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MARS SPINNING SUPPORT MODULE STUDY

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SYSTEMS LABORATORIES

CAT-31

VOLUME I
SUMMARY

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1. INTRODUCTION

Hughes Aircraft Company's Space Systems Division, under contract* to the National Aeronautics and Space Administration's Langley Research Center (NASA/LRC), has completed a study of various Mars Spinning Support Module (MSSM) designs to be used by a landing vehicle in the Mars 1973 Program. The results of this study, which was conducted during a 2-month time period, are presented in three separate volumes that collectively constitute the final report.

This document, Volume I, presents a summary of the significant technical features of the various designs studied, the guidelines and ground rules directing the study, and a tentative program plan for design, development, and delivery of support modules for the Mars 1973 mission.

Volume II presents the detailed analysis and data that supports the design performance and parameter values selected. A large portion of the information was available from past and current Hughes experience with similar spinning earth communications satellites, and was supplemented with new analysis and studies dictated by unique support module features.

Volume III presents program plan and cost estimates for the support module designs considered during the study. This data is based on Hughes' design, development, testing, and delivery of similar systems from Syncom through Surveyor to the present complex Advanced Technology and Intelsat spacecraft. The backlog of actual cost data attached to the development and fabrication of satellite components was obtained from the appropriate Hughes divisions for use in estimating support module costs.

The NASA is considering a Mars 1973 mission as one of a group of missions directed toward exploration of the planet Mars using automated spacecraft, with the primary objective of obtaining scientific data from the Mars atmosphere and surface. Present plans envision two landing vehicles being placed on earth-Mars trajectories by separate Titan III-C boost vehicles in July/August 1973, and landing on the Martian surface in January/February 1974. The limited, but flexible, NASA budgetary considerations result in a spectrum of mission and landing vehicle design possibilities that satisfy Mars 1973 mission objectives. Hughes has been directed by NASA/LRC to study a direct entry mode mission utilizing an aeroshell plus a hard landing vehicle with an attached supporting module.

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*Contract No. NASI-8638, August 30, 1968.

The function of the support module (SM) is to provide trajectory control via earth command, relay telemetry and commands to/from the earth, provide power, and initiate functions for the landing vehicle while enroute to Mars. The SM will separate from the aeroshell/lander (A/L) near Mars encounter, and be deflected to fly by Mars relaying entry and video data from the lander to the earth, as the lander accomplishes the descent and post-landing operations. The passive spin-stabilized, rather than three-axis, attitude control technique for the support module – aeroshell/lander (spacecraft) was also specified as a key requirement for the designs.

In addition to the direct entry mode of lander delivery, directives were given to study a flyby mode mission wherein the A/L vehicle is deflected from the SM near Mars encounter and placed on an entry and landing path. The SM will perform its relay function and immediately initiate a braking maneuver with a propulsion system capable of substantial ΔV capability, placing it into an elliptical orbit about Mars rather than flying by. The orbit will be selected so the SM will pass over the lander once each Martian day, relaying data to earth for the lifetime of the lander.

The study resulted in three different SM designs that satisfy the specified mission requirements and Titan III-C payload limitations; a flyby SM which relays entry and video data (Flyby-Video Data), a similar SM which relays only entry data as it flys by Mars (Flyby-Entry Data), and an orbiting design which has the full video data capability in addition to its orbiting and long lifetime relay capability (Orbiter). Several alternate designs were also considered, but eliminated early in the study. The study concluded with a comparison of the Flyby and Orbiter design weights and program costs.

The requirement that a Mars exploratory vehicle shall have an extremely low probability of contaminating Mars necessitates incorporation of a sterilization program into the Mars 1973 Program. Because of this, the A/L must be sterilized before launch, regardless of the particular mission mode selected. The SM need not be sterilized if the flyby mode mission is selected (Orbiter design), but must be sterilized if the direct entry mode is utilized (Flyby design) because of the possibility of impacting Mars should the deflection maneuver fail to be executed properly.

The remainder of this volume, presents the following information: 1) Section 2 presents a brief comparative summary of the three SM designs and the primary study conclusions, 2) Section 3, is a review of the general and detailed guidelines and ground rules specified or agreed to by NASA/LRC which are the framework of the study, 3) Section 4 contains a mission profile depicting major events and SM implications on the profile, 4) Section 5 is a description of the Flyby-Video Data configuration and summary of the main subsystem design and performance, 5) Section 6 is a similar presentation of the Flyby-Entry Data design and performance, with emphasis on video versus entry design differences, 6) Section 7 contains a description of the Orbiter design, pointing out its key features, and 7) Section 8 presents a program plan for a Mars 1973 mission support module which highlights major design, fabrication, testing, and delivery milestones for the flight and associated test models and operational support equipment.

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2. SUMMARY/CONCLUSIONS

SUMMARY

The gross characteristics of the Flyby-Video Data and Orbiter MSSM designs which received the major study effort are illustrated in Figures 1 and 2. The Flyby-Entry Data design, which also received considerable attention, is similar to the Video Data design, the primary difference being the incorporation of a much smaller square planar antenna. Table 1 is a list of subsystem parameter values and characteristics for the three MSSM designs. Since most of the SM design problems were associated with the structural, flight control, and telecommunications subsystems, the major study efforts were devoted to these subsystems.

The Flyby-Video and Entry Data SM designs weigh 558 and 521 pounds, respectively, including the propellant. The Orbiter design weighs 1120 pounds, including the reaction control and orbital insertion system propellant. These weights added to the A/L, adapter, and canister weights result in total weights which are within the Titan III-C payload capabilities. The net weight margins are 105, 138, and 90 pounds, respectively, interpreted as available for SM, adapter, and canister contingencies.

The solar panel bounding circle diameters are 113 and 118 inches, respectively, for the Flyby and Orbiter designs as required to accommodate 37.4 square feet of solar panel area and the planar antenna. The diameter is limited by a necessity for adequate clearance between the panels and canister (or Titan III-C shroud for the Orbiter design). The Orbiter design central cylinder height, 38 inches, is greater than that of the Flyby design (22 inches), because it contains a solid propellant retro engine. The structural layout and lightweight aluminum material selection is based on Hughes experience with similar spinning earth communications satellites.

The attitude and reaction jet control system utilizes a standard blowdown hydrazine monopropellant design with ten 5-pound thrusters operating in a pulsed mode to achieve velocity, attitude, and spin control. Four sun sensors and a star sensor also are used to obtain attitude data. The Flyby SM-A/L designs are passively spin stabilized, although the Orbiter design requires a single active damper to prevent drifting of the spin axis.

