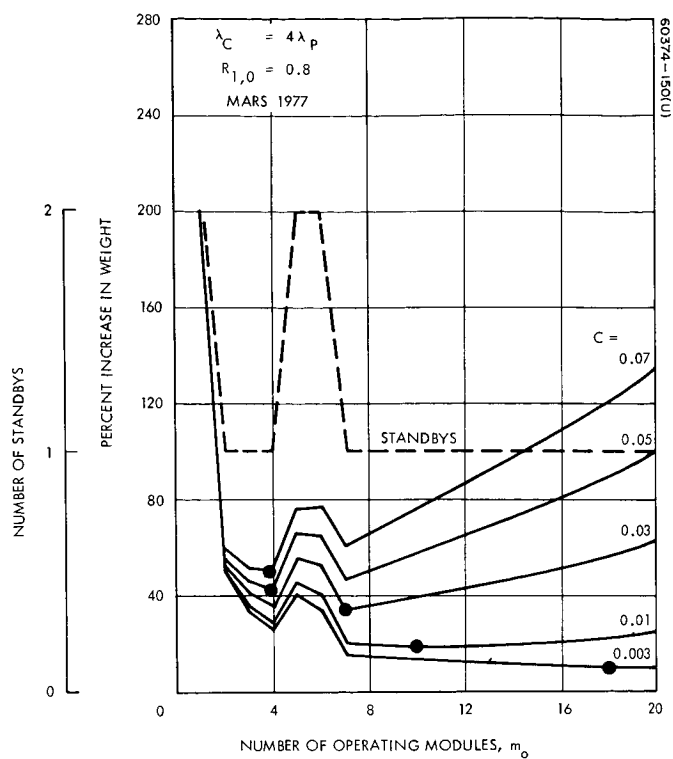


Figure F-6. Weight Penalty to Obtain 0.99 System Reliability

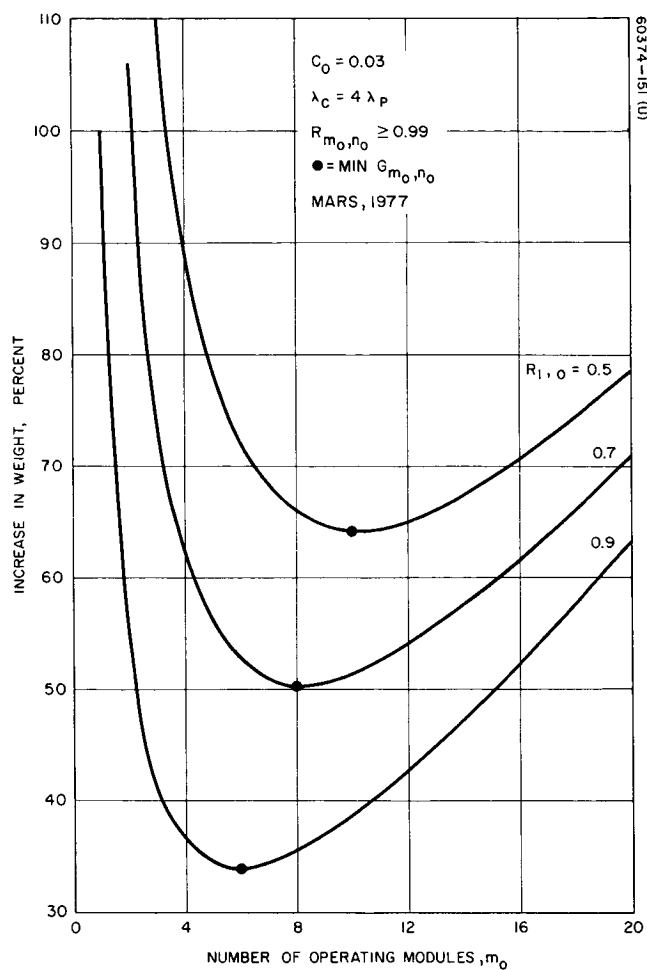


Figure F-7. Effect of $R_{1,0}$ on Weight Penalty

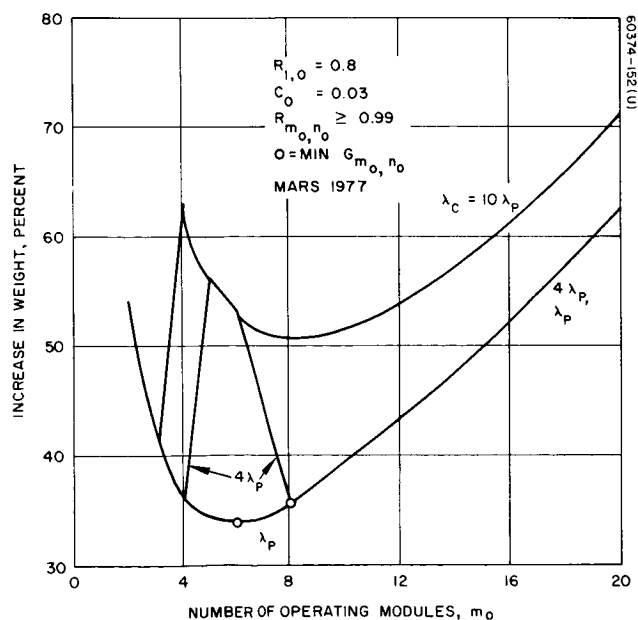


Figure F-8. Effect of λ_C/λ_P on Weight Penalty

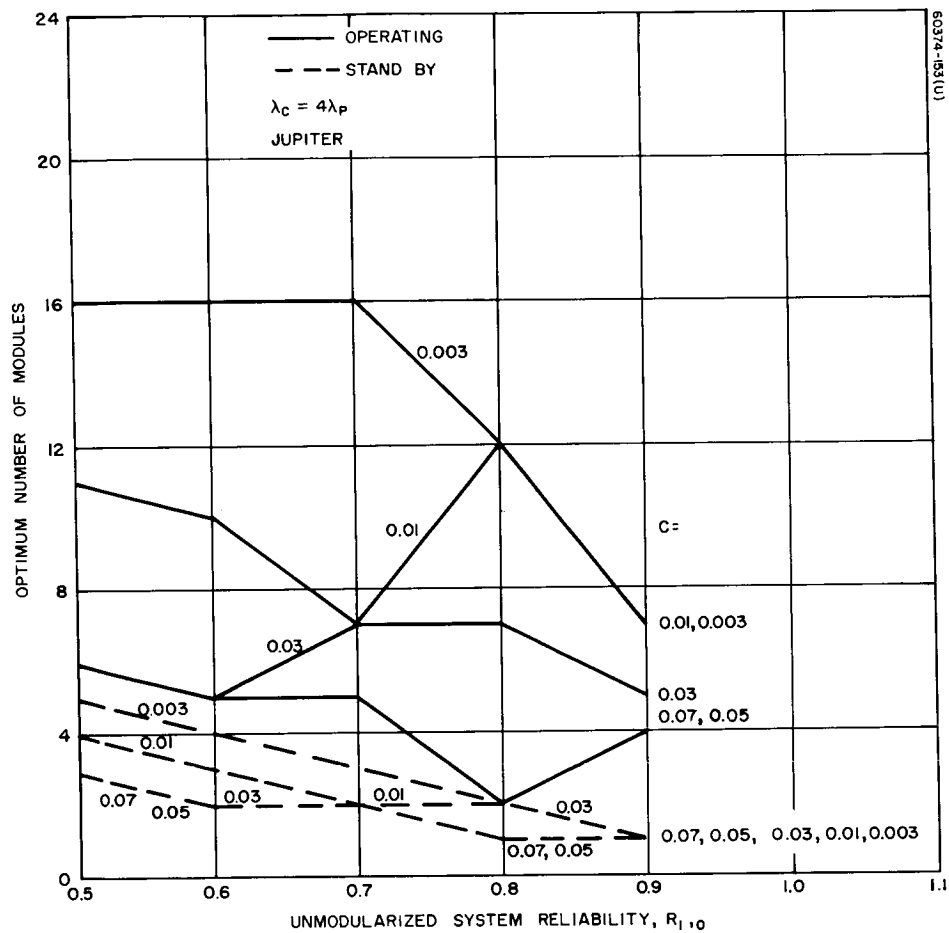


Figure F-9. Optimum Number of Modules to Obtain 0.99 System Reliability

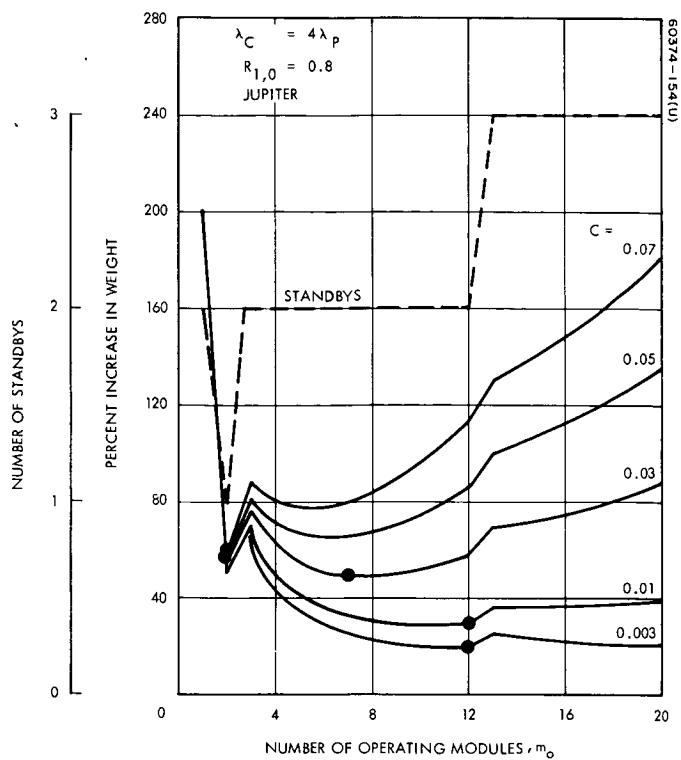