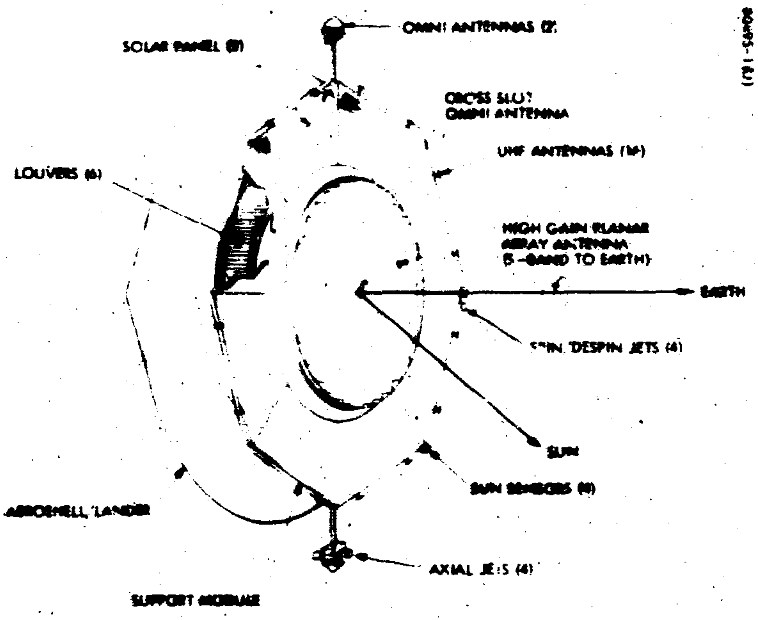

Figure 1. Mars Flyby Design

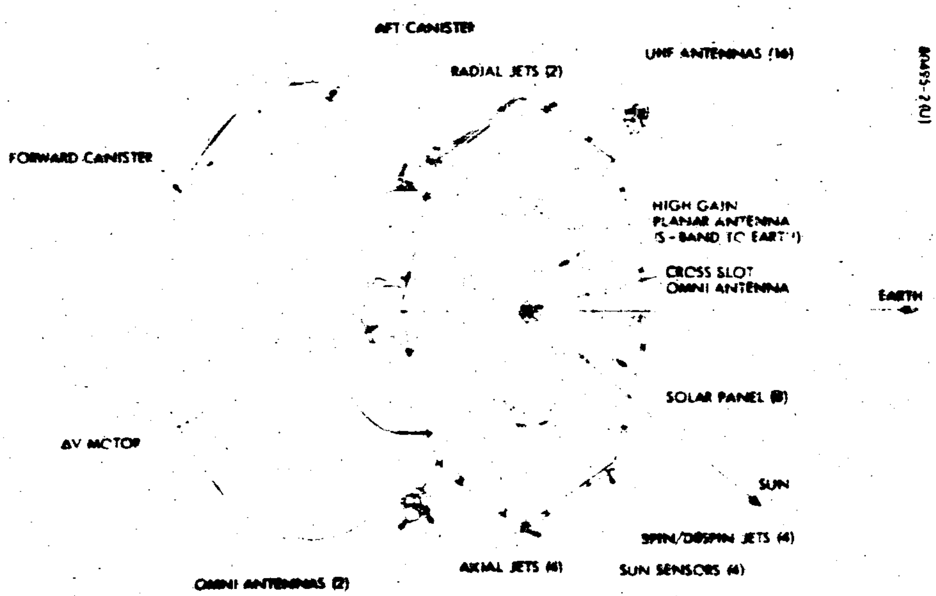

Figure 2. Mars Orbiter Design

The major elements of the communications subsystem are the S-band 18 and 21 square feet high-gain planar antennae, and 40 watt-TWT transmitters for the Flyby-Video Data and Orbiter SM designs, respectively. These, along with the A/L to SM UHF receiving capability, complete the link for transmission of Mars scientific data to earth. In addition, a 3-watt transmitter, cross slot antennae, and backup antenna with complete redundancy perform the remaining telecommunications function. The Flyby-Entry Data design has a 3.7-square-foot planar antenna and 20-watt transmitters for the primary data link.

The remaining key design aspects are the power and thermal subsystems. The solar panels provide 85 watts of power at Mars encounter and two batteries, each with 283 watt-hours capacity and singly able to support the mission, sustain the high-power periods. The thermal design utilizes a superinsulation blanket and active louvers to maintain compartment temperatures between 40 to 100°F.

CONCLUSIONS

- The three MSSM designs synthesized and presented, all of which satisfy specific flyby or orbiter mission requirements and the allotted payload capability of the Titan III-C, are recommended as candidates for the Mars 1973 Program.

- The utilization of spin stabilization, simple structural and subsystem mechanization, backup modes of operation, extensive redundancy, wide operating margins and the resultant weight margins in the MSSM designs, coupled with Hughes' extensive background with similar vehicles, places a high degree of confidence in the study results.

- The potential development and integration problems have been minimized by maintaining a simple mechanical interface and a single power, command, and telemetry line between the A/L and SM designs

- The A/L design was simplified by the elimination of its spin control system due to the SM capability to perform the spindown function prior to A/L-SM separation.

- The questions associated with sterilization of the Flyby SM designs, although not critical, are eliminated from the nonsterilized Orbiter SM design considerations. These refer specifically to star sensor photomultiplier tube and battery procurement and confidence in estimating program plans and costs.

- The Orbiter design offers the following significant advantages over the Flyby designs:

  1) Greater Mars scientific data return because of its periodic communication with the A/L while in Mars orbit.

2) The potential to return a significant quantity of data while in orbit, even if the initial entry and video data were lost.

3) The possibility of incorporating scientific instrumentation for gathering data while in orbit. This feature would also allow some mission goals to be achieved even if the A/L failed.

- The estimated cost of a Flyby SM program is $1.7 M more than the cost of a similar Orbiter SM program. The Orbiter cost is reduced because sterilization is not required.

TABLE 1. MSSM DESIGN SUMMARY

	Flyby		Orbiter (solid propellant)
	Video	Entry	
Weights, pounds			
Support module	558	521	1120
Aeroshell/lander	1100	1100	1186
Adapter canisters	368	368	344
Total	2026	1989	2650
Margin (SM, adapter, canisters)	105	138	90
Dimensions			
Solar panel area, square feet	37.4	37.4	37.4
Solar panel outer diameter, inches	113	113	117.6
SM cylinder, diameter/height, inches	67.5/21.5 in	67.5/21.5 in	72.8/38 in
A/L diameter, inches	121	121	121
Canister, diameter/height, inches	125/83.5	125/83.5	125/61.5
Flight Control/Propulsion			
Attitude/velocity jets	6	6	6
Spin jets	4	4	4
Sun/star sensors, accuracy	4/1, <1	4/1, <1	4/1, <1
ΔV accuracy	5%	5%	5%
θ accuracy	2.6	2.6	2.4
Hydrazine RCS			
Tank diameter, inches	14.2	13.9	14.2
Jet thrust, pounds	3 to 5	3 to 5	3 to 5
Propellant weight, pounds	120	110	115
Retro engine weight, pounds			406
Retro ΔV capability			1.7 km/s
Nutation damper	Passive mercury tube	Passive mercury tube	Active accel./motor
Communications			
Planar antenna gain, dB	29.2	22.0	30.1
High-gain antenna beamwidth	5.2°	11.4°	4.6°
High-gain antenna area, square feet	17.9	3.7	21.2
S-band transmit power, watts	3/40	3/20	3/40
S-band frequency, up/down, MHz	2113/2295	2113/2295	2113/2295
Link ranges, km			
S-band	160×10^6	160×10^6	182×10^6
UHF entry	≤5530	≤5530	≥5530
UHF video	≤5000		≥5000
Bit rate capability			
Telemetry, bits per second	17000	2000	16000
Commands, bits per second	≥300	≥300	≥300
Cross-slot antenna gain, 120°, dB	-1	-1	-1
Turnstile antenna gain, dB	≥-6	≥-6	≥-6
UHF receive antennas	16, 1/2 λ dipoles	16, 1/2 λ dipoles	16, 1/2 λ dipoles
UHF receive antenna gain, dB	≥-4.2	≥-4.2	≥-4.2
A/L transmit power, watts	50	35	50
A/L antenna gain, 70° dB	0	0	0
UHF frequency, MHz	400	400	400
Power			
Solar panel power, watts	≥85	≥85	≥85
Total battery capacity, watt-hours	566	566	632
Battery type	Ag-Zn	Ag-Zn	Ag-Zn
Sterilization Required	Yes	Yes	No

3. STUDY GUIDELINES

The initial ground rules specified by NASA/LRC were recognized early in the MSSM study to be subject to change and various interpretations because of the flexibility inherent in the Mars 1973 Program planning* and possible SM implications. These circumstances led to a loose interpretation of many of the ground rules, although they were bounded by sound engineering practices. A guideline, rather than hard ground rule, approach was the framework used to direct the study. The resultant guidelines used for the Flyby-Video Data and Orbiter SM design studies agreed upon during study are presented in the following paragraphs. The Flyby-Entry Data study utilized the same directives as the Video Data study, with some differences as noted under the Video Data study guidelines.