Figure F-10. Weight Penalty to Obtain 0.99 System Reliability

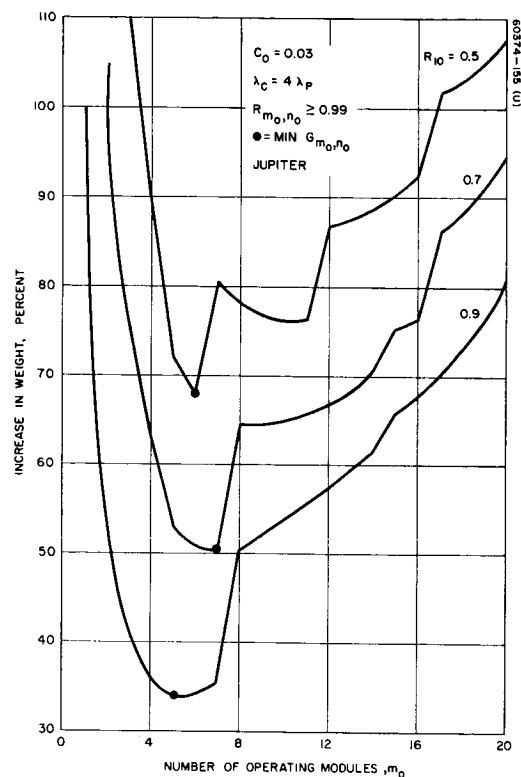


Figure F-11. Effect of $R_{1,0}$ on Weight Penalty

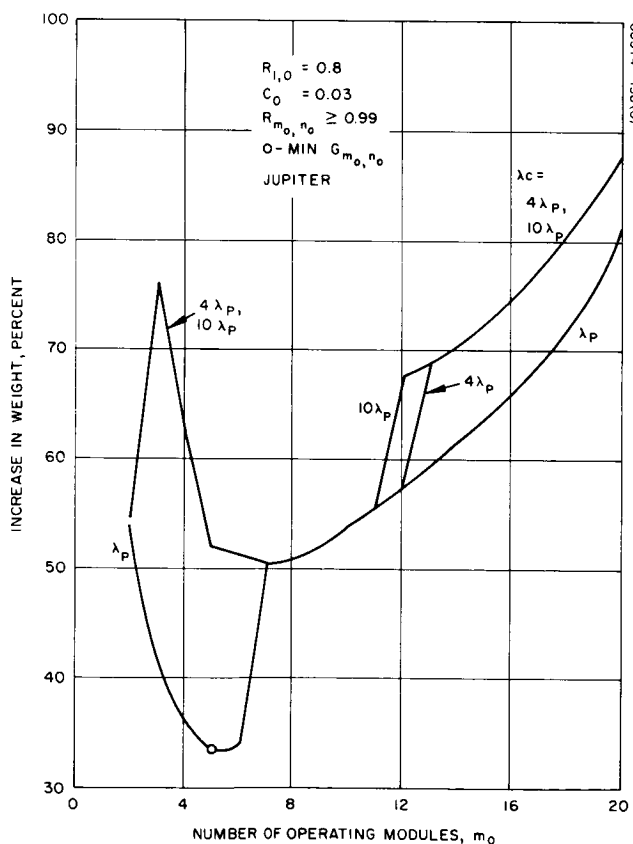


Figure F-12. Effect of λ_C/λ_P on Weight Penalty

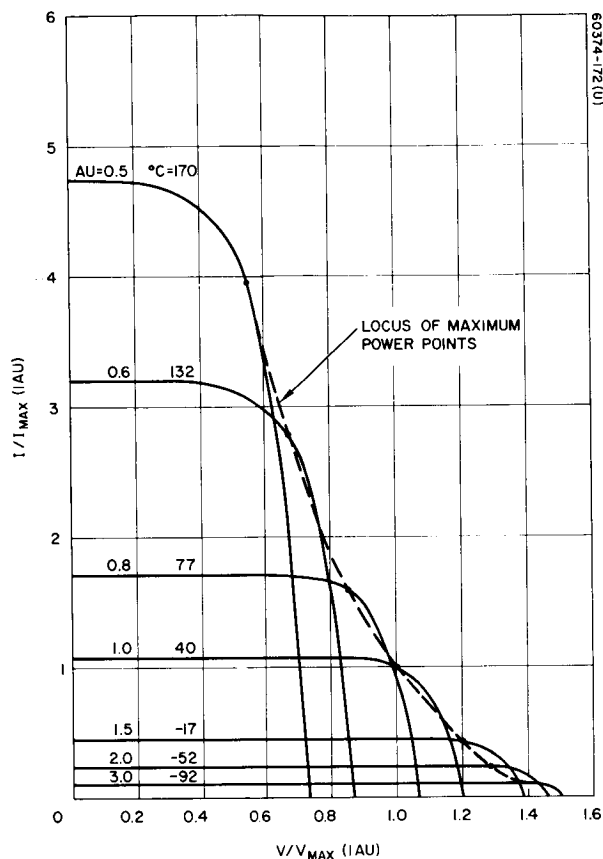


Figure G-1. Normalized Family of V-I Characteristics for Solar Array in Thermal Equilibrium at Distance of 0.5 to 3.0 AU from Sun