FLYBY-VIDEO DATA

- The Mars 1973 mission shall have two spacecraft, launched by separate Titan III-C boosters, placed on earth-Mars trajectories, with the landings on Mars separated by 10 days. The launch and arrival dates are:

	Spacecraft 1	Spacecraft 2
Launch	July 12 to August 9, 1973	July 12 to August 11, 1973
Arrival	January 12, 1974	January 22, 1974

- The Direct Entry Mode shall be used to deliver the spacecraft to Mars.
- The maximum payload capability of the Titan III-C for this mission period is 2080 pounds, which includes the adapter between the Titan III-C last stage and the capsule.
- The inboard profile of A/L and canisters is shown in Figure 3. Although the actual dimensions were changed throughout the study, the general layout was used. The A/L shall weigh 1100 pounds. The aeroshell, interface structure, and canister parameters shall be determined during the study. The thruster (deflection rocket) shall be removed, interface structure modified, and canisters changed to accommodate the SM.

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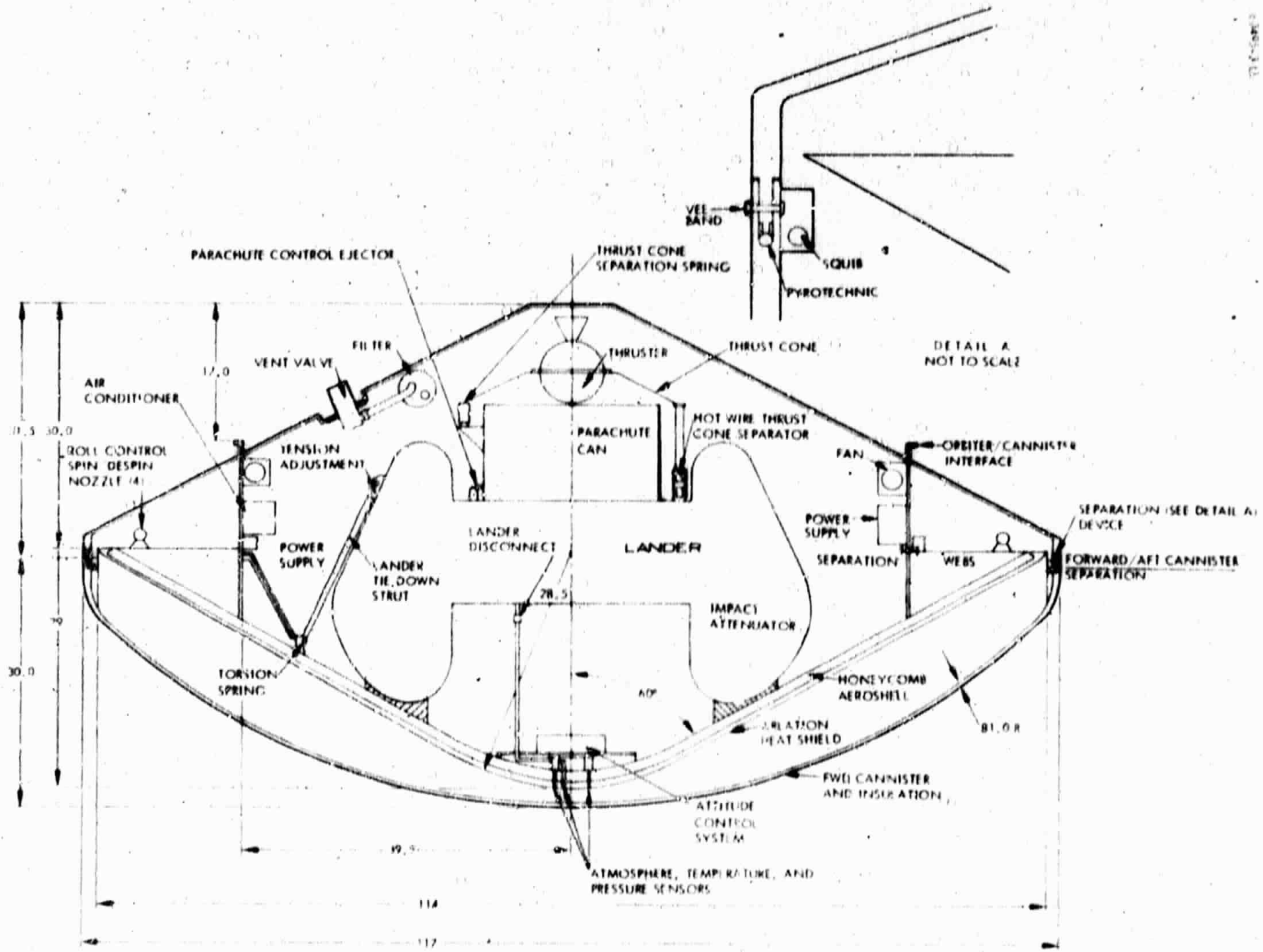
*Titan Mars '73 Mission Definition (For Planning Purposes Only), No. M73-101-2, Aug. 12, 1968; Mars 1973 Reference Mission Design (Direct Entry, For Planning Purposes Only), No. M73-103-1, June 26, 1968, NASA/LRC.



WEIGHT, lb

FORWARD CANISTER	62
AFT CANISTER	214
SUPPORT EQUIPMENT	52

Figure 3. Inboard Profile of Aeroshell/Lander

- The capsule orientation and adapter design shall be determined during the study. The Titan III-C last stage and shroud characteristics shall be a consideration in their selection.

- The SM shall be attached to the A/L and encapsulated for purposes of heat sterilization. A temperature of 135°C shall be applied to the capsule for a 24-hour period.

- The spacecraft (SM and A/L) is spin stabilized during the mission.

- The SM shall fly by Mars by performing a deflection maneuver no earlier than 1 day before Mars encounter.

- The SM shall relay A/L entry data at a rate of 2000 bits per second, and a total of 10^7 bits of video data at a nominal rate of 16,000 bits per second. The video data will be available from the lander 2 minutes after landing. The Flyby-Entry Data design shall have capability to relay only entry data at 2000 bits per second.

- The interface requirements between the SM and A/L shall be minimized and simplified.

- The SM program shall have a high reliability and minimum cost, utilizing existing and proven hardware and design techniques.

- The SM shall have a midcourse correction capability of 65 meters/second.

- The SM velocity control accuracy shall be significantly better than the orbit determination accuracy 5 to [illegible] days before Mars encounter.

- The planetary entry angle shall be between -18 and -30 degrees.

- The Jet Propulsion Laboratories Deep Space Net (DSN) shall be used for communicating with the SM during the Mars 1973 mission. Two 210-foot diameter antenna systems will be available, one each at Goldstone, California, and Woomera, Australia. The output power shall be 400 kilowatts, and the receiver noise based on 33 ±8°K.

- Telemetry data from the A/L shall be transmitted by the SM at a rate of 1050 bits per hour during the cruise phases of the mission. A rate of 2000 bits per second shall be required prior to separation of the A/L from the SM.

- Commands from the DSN shall be supplied to the A/L via the SM during the cruise phases of the mission. The nominal command word length shall be 21 bits.

- The SM shall have a transponder operation to enable the DSN to obtain range and range rate data for determination of the spacecraft trajectory.

- The SM shall incorporate a sequencer to automatically initiate A/L functions prior to A/L separation from the SM.

- The SM shall receive entry and video data from the A/L at a frequency of 400 MHz (UHF). The A/L shall supply 35 watts of power for entry data and 50 watts of power for video data transmission. The A/L shall have two cavity back slot antennae with +6 dB of gain along the A/L axis of symmetry, falling off to 0 dB of gain 70 degrees off the axis.

- The SM shall supply 25 watts of continuous power to the A/L during the mission.

- The SM shall incorporate two battery chargers for the A/L use during the mission.

ORBITER

Although many of the Orbiter guidelines are identical or similar to those of the Flyby design study, there are differences in and additional guidelines, which are as follows:

- The launch and arrival dates are:

	Spacecraft 1	Spacecraft 2
Launch	July 13 to August 10, 1973	July 20 to August 12, 1973
Arrival	January 30, 1974	February 9, 1974

- The Flyby Mode shall be used to deliver the spacecraft to Mars.

- The maximum payload capability of the Titan III-C for this lower energy mission period is 2650 pounds, which includes the adapter.

- The inboard profile shown in Figure 3 was initially intended for use with an Orbiter design. The actual dimensions used were the same as those of the Flyby SM study. The A/L shall weigh 1186 pounds, which includes the solid rocket and interface structure.

- The SM will not be sterilized; therefore, it need not be inside the canister.

- Since the initial trajectory is a Mars flyby path, the A/L shall perform a deflection maneuver no earlier than one day before Mars encounter to place it on a landing course.

- The SM shall execute a braking maneuver after performing its relay function (identical to Flyby requirements), which results in the SM attaining a 1000 x 33,000-kilometer altitude elliptical orbit about Mars. The SM shall have a 24.6-hour period orbit, which is synchronous with respect to the lander. The relaying of data during the initial flyby is a primary mission goal. The attainment of orbit is secondary.

- The SM shall relay data from the A/L during each communication period for a minimum of 3 days.

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4. MISSION PROFILE

The sequence of events for the Flyby and Orbiter design missions are almost identical from earth launch to A/L-SM separation near Mars. The series of events differ after separation because: 1) the Flyby-Video/Entry Data SM designs must perform a deflection maneuver, whereas the A/L lander performs this function in the Orbiter design concept, 2) the aft canister must be removed from the Orbiter SM, while it remains with the third stage of the Titan III-C in the case of the Flyby SM design, and 3) the Orbiter SM executes an orbital insertion maneuver, and the Flyby SM does not. The mission events which apply to the Flyby-Video Data (or Entry Data) design constitute the mission profile presented in this section. The significant Orbiter design differences will be presented in Section 7.

The Mars 1973 mission will begin at Cape Kennedy during the July/August 1973 period with the spaced launching of two Titan III-C vehicles, each carrying an A/L-SM spacecraft. A minimum of 30 days is desired for the launch period, with a 2-hour launch window for each day. The mission profile is almost identical for each spacecraft, the primary difference being a 10-day spacing in the January 1974 Mars arrival dates.

The boost phase utilizes the parking orbit technique to enable the third stage of the Titan III-C to place the spacecraft on a nominal direct entry trajectory to Mars. While in parking orbit the forward portion of the sterilization canister is removed, thereby exposing the aeroshell of the spacecraft. After the third stage and payload have attained the proper injection conditions, an attitude maneuver is performed by the third stage to position the spacecraft for the desired cruise orientation. This is followed by separation and spinup of the spacecraft, leaving the aft canister and adapter attached to the last stage, which subsequently performs a retro maneuver.

The initial spacecraft spin speed of 25 rpm is imparted by the SM spinup control system. The spin axis direction is allowed to remain gyroscopically fixed with respect to an inertial reference system for approximately the first 90 days of flight. The spin axis direction, which is initially within a few degrees of the ecliptic plane and at an angle of 45 degrees with respect to the sun's direction (Figure 4), has been selected on the basis of sustaining adequate solar panel power generation and thermal balance, spacecraft to earth telecommunications, and having the proper