is implicitly assumed that the V-I characteristic of a solar cell can be described by the exponential curve of Equation 1 over the entire temperature and light intensity ranges encountered between 0.4 and 4.0 AU from the sun. It is known from experimental measurements by Heliotek, however, that this is not possible. At the high temperatures and high light intensities encountered at distances less than 0.9 AU variations in the parameters I_s , I_0 , and λ are no longer consistent enough to enable Equation 1 to fit the actual V-I curves. It turns out that Equation 1 will predict the correct qualitative behavior of the cell but not with quantitative accuracy at distances less than 0.9 AU from the sun.

The variations in I_s , I_0 , and λ have been found by studying References 1, 2, and 4 and from information supplied by Heliotek (Reference 4).

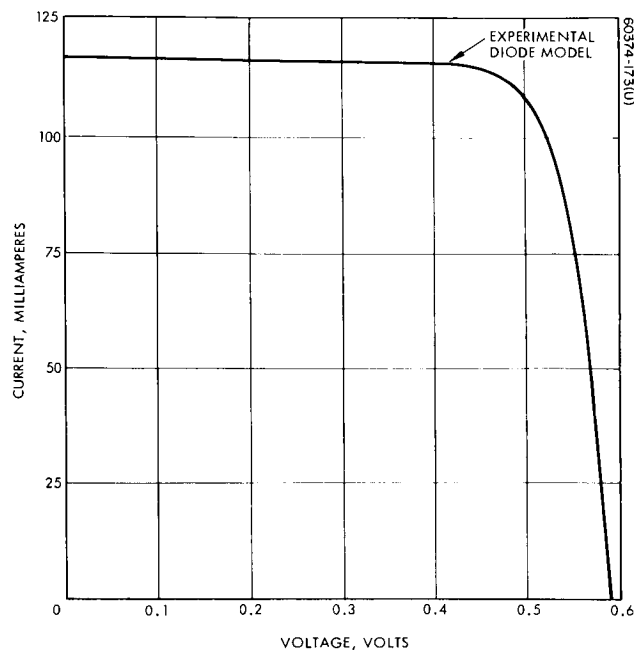


Figure G-2. Single Cell V-I Curve

An excellent fit is obtained as seen in Figure G-2. Since such good agreement is obtained for the V-I curve at a temperature of 25°C and input power density of 140 mw/cm², it is expected that when the temperature and flux dependence of the parameters I_s , I_0 and λ are known, an accurate family of characteristics can be determined. Here, it

The parameter λ is independent of incident flux. Its dependence in temperature is given by

$$\lambda = \frac{q}{AkT} \quad (2)$$

where q is the electronic charge, k is the Boltzmann constant, $q/k = 1.16 \times 10^4$ o/volt, A is a number between 1 and 2, and T is the absolute temperature. It was found that $A = 1.3$ yields the best fit.

The short circuit current parameter I_s varies linearly with incident light flux and therefore is proportional to $1/R^2$, R being the distance from the sun in astronomical units. The temperature dependence has been given by Heliotek at $92 \mu A/^{\circ}C$ for $0 < T < 110^{\circ}C$ and $136 \mu A/^{\circ}C$ for $-62 < T < 0^{\circ}C$. The equation for I_s as a function of T and R is therefore

$$I_s(T, R) = (0.1147 + 0.000092 T) \frac{1}{R^2} \quad T > 0^{\circ}C \quad (3)$$

$$I_s(T, R) = (0.1147 + 0.000136 T) \frac{1}{R^2} \quad T < 0^{\circ}C$$

The parameter I_o is known to be independent of flux. Its temperature dependence may be found indirectly from data on the variation of the open circuit voltage V_{oc} with temperature for a fixed flux. This data is given at 140 mw/cm^2 as variations of $2.2 \text{ mv}/^{\circ}C$ for $0 < T < 110^{\circ}C$ and $1.3 \text{ mv}/^{\circ}C$ for $-62 < T < 0^{\circ}C$. From Equation 1 it follows that

$$I_o = I_s / \left(e^{\lambda V_{oc}} - 1 \right) \quad (4)$$

With

$$V_{oc}(T, 1 \text{ AU}) = 0.59 + 0.0022(25-T) \text{ for } T > 0^{\circ}C$$

and

$$V_{oc}(T, 1 \text{ AU}) = 0.59 + 0.0022(25) - 0.0013 T \text{ for } T < 0^{\circ}C \quad (5)$$

the function $I_o(T)$ is then

$$I_o(T) = I_s(T, 1 \text{ AU}) / \left(e^{\lambda V_{oc}(T, 1 \text{ AU})} - 1 \right) \quad (6)$$

Using the above equations for $\lambda(T)$, $I_s(T, R)$, and $I_o(T)$ in Equation 1 V-I characteristics for a single solar cell may be drawn for specified distances and temperatures. Instead of scaling these characteristics to represent an array of cells, it is more convenient to normalize them and work with $I(T, R)/I_{\max}(T_1, 1 \text{ AU})$ and $V(T, R)/V_{\max}(T_1, 1 \text{ AU})$. Here $I_{\max}(T_1, 1 \text{ AU})$ and $V_{\max}(T_1, 1 \text{ AU})$ are the current and voltage at the maximum power transfer point on the characteristic curve for which $R = 1 \text{ AU}$, $T = T_1$ = the array temperature at 1 AU. The calculation of T_1 and in fact the entire $T(R)$ curve are considered next.

THERMAL BALANCE

A thermal balance for the array operating at a distance R (astronomical unit) from the sun yields the equation

$$T = \left[(1 - \eta(T, R)) \frac{B}{R^2} \right]^{1/4} \quad (7)$$

which relates the equilibrium temperature T to the distance R . The function $\eta(T, R)$ is the efficiency of electrical conversion for the cells and the constant B depends on the solar flux at 1 AU, the emissivity and absorptivity of the array, and the Stefan-Boltzmann constant. It would appear that $\eta(T, R)$ can be calculated from the diode model equation and then substituted in Equation 1 to find $T(R)$. This direct approach is not possible because there is no such thing as a constant temperature, constant flux V-I characteristic for a solar-cell in thermal equilibrium in free space. This is so because the temperature of the cell will vary as the operating point changes and different amounts of incident power are converted to electrical power. However, it is not necessary to know the exact expression for $\eta(T, R)$ in order to obtain a good approximation to $T(R)$. With 16 percent as the maximum value of η , the true $T(R)$ curve must lie between a curve determined from Equation 1 with $\eta = 0$ and one based on $\eta = 16$ percent. Calling these $T_o(R)$ and $T_{16}(R)$ it is seen that

$$T_{16}(R) = (1 - 0.16)^{1/4} T_o(R) = 0.963 T_o(R)$$

Less than a 4 percent error in the equilibrium temperature will be made if the assumption $\eta = \text{constant}$ is made in Equation 7. Using this assumption then, it follows that $T \sim R^{1/2}$ and finally $T(R)$ is taken to be

$$T(R) = \frac{313}{R} \text{ } ^\circ\text{K}$$

so that $T(1 \text{ AU}) = 40^\circ\text{C}$, which is consistent with Boeing Company solar panel design data.