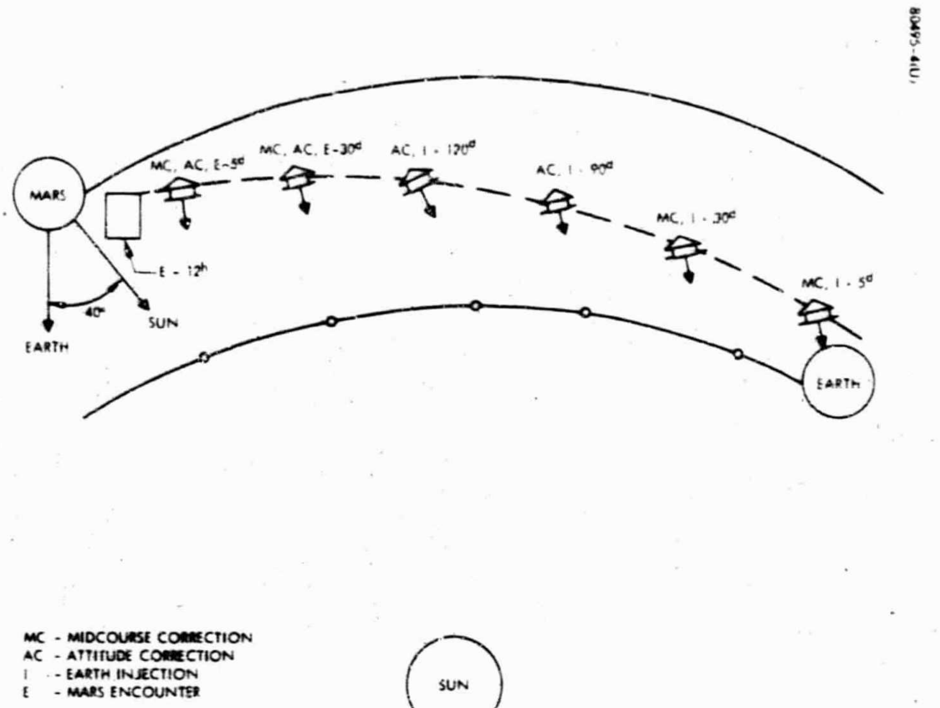


Figure 4. Mission Profile, Cruise Phase

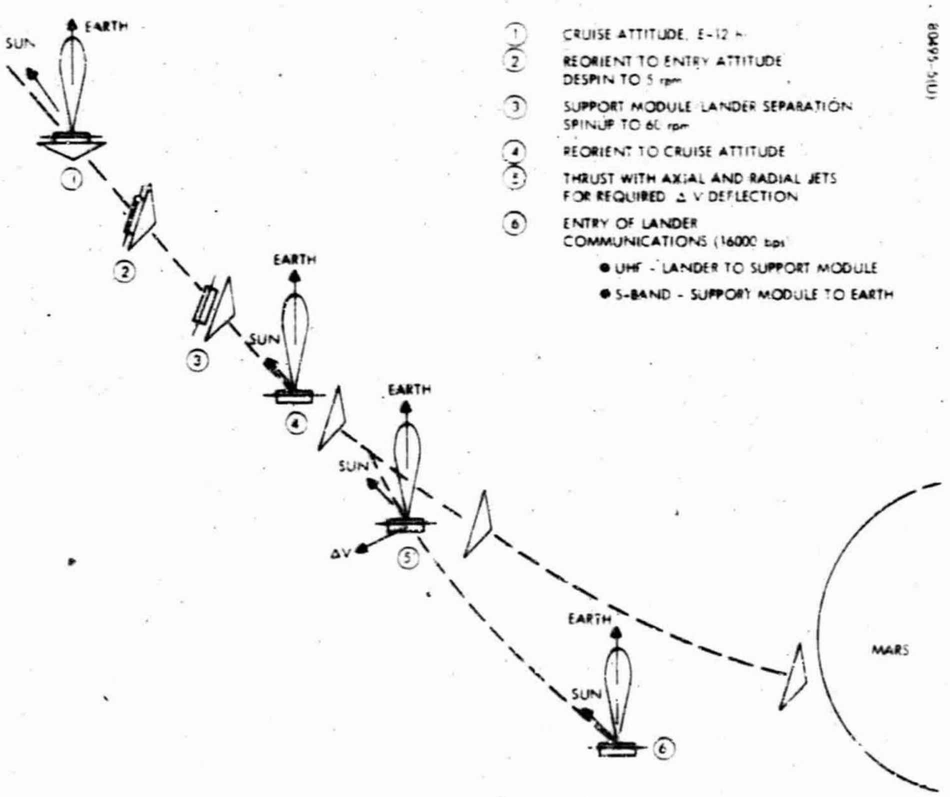


Figure 5. Mission Profile, Encounter Phase

sun/star sensor geometry at attitude determination and trajectory control times. The wide range of acceptable SM spin axis orientations permits a high degree of flexibility in designing the mission for the Mars 1973 Program.

The A/L subsystems will be commanded and monitored via the SM immediately after injection, and the SM will begin supplying 25 watts of continuous power to the A/L and acting as a tracking beacon for the DSN. Early in the cruise phase of the mission, SM sun and star sensor data will be telemetered to the earth, and the spacecraft attitude established to well within 1 degree. The SM cross slot antenna will be used for telecommunications during the first 90 days of the mission.

The mission has been designed to accommodate a maximum of four midcourse velocity corrections for a total of 65 meters/second, although only two or three are likely to be required to compensate for injection errors and achieve the high degree of final trajectory control accuracy necessary for landing. The first velocity correction is nominally planned for 5 days after injection, and the second at 30 days based on propellant, tracking accuracy, and execution error considerations. The corrections are performed without changing the spin axis orientation by using the pulsed jet reaction control techniques utilized on Hughes communication satellites.

To sustain adequate communications from approximately 90 days after earth injection to Mars encounter, the spacecraft spin axis must be changed, at most, four times to keep the earth within the beamwidth of the high-gain planar array. The attitude maneuver times selected will ensure that tracking and orbit determination accuracy are not degraded. Also, the final two midcourse corrections will be performed during this period, with the last one occurring 5 days before Mars encounter.

The A/L separation from the SM at 12 hours prior to landing (Figure 5), is preceded by a complete checkout of the A/L, loading and verifying commands for the SM automatic sequencer, initiating A/L subsystems, first spacecraft attitude rotation to align the A/L for entry, and a spin down to 5 rpm by the SM. A/L separation is then completed and the SM spun up to 60 rpm and re-oriented to direct the planar at the earth. Communications are maintained with the earth during this period via the cross slot and high-power transmitter, and the UHF link provides for A/L to SM communications. This phase is completed with a velocity maneuver of 100 meters/second being imparted by the SM to deflect onto a Mars flyby path.

The SM will utilize the high-power transmitter and planar antenna to relay A/L entry and video data back to earth (maximum range of 160×10^6 kilometers) during the final flyby phase of the mission. A nominal bit rate of 16,000 bits per second will be used to return 10^7 bits of video data to earth during the postlanding data gathering period of the Mars lander. The SM trajectory periapsis altitude is nominally 3700 kilometers. The A/L-SM maximum communications range prior to landing is 5530 kilometers, and during video transmission, 5000 kilometers. Due to trajectory and Mars atmosphere uncertainties, the post-landing time can vary from 13 to 24 minutes.

5. FLYBY DESIGN - VIDEO DATA

VEHICLE CONFIGURATION/WEIGHT

Configuration

The Flyby-Video Data SM design inboard profile is shown in Figure 6. The major structural features include a 67.5-inch diameter cylindrical section, eight trapezoidal solar panel sections arranged radially about the central cylinder, sixteen panel support struts, main equipment circular shelf in plane of solar panels, 60-inch diameter high-gain antenna support ring, and necessary support arms, insulation material, and active thermal control louvers. The subsystem equipment is arranged within the cylindrical section to achieve static and dynamic balance and is mounted to satisfy thermal, volume, assembly, and testing considerations. The materials selected (mainly aluminum) and design practices incorporated are similar to those used for Hughes communication satellites. The structure, as is the rest of the SM, is designed to withstand the high sterilization temperature.

The simple cylindrical section allows ample volume for equipment and matches the A/L and booster adapter interface dimensions. The strength of the semimonoque cylinder satisfies the requirement to support the 1100-pound A/L and its own equipment during the high acceleration and vibration boost periods of the Titan III-C. The length of the cylinder minimizes A/L interface structure and canister weight and enhances spin axis stability of the SM and total spacecraft. Two active thermal louvers which dissipate transmitter heat energy are attached about the cylinder.

The trapezoidal shape of the solar panel sections will facilitate manufacture and assembly. The outer 113-inch diameter bounding the panels is set by the aft canister dimension, UHF antennae and spin jet mounting, and margin required to achieve spacecraft separation at earth injection. The inner dimension of the panels allows for the maximum planar array diameter and clearance for interfacing the SM and adapter while maintaining 37.4 square feet of panel area to supply the required 85 watts of power at Mars encounter. A series of struts attached behind the panels and to the cylinder provide support during the boost phase. The panel cells are n-on-p type with a fused silica cover slip.

The S-band high-gain antenna is at one end of the SM above four thermal louvers that support and radiate through it, and is constrained by a 60-inch diameter ring. The antenna is constructed from rectangular cross slot cavity sections, and is energized by a waveguide running from the transmitters within the SM cylinder to its back surface. An integral part of the antenna is a special cross slot antenna located in its center and used for communications primarily during the early mission phases. The antenna height will enable a solar energy incidence angle of 40 degrees to be used without shadowing the solar panels.

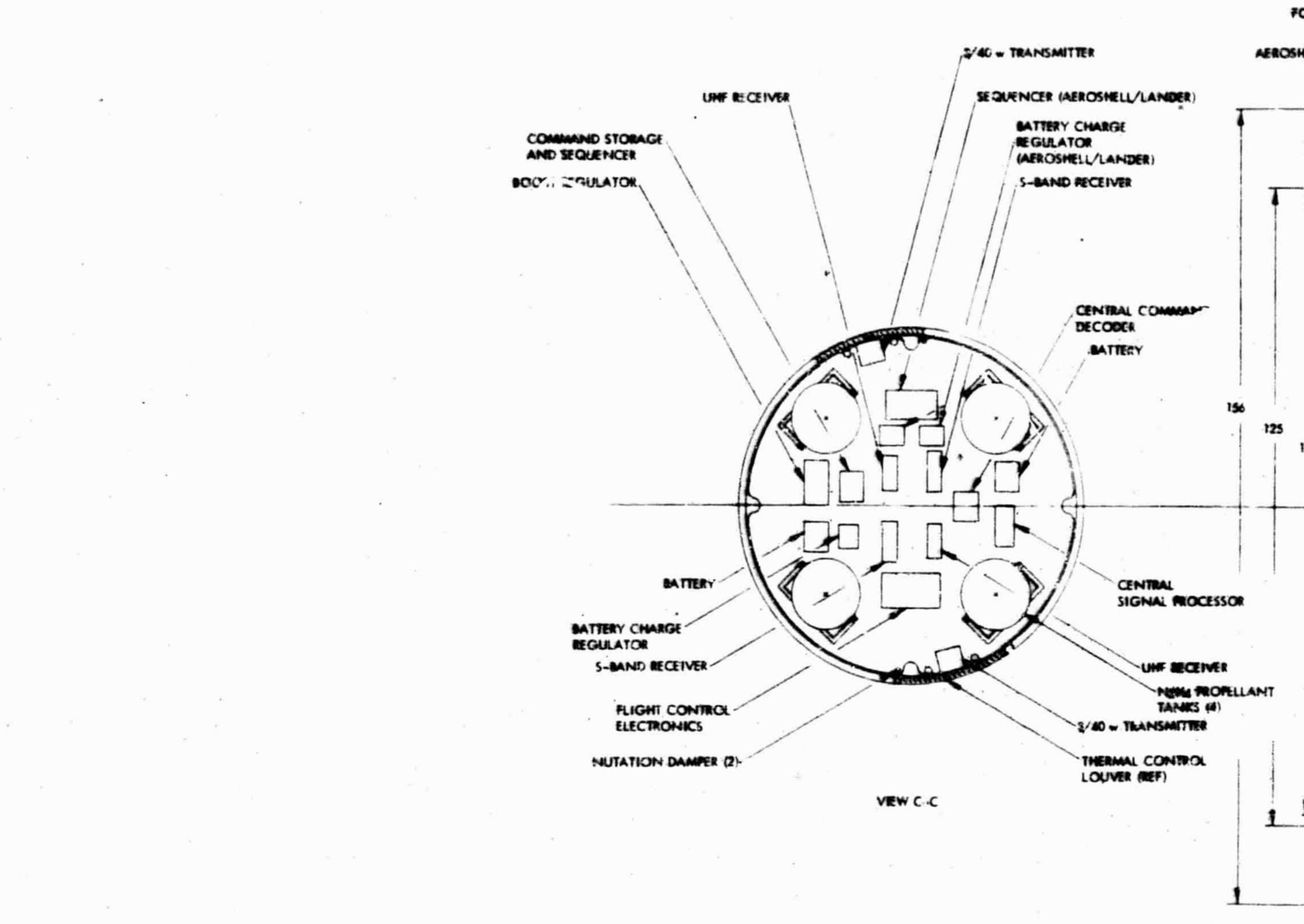
The majority of SM equipment; electronics, batteries, and propellant tanks, are attached to one side of a shelf that is fastened to the cylinder in the plane of the solar panels. The four louvers and planar antenna are attached to the opposite side of the shelf. The shelf is designed to support the equipment while undergoing heavy loads during boost. The shelf concept enhances accessibility, testing, and installation while the SM is in development. The shelf also conducts heat from the equipment to the louver system. The equipment not attached to the shelf are the transmitters with their TWTs and the attitude control system mercury tube nutation dampers, that are all mounted about the cylinder wall. A thermal blanket surrounds the equipment and closes off the other end of the SM cylindrical section. A 40 to 100° F temperature is maintained within the equipment compartment by the thermal blanket/louver control system.

The four spin jets and four sun sensors which are used in the attitude control system are mounted to the panel support struts just below the panel outer edges. The two radial control jets and a star sensor are mounted below the panels on the struts and cylinder, respectively. Their locations are chosen to eliminate jet impingement and stray light problems. In addition, one jet thrusts through the spacecraft and the other through the SM center of gravity. The remaining four axial control jets are mounted on arms that swing out from the panels. The two arms place the jets far enough outboard such that jet plume impingement on the A/L is insignificant. The use of coiled propellant lines to supply the axial jets will eliminate internal line flaking, trapping potential valve-jamming materials, and will provide a spring action for arm deployment. The system design allows for mission completion even if both arms should fail to extend, since two of the axial jets protrude past the panels while in the stowed position. Backup antennas are also mounted on the deployable arms.

The SM design includes such items as adapter and A/L interface separation mechanisms and necessary electrical lines and harnesses. Although these items are not shown on the drawings, they are accounted for in estimating SM weights.

Weight

Table 2 lists the SM, A/L, adapter, and canister weights which form the total launch weight (not including shroud). The A/L weight was fixed early in the study by Hughes-LRC agreement, and the adapter, canisters,