SOLAR ARRAY PERFORMANCE

Upon substitution of the $T(R)$ curve into the diode model equation the analysis of the family of single cell V-I characteristics is complete. When normalized as described earlier these curves also hold for any size array. In Figure G-1 V-I curves are shown for distances of 0.5 to 3 AU. The corresponding equilibrium temperatures and maximum power points are indicated. In Figure G-3 the normalized voltage, current, and output resistance of the array at maximum power are given for distances of 0.5 to 4 AU.

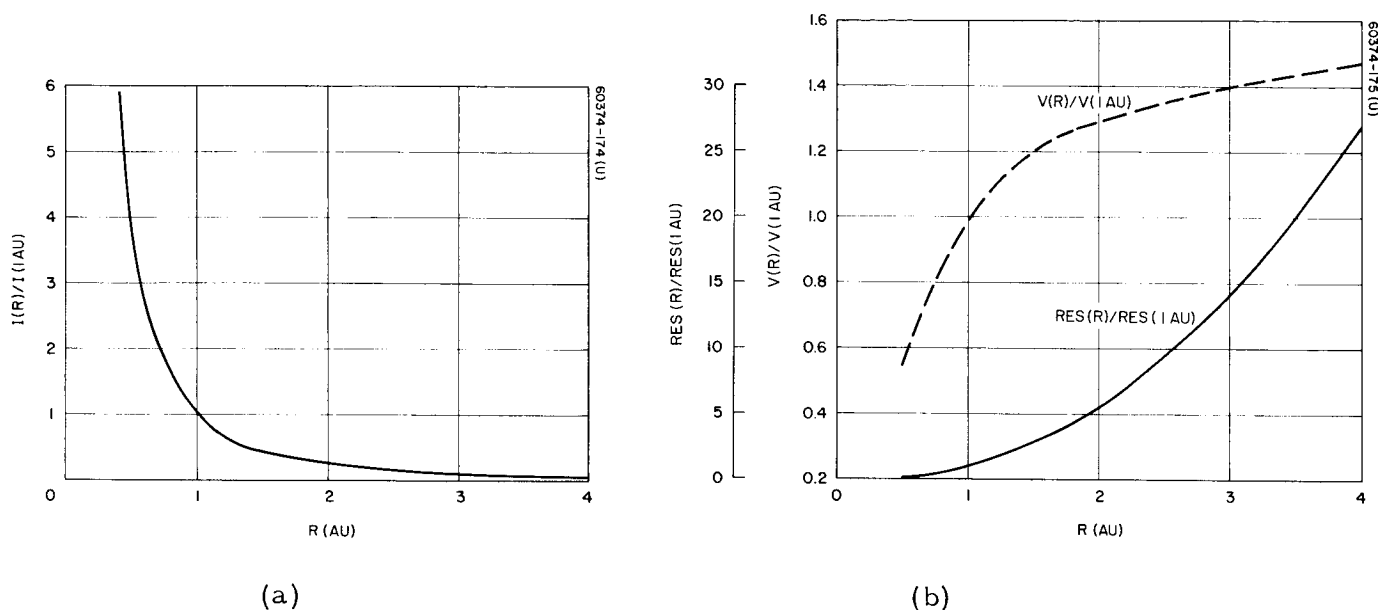


Figure G-3. Normalized Voltage Current and Output Resistance at Maximum Power

Three curves of maximum available power versus R are shown in Figure G-4 for equilibrium temperatures of 25°, 40°, and 60°C at 1 AU. Notice that all three curves are close together for distances of 0.6 to 4 AU. Since radiation other than light becomes important if the array spends time inside 1 AU, the behavior of these curves in the regions where they differ is questionable.

The last curve shown in Figure G-5 is of $\eta(T, R)$ on T and $R(T)$. Notice that the maximum η is 12.7 percent.

The validity of the constant η approximation is now easily checked. The highest temperature of interest is 406°K (at 0.6 AU) and the largest error in $T(R)$ is therefore 15 degrees. But, as shown in Figure G-4 errors of 15 degrees are unimportant in the calculation of the maximum available power over the range of 0.6 to 4 AU.

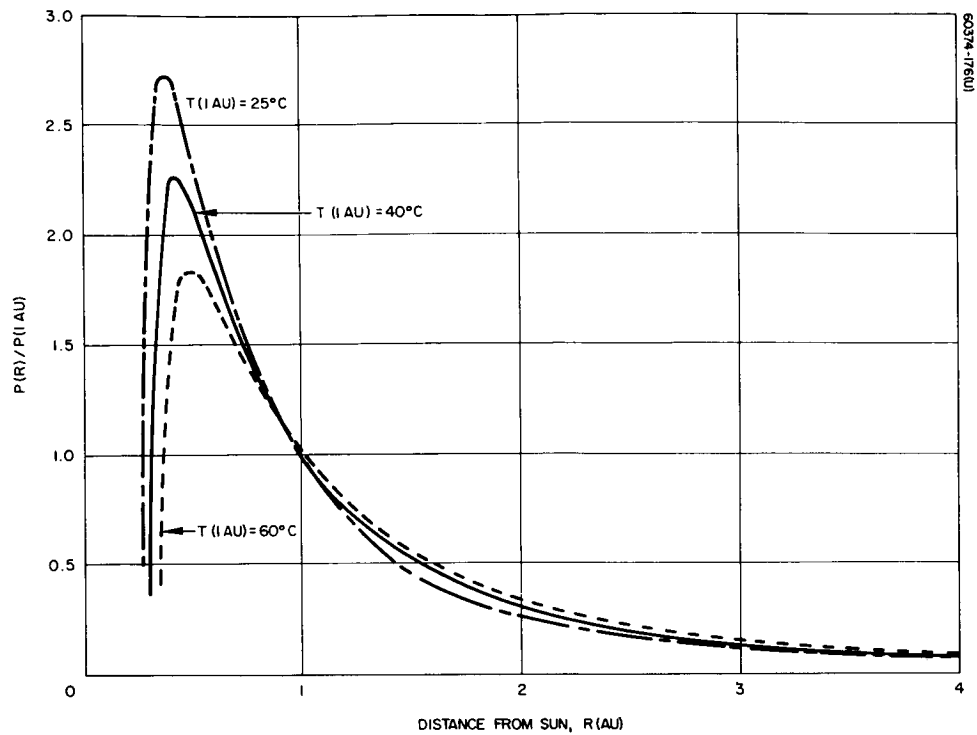


Figure G-4. Normalized Maximum Available Power versus Distance from Sun for Equilibrium Temperatures of 25°C , 40°C , and 60°C at 1 AU