VIEW C-C

DIMENSIONS IN INCHES

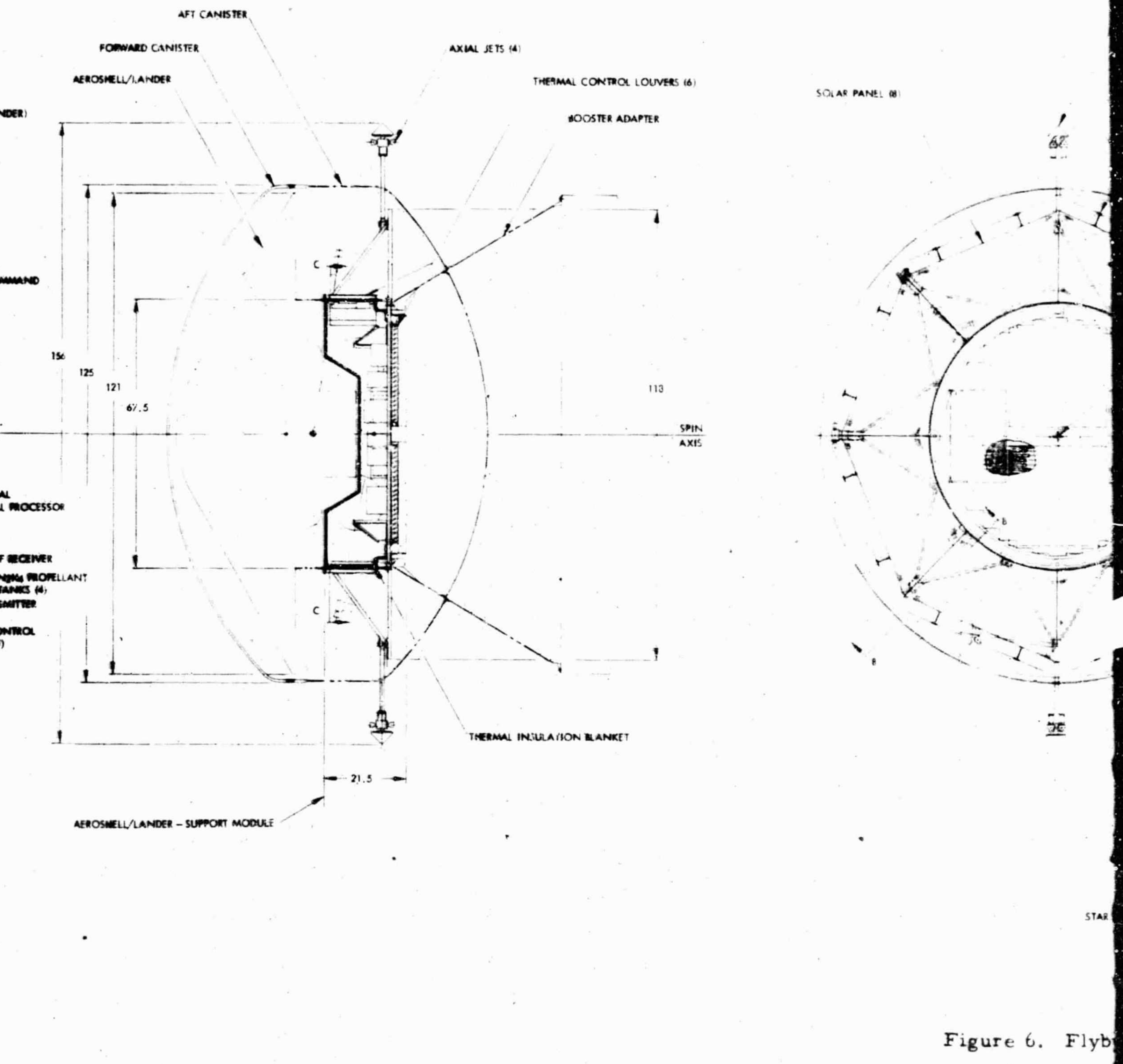


Figure 6. Flyb

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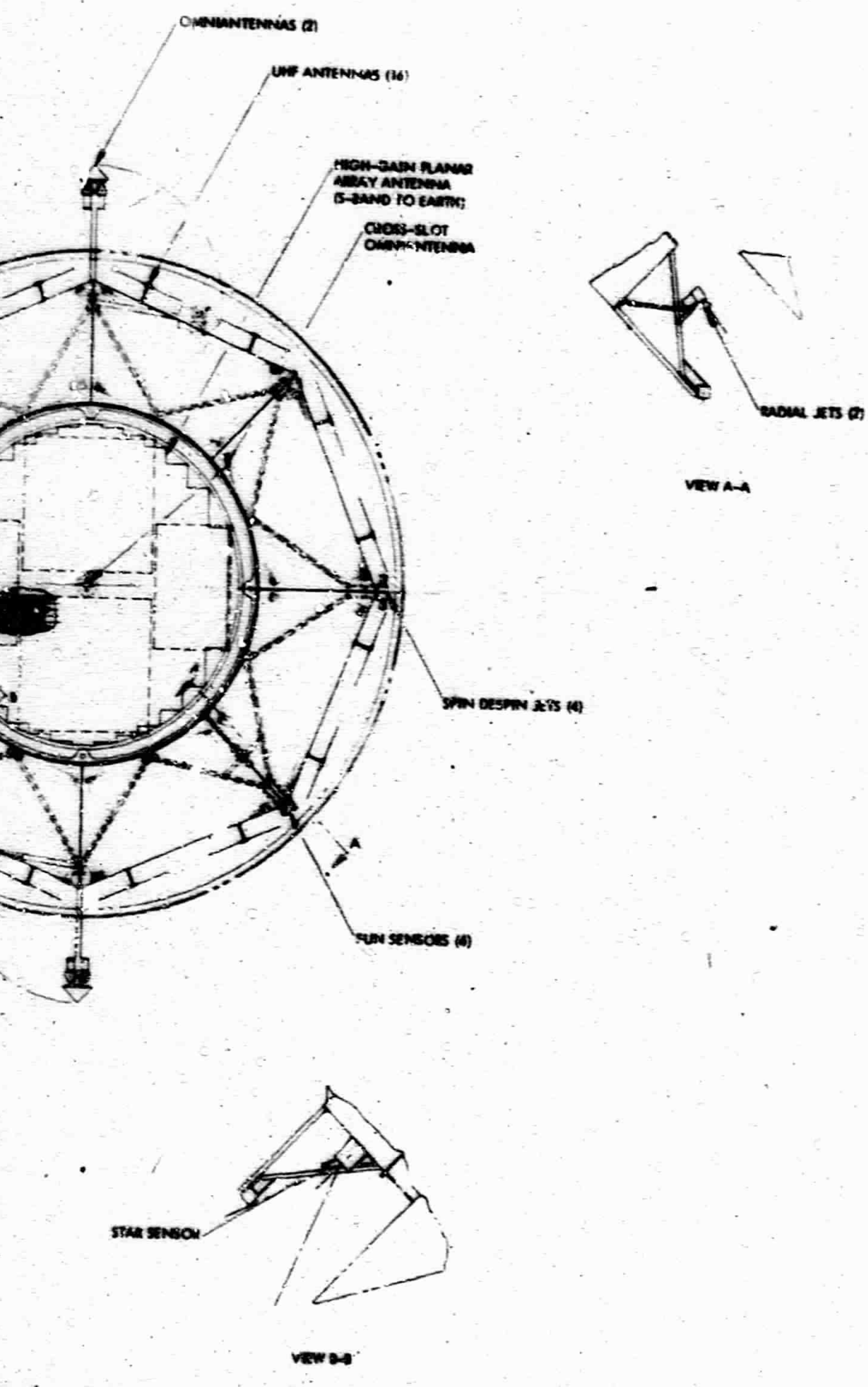

e 6. Flyby – Video Data SM Design Inboard Profile

TABLE 2. MSSM - LAUNCH WEIGHT DISTRIBUTION, FLYBY - VIDEO DESIGN

Item		Weight, pounds
Aeroshell/lander		1100.0
SM weight (Includes 51 pounds contingency)		557.9
SM/booster adapter		117.2
Structure	77.2	
Umbilical	35.0	
Pyrotechnic separation logic	5.0	
Aft canister		181.8
Structure	128.8	
Pressure and vent	50.0	
Separation harness	3.0	
Forward canister		69.0
Launch weight		2025.9
Titan payload	2080	
Total contingency for SM, adapter, and canisters		105.1

and SM weights determined as part of the study. The adapter and canister weight estimates include provisions for cables, pyrotechnics, and pressure control within the canister. The SM weight includes 51 pounds for contingencies. The difference between the Titan III-C payload capability and total launch weight is 54 pounds. If the A/L weight is assumed to contain its own margin, 105 pounds (51+54) are available for SM, adapter, and canister contingencies.