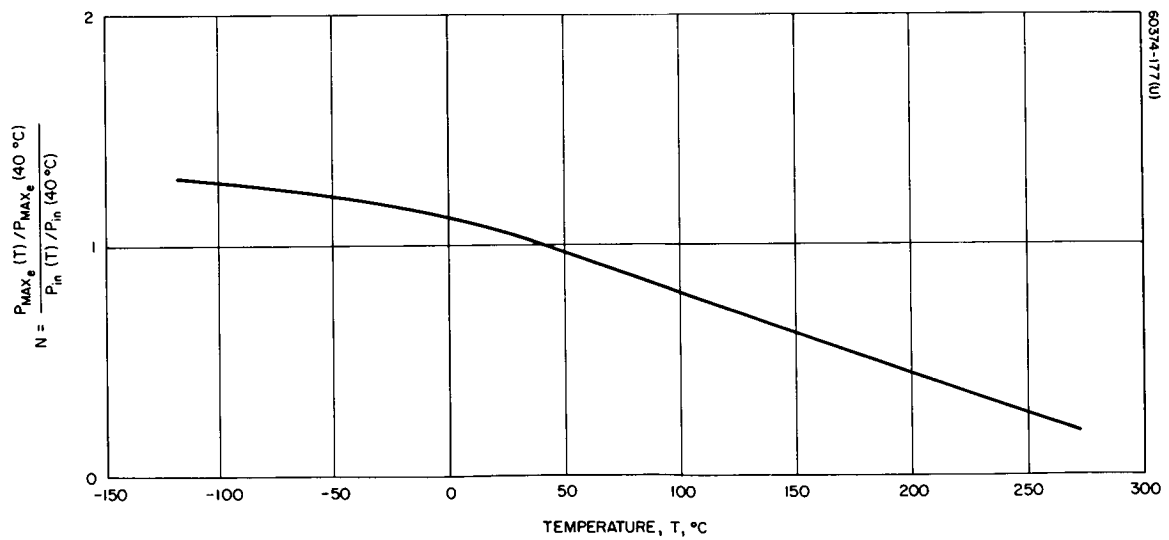


Figure G-5. Efficiency versus Temperature

The variation of the power as a function of distance is qualitatively as shown in Figure G-4. There is a power peak that is very likely located inside 1 AU and is greater than the power at 1 AU. Appendix D contains a discussion of a special effort which was made to determine a power versus distance curve which is accurate to distances of 0.4 AU.

A comparison of that power versus distance curve (Figure D-5) and those in Figure G-4 shows that there is good agreement for distances larger than 1 AU and that the power peak is smaller and lies closer to 1 AU than indicated by the curves in Figure G-4.

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1. M. B. Prince, "Silicon Solar Energy Converters," J. App. Phys. Volume 26, Number 5, May 1955.
2. J. J. Loferski, "Recent Research on Photo Voltaic Solar Energy Converters," IEEE, 1963.
3. J. J. Wysocki and Paul Rappaport, "Effects of Temperatures on Photovoltaic Solar Energy Converters," J. App. Phys., Volume 31, Number 3, March 1966.
4. G. Ralph, Heliotek Corporation, private communication.

APPENDIX H. EFFECT OF SOLAR PANEL VOLTAGE ON SYSTEM SPECIFIC WEIGHT

A study was conducted to determine the effect of solar panel output voltage on the combined weight of the solar panel and power conditioning system. The analysis was made on the assumption that the solar array specific weight is 50 lb/kw and is not a function of output voltage.

The major factors that can lead to reductions in system specific weight for the higher solar panel voltages are the decrease in transmission line weight, the increase in power conditioning system efficiency, and possible modifications of the power conditioning system design. Each of these factors are evaluated quantitatively in this appendix for two ranges of voltages: 40 to 60 volts and 60 to 90 volts.

First consider the optimization of transmission line weight. The equation for total system weight is

$$\alpha_T = \alpha_{PC} + \alpha_{SP} + \alpha_L + \alpha_{SL} + \alpha_R + \alpha_{FC} \quad (1)$$

where

- α_T = total system specific weight
- α_{PC} = specific weight of power conditioning (excluding redundancy)
- α_{SP} = specific weight of solar array
- α_L = specific weight of transmission line
- α_{SL} = specific weight of solar array to supply line losses
- α_R = specific weight of redundant circuits
- α_{FC} = specific weight of power conditioning frame, control circuits and frame wiring.

(In this case all specific weights will be referenced to the power conditioning output except where otherwise noted.)

Differentiating Equation 1 with respect to line resistance (R_L) and setting it equal to zero to find the minimum α_T gives

$$\frac{d\alpha_T}{dR_L} = \frac{d\alpha_L}{dR_L} + \frac{d\alpha_{SL}}{dR_L} = 0 \quad (2)$$

or

$$\frac{d\alpha_L}{dR_L} = - \frac{d\alpha_{SL}}{dR_L} \quad (3)$$

Since

$$\alpha_L = \frac{V_L d}{P} = \frac{\ell_L A_L d}{P} \quad (4)$$

where

V_L = volume of line

d = density of line

ℓ_L = length of line

A_L = area of line

P = power

and

$$R_L = \frac{\ell_L}{A_L} \rho \quad (5)$$

where

ρ = resistivity of line

then Equation 4 can be written

$$\alpha_L = \frac{\ell_L^2 \rho d}{R_L P} \quad (6)$$

Furthermore,

$$\alpha_{SL} = \frac{K_{sp} I_L^2 R_L}{\rho} \quad (7)$$

where

K_{sp} = specific weight of solar panel (referenced to solar panel output power)

I_L = current in line

Substituting Equations 6 and 7 into Equation 3 gives

$$\frac{d}{dR_L} \left(\frac{\ell L^2 \rho d}{R_L} \right) = - \frac{d}{dR_L} (K_{sp} I_L^2 R_L) \quad (8)$$

Differentiating and solving Equation 8 for R_L gives

$$R_L = \frac{\ell L}{I_L} \left(\frac{\rho d}{K_{sp}} \right)^{1/2} \quad (9)$$

Assuming $K_{sp} = 50$ lb/kw and a line length of 25 feet, then, for an aluminum transmission line where $\rho = 1.11 \times 10^{-6}$ ohms/in²/in and $d = 0.097$ lb/in³, the line resistance for a minimum weight system can be shown to be

$$R_L = \frac{0.882}{I_L} \text{ ohms/kw (reference to solar panel output)} \quad (10)$$

when I_L is given in amp/kw. For a 40- to 60-volt line voltage (assuming a 92.5 percent power conditioning efficiency)

$$I_L = 26.8 \text{ amp/kw}$$

whereas, for a 60-90 volt line voltage,

$$I_L = 18.0 \text{ amp/kw}$$

Thus, finding R_L from Equation 10 and substituting into Equation 6 gives

$$\alpha_L = 1.19 \text{ lb/kw (for 40 to 60 volts)}$$

and

$$\alpha_L = 0.79 \text{ lb/kw (for 60 to 90 volts)}$$

Also,

$$\alpha_{SL} = 1.18 \text{ lb/kw (for 40 to 60 volts)}$$

and

$$\alpha_{SL} = 0.79 \text{ lb/kw (for 60 to 90 volts)}$$

Hence,

$$(\alpha_L + \alpha_{SL}) \text{ 40 to 60 volts} - (\alpha_L + \alpha_{SL}) \text{ 60 to 90 volts} = \Delta\alpha \quad (11)$$

and

$$\Delta\alpha = 0.78 \text{ lb/kw}$$

It was verified during the analysis that the optimum solution for minimum weight does not result in excessive wire temperature. As indicated, the calculated specific weights were predicated on the use of aluminum wire that is superior to copper in weight-to-conductivity ratio. The use of copper would exaggerate the already substantial difference in system weight that exists for the two different line voltages.

The effect of line voltage on power conditioning efficiency and design will now be discussed. The discussion will use as an example a beam supply module and then the results assumed applicable to the total system. A layout of the power-conditioning panel for the oxide cathode system is shown in Figure H-1, illustrating the size of the array and weights of the individual modules. The array is thermally self-sufficient, maintaining a plate temperature of 60°C by radiation only, with transistor junction temperatures of 70°C or less.

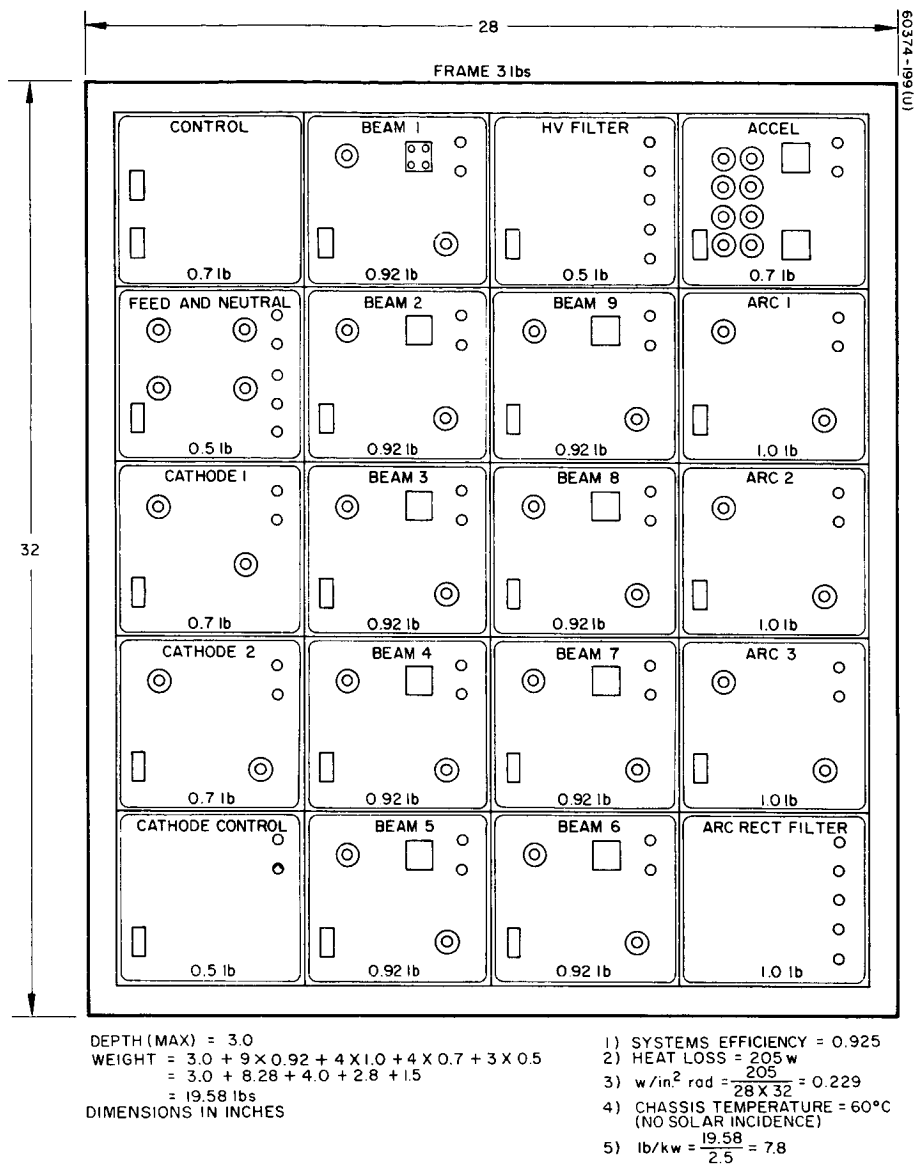


Figure H-1. 2.8 kw Ion Engine Power Conditioning Assembly

The power transistor chosen for the system inverters is the Solitron MHT 7805 (formerly Minneapolis-Honeywell). Characteristics of this transistor are: V_{ce0} 325, V_{cb0} 350, I_c 10A cont., 15 ampere surge, $h_{fc} = 20$ to 60 at 5 amperes, $V_{ce \text{ sat}}$ 0.5 at 5 amperes, $F_T = 50 \text{ mc}$, $T_j = 200^\circ\text{C}$, $\theta = 2^\circ\text{C/watt}$, $V_{BE \text{ SAT}} = 1.2$.

CASE I: 250-WATT BEAM SUPPLY MODULE, 40- TO 60-VOLT LINE

The current in a module is given by

$$I_L = \frac{250 \text{ watts}}{\eta_r \times \eta_t \times V_p} = 6.66 \text{ amperes}$$

where

η_r = rectifier efficiency = 0.99

η_t = transformer efficiency = 0.98

V_p = transformer primary voltage = 38 volts (assuming a 1-volt wiring and induction drop)

Consider now the losses in a module as follows:

- 1) Saturated transistor switch loss, P_{SAT} - From the transistor specifications, the transistor drop is 0.5 volt per 5 amperes, therefore

$$P_{SAT} = 6.66 \text{ amperes} \left(\frac{0.5 \text{ volt}}{5 \text{ amperes}} \times 6.66 \text{ amperes} \right) = 4.44 \text{ watts}$$

- 2) Transistor switching transient loss, P_{SW} - The peak power switched is $I_L \times V_M$ where V_M is the mean line voltage (i. e., 50 volts). Therefore, assuming a triangular power pulse, the average power in the pulse during switching is $(I_L \times V_M)/2$. Furthermore, since the ratio of the switching period to the half cycle time is 0.1:25, and since there are two transistors

$$P_{SW} = \frac{6.66 \text{ amperes} \times 50 \text{ volts}}{2} \times \frac{0.1}{25} \times 2 = 1.33 \text{ watts}$$

(Note: The exact solution for inductive load switching yields 2.2 watts (2Txs), for resistive load 0.6 watts (2Txs). The solution given is, therefore, an approximation for a typical switching envelope.)

- 3) Base power loss, P_{BASE} - where P_{BASE} is equal to the base-to emitter drop (1.5 volts) times I_{BASE} - Since the circuit $\beta = 15$,

$$P_{BASE} = 1.5 \text{ volts} \times \frac{6.66 \text{ amperes}}{15} = 0.67 \text{ watt}$$

The total transistor power loss is then

$$P_{Txs} = P_{SAT} + P_{SW} + P_{BASE} = 6.44 \text{ watts (2Tsx)}$$

- 4) Series resistor loss in base circuit, P_{BD} — By design this loss is three times the loss in the base itself. Therefore,

$$P_{BD} = 3 \times 0.67 \text{ watts} = 2.01 \text{ watts}$$

The total power loss in a module is then

$$P_r \text{ (rectifier)} = 2.50 \text{ watts}$$

$$P_t \text{ (transformer)} = 5.00$$

$$P_{Txs} \text{ (transistor)} = 6.44$$

$$P_{BD} \text{ (base drive)} = \underline{2.01}$$

$$P_T \text{ (total loss)} = 15.95 \text{ watts}$$

The module efficiency, η_m , is

$$\eta_m = \frac{250 \text{ watts}}{266 \text{ watts}} = 93.8 \text{ percent}$$

Since the weight of a module is 0.93 pound, its specific weight, α_m , is

$$\alpha_m = \frac{0.93 \text{ lb}}{0.250 \text{ kw}} = 3.72 \text{ lb/kw}$$

The area of radiation plate, A_m , required to maintain a 60°C average module temperature (assuming $1/3 \text{ watt/in}^2$ dissipation) is

$$A_m = 16 \text{ watts} \times 3 \text{ in}^2 / \text{watt} = 48 \text{ in}^2$$

CASE II: 250-WATT BEAM SUPPLY MODULE, 60- TO 90-VOLT LINE

An equivalent analysis assuming a 60- to 90-volt line results in the following:

$$I_{\text{Line}} = \frac{250 \text{ watts}}{0.99 \times 0.98 \times 58 \text{ volts}} = 4.5 \text{ amperes}$$

$$1) \quad P_{\text{SAT}} = 4.5 \text{ amperes} \left(\frac{0.5 \text{ volts}}{5 \text{ amperes}} \times 4.5 \text{ amperes} \right) = 2.03 \text{ watts}$$

$$2) \quad P_{\text{SW}} = \frac{4.5 \text{ amperes} \times 75 \text{ volts}}{2} \times \frac{0.1}{25} \times 2 = 1.35 \text{ watts}$$

$$3) \quad P_{\text{BASE}} = 1.5 \text{ volts} \times \frac{4.5 \text{ amperes}}{15} = 0.45 \text{ watt}$$

The total transistor power loss is then

$$P_{\text{TXS}} = 2.03 + 1.35 + 0.45 \text{ watts} = 3.83 \text{ watts (2TXs)}$$

$$4) \quad P_{\text{BD}} = 3 \times 0.45 \text{ watts} = 1.35 \text{ watts}$$

The total power loss in a module is then

$$P_r \text{ (rectifier)} = 2.50 \text{ watts}$$

$$P_t \text{ (transformer)} = 5.00$$

$$P_{\text{TXS}} \text{ (transistor)} = 3.83$$

$$P_{\text{BD}} \text{ (base drive)} = \underline{1.35}$$

$$P_T \text{ (total loss)} = 12.68 \text{ watts}$$

$$\eta_m = \frac{250 \text{ watts}}{263 \text{ watts}} = 95.1 \text{ percent}$$

$$\alpha_m = \frac{0.92 \text{ lb}}{0.250 \text{ watt}} = 3.68 \text{ lb/kw}$$

$$A_m = 13 \text{ watts} \times 3 \text{ in}^2/\text{watt} = 39 \text{ in}^2$$

TABLE H-1. SYSTEM SPECIFIC WEIGHTS

Item	40 to 60 Volts	60 to 90 Volts	Difference
Solar array to supply PC output	50.0	50.0	--
Solar array to supply PC losses	3.9 (92.2 percent average efficiency)	3.2 (93.5 percent average efficiency)	0.7
Solar array to supply line losses	1.2	0.8	0.4
Line excluding insulation	1.2	0.8	0.4
Line insulation	0.3	0.3	--
Operating modules (average)	4.6	4.4	0.2
Redundant modules (average)	2.0	1.9	0.1
Control circuits	0.2	0.2	--
Frame and wiring	<u>1.3</u>	<u>1.1</u>	<u>0.2</u>
Total	64.7 lb/kw	62.7 lb/kw	2.0 lb/kw

Table H-1 summarizes the results obtained from the above analysis. Approximately one-third the total difference in weight for the two line voltages considered was due to the effect on solar array weight caused by a difference in efficiency of the inverters; another third was comprised of equal weight changes chargeable to actual transmission line weight and weight of solar array needed to supply line losses. The remaining difference was accounted for by small changes in the weight of the power conditioning modules and frame and wiring.

Note that the total weight difference in Table H-1 is equal to the weight of the total module redundancy in the system. Another way to assess the importance of this weight saving is to consider that the redundancy in the 60- to 90-volt system could be doubled without exceeding the total system weight of the 40- to 60-volt system, thereby tremendously increasing reliability without increasing total weight. This procedure is not recommended since the present redundancy is considered adequate, but this comparison serves to illustrate the importance of the line voltage.

APPENDIX I. ATTITUDE CONTROL FUEL CALCULATIONS FOR MARS MISSIONS

PARAMETER VALUES, GROUND RULES, AND ASSUMPTIONS

Δt = travel time = 423 days (maximum) = 36.6×10^6 seconds

I_{sp} { = specific impulse of cold gas (nitrogen) = 70 seconds
= specific impulse of hot gas (nitrogen) = 123 seconds

P = solar radiation pressure at 1 AU = 9.37×10^{-8} ft-lb/ft³

A = area of solar array = 1120 ft² = 1.04×10^6 cm²

γ = reflection coefficient $\cong 0.15$

l_D = radius arm of disturbance torques, assumed to be
2 percent of diameter of circle of equivalent area

Orbit time = 180 days

TRANSIT FUEL CONSUMPTION

l = l_z, l_y, l_x = gas jet moment arms = $8 + 7/2 = 11.5$ feet

$$1120 \text{ ft}^2 = \pi \frac{d^2}{4}$$

d = 37.8 feet

so,

$$l_D = (37.8)(0.02) = 0.76 \text{ foot}$$

$$\begin{aligned}
T_1 &= \text{disturbance torque from solar radiation pressure (at 1 AU)} \\
&= (A)(\ell_D)(1 + \gamma)(P) \\
&= (1120 \text{ ft}^2)(0.76 \text{ ft})(1.15)(9.37 \times 10^{-8} \text{ ft-lb/ft}^3) \\
&= 0.92 \times 10^{-4} \text{ ft-lb}
\end{aligned}$$

$$\begin{aligned}
T_2 &= \text{maximum solar wind torque} = 1.23 \times 10^2 \text{ dyne-cm (at 1 AU)} \\
&= 0.09 \times 10^{-4} \text{ ft-lb}
\end{aligned}$$

$$\begin{aligned}
T_3 &= \text{solar flare torque} = 6.7 \times 10^2 \text{ dyne-cm} \\
&= 0.49 \times 10^{-4} \text{ ft-lb (at 1 AU)}
\end{aligned}$$

Regarding T_2 and T_3 as random and independent they will be represented as bounded (with 99 percent confidence) at

$$10^{-4} \sqrt{(0.49)^2 + (0.09)^2} = 0.50 \times 10^{-3} \text{ ft-lb}$$

So, the disturbance torque is bounded, with 99 percent confidence, as $0.92 + 0.50 = 1.42 \times 10^{-6} \text{ ft-lb (at 1 AU)}$.

The distance from the sun varies (approximately linearly with time) from 1.0 to 1.56 AU (see Figure I-1).

$$T_d (\text{maximum}) = 1.42 \times 10^{-4} \text{ ft-lb}$$

$$T_d (\text{minimum}) = 0.92 \times 10^{-6} \left(\frac{1}{1.56} \right)^2 = 0.38 \times 10^{-4} \text{ ft-lb}$$

$$\text{Average disturbance torque} = T_d(AV) = \frac{10^{-4}(1.42 + 0.38)}{2}$$

$$= 0.90 \times 10^{-4} \text{ ft-lb}$$

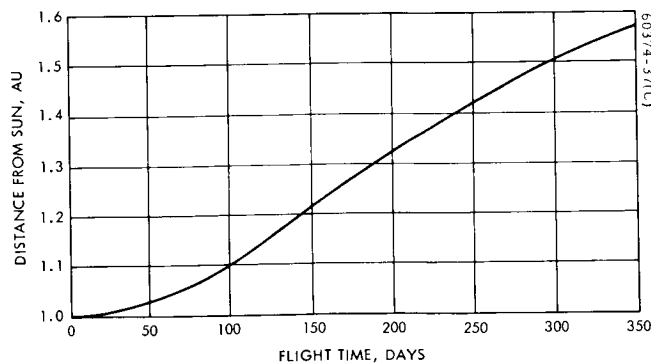


Figure I-1. Distance from Sun versus Time for Earth-Mars Rendezvous Reference Trajectory

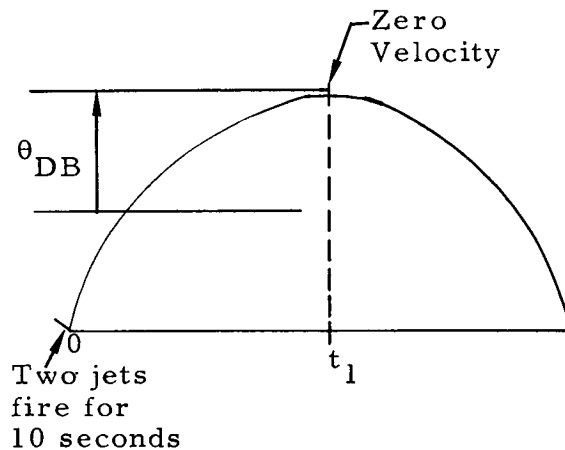
$$W_1 = \text{transit fuel consumption} = \frac{\Delta t T_d (AV)}{I_{sp}}$$

$$= \frac{(36.6 \times 10^6)(0.90 \times 10^{-4})}{(1.23 \times 10^2) \frac{11.5 \text{ ft}}{1.414}} = 3.29 \text{ lb}$$

NOTE: This is for one axis control about vehicle thrust vector.

The thrust level is set so that the minimum disturbance torque will result in an ideal soft limit cycle

I = inertia
T = thrust level
R = radius arm
 θ_{DB} = deadspace
= 1 degree
 t_{ON} = jet on-time
= 100 msec



For zero velocity at t_1

$$2t_{ON} \frac{TR}{I} = \frac{(T_{d_{min}})(t_1)}{I}$$

where

$$t_1 = \frac{2\theta_{DB}}{(\text{average velocity})} = \frac{2\theta_{DB}}{t_{in} \left(\frac{TR}{I} \right)}$$

Substituting for t_1 and solving for T

$$T = \frac{(\theta_{DB} T_{d_{min}} I)^{1/2}}{t_{ON} R} = \frac{\left[\left(\frac{1}{57.3} \right) \text{ rad} (0.38 \times 10^{-4} \text{ ft-lb}) (4500 \text{ slug-ft}^2) \right]^{1/2}}{(10^{-1} \text{ sec})(11.5 \text{ feet})}$$

$$= 0.048 \text{ pound}$$

This thrust level is virtually the same as for the Vela thruster so the unit weight will be assumed the same as for Vela.

ORBIT FUEL CONSUMPTION

Three axis cold gas jet control will be on during this time so $I_{sp} = 70$ seconds.

For a 180 day orbit, 0.4-pound gas usage is predicted for an I_{sp} (pulsed) of 35 seconds. The fuel usage in orbit is a function of $(inertia)^{+1}$, $(I_{sp})^{-1}$ and $(reaction\ jet\ moment\ arm)^{+1}$. Therefore, the Mars orbiter fuel consumption is taken to be the Mariner figure times the ratio of I_{sp} , moment arms, and inertias.

$W_2 =$ orbit fuel consumption

$$= (0.4\ lb) \frac{\left[\frac{900}{340} + \frac{900}{340} + \frac{700}{510} \right]}{3} \left(\frac{35}{70} \right) (1) = 0.44\ pound$$

RATE DISSIPATION FUEL CONSUMPTION

It is assumed that no overshoot occurs during rate dissipation so that

$$\omega_o = \text{initial rate} = \frac{(T)(R)}{I} (\text{time})$$

and

$$\frac{\omega_o}{I_{sp}} = \frac{R}{I} \left[\frac{(T)(\text{time})}{I_{sp}} \right] = \frac{R}{I} W_3$$

where

$W_3 =$ fuel consumption during rate dissipation.

Thus

$$W_3 = \frac{I_{\omega_o}}{R I_{sp}} = \frac{(5 \times 10^3\ \text{slug-ft}^2) \left(\frac{3}{57.3\ \text{sec}} \right)}{(11.5\ \text{feet})(70\ \text{seconds})}$$

$$W_3 = 0.325\ pound$$

ACQUISITION FUEL CONSUMPTION

Using position plus rate feedback, the system can be made to attain and dissipate a velocity (about a given axis) with little overshoot — say 10 percent. The same transient (polarity reversed) would be exhibited when the maneuver is terminated.

So, acquiring the sun by two sequential maneuvers and allowing for a third maneuver for star acquisition means that the fuel requirements for acquisition are

$$W_4 = 2(1 + 0.1) \sum_{i=1}^3 \frac{I_i \omega_i}{R_i I_{sp}}$$

$$\omega_i = \text{velocity attained} = \text{rotation rate} = 0.5 \text{ deg/sec}$$

$$W_4 = 2.2 \left\{ \frac{0.5/57.3 \text{ rad/sec}}{(11.5 \text{ feet})(70 \text{ seconds})} [4500 + 4500 + 5000] \text{ slug-ft}^2 \right\}$$
$$= 0.335 \text{ pound}$$

Two separate acquisitions are allowed for should the spacecraft be hit by a meteor and lose optical reference. So

$$W_4' = 2(0.335) = 0.67 \text{ pound}$$

PANEL EJECTION FUEL CONSUMPTION

It is assumed that the ejection of the outer solar panels will cause no worse a transient than initial rate dissipation. So

$$W_5 = \text{fuel required for panel ejection maneuver}$$
$$= W_3 = 0.325 \text{ pound}$$

TOTAL FUEL CONSUMPTION

$$W = W_1 + W_2 + W_3 + W_4' + W_5$$

$$= 3.29$$

$$0.44$$

$$0.325$$

$$0.67$$

$$0.325$$

$$\hline 5.05$$

Allowing for a 300 percent safety factor

$$W = 3(5.05) = 15.15 \text{ pounds} \cong 15.2 \text{ pounds}$$