The basic subsystem weights which constitute the total SM weight of 558 pounds are listed in Table 3. A major portion of the weights is based on similar Hughes equipment weights. The propellant weight includes 11 pounds for contingency which, added to a 40-pound (10 percent)

TABLE 3. MSSM SUBSYSTEM WEIGHT ALLOTMENT, FLYBY DESIGN - VIDEO DATA

Item	Weight, pounds	
RF Subsystem		80.7
3-watt transmitter (2)	41.2	
40-watt transmitter (2)		
High-gain antenna (1)	14.0	
UHF antennas (16)	3.0	
Turnstile antenna (2)	1.0	
S-band receiver (2)	8.0	
UHF receiver (2)	4.7	
RF switches (3)	1.5	
Coaxial cables and attach clips	5.5	
Cross slot omni	0.5	
Diplexer (3)	1.3	
Signal Processing		30.7
Central command decoder (2)	5.7	
Central signal processor	10.0	
Command storage and sequencer (S/M)	5.0	
Sequencer (A/L)	10.0	
Propulsion		160.6
5-pound thrusters (10)	11.0	
14.2-inch diameter tanks (4)	21.6	
Valves, lines, filters, and pressurant	8.0	
Propellant (includes 11 pounds contingency)	120.0	
Flight Control		30.0
Flight control electronics	17.0	
Star sensors (1)	5.1	
Sun sensors (4)	1.0	
Local cables, connectors, and clips	3.4	
Nutation damper	2.5	
Accelerometer and acceleration switch	1.0	
Structure		117.2
Upper ring	11.7	
Upper cylinder	21.1	
Central ring	11.6	
Lower cover	2.0	
Equipment shelf	21.0	
Thermal blankets and tie downs	4.5	
Upper thermal blanket support	6.5	
Harness	11.0	
Solar panel support tubes	7.0	

TABLE 3. MSSM SUBSYSTEM WEIGHT ALLOTMENT, FLYBY DESIGN - VIDEO DATA - Concluded

Item		Weight, pounds	
Balance weights		4.0	
Thermal control louver (6)		15.6	
Radial jet supports (2)		1.2	
Mechanisms			14.8
Separation mechanism		11.6	
SM/lander	4.8		
SM/adapter	6.8		
Axial jet deployment (2)		3.2	
Electrical Power			84.1
Solar panel (substrate, cells, interconnections)		36.8	
Boost regulator		8.5	
Battery charge regulator (1 S/M 2 A/L)		10.5	
Battery (2)		28.3	
Total		518.1	
Contingency		39.8	
Total support module weight		557.9	

contingency on SM dry weight, yields the 51 pounds shown in Table 2. An A/L sequencer and two battery chargers are included in the SM weight. The SM structural weights are based on reasonable design strengths, including an additional 10 percent weight increase for sterilization. The SM structural weight is approximately 16 percent of the dry weight, which compares to the 10 to 15 percent values for Hughes spacecraft.

Table 4 presents the moment of inertia estimates for several loading conditions. The spin-axis (roll)-to-pitch moment of inertia ratio is well above 1.0, thereby ensuring stability about the SM spin axis during the entire mission.

ATTITUDE AND VELOCITY CONTROL/PROPULSION

The spin, attitude, and velocity control of the spacecraft is implemented by the SM utilizing components, subsystems, and techniques which have undergone extensive flight experience in spin-stabilized missions, including Syncom II and III, Intelsat I and II, and Applications Technology Satellites (ATS) 1 and 3. Additional flight experience is in the offing on ATS-E, Air Force Tactical Communications Satellite, and Intelsat IV, which are currently in various phases of flight vehicle development. The basic elements of the system are a spinning spacecraft with adequate gyroscopic

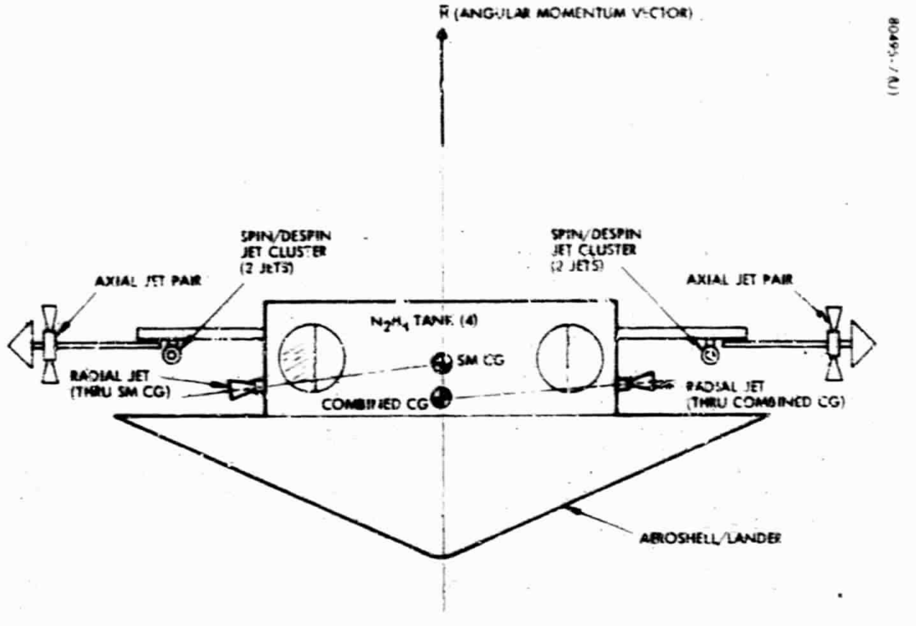

Figure 7. General Arrangement of Reaction Control System

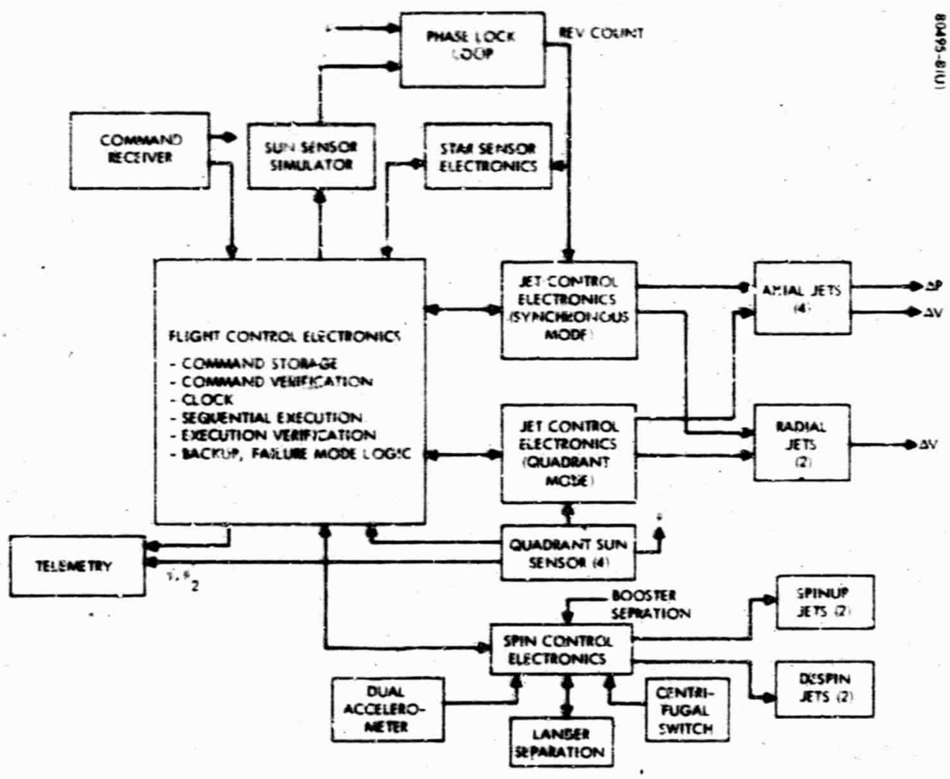

Figure 8. MSSM Flight Control Electronics

TABLE 4. MOMENTS OF INERTIA, FLYBY - VIDEO DESIGN

Vehicle State	Slug-ft^2
A/L and SM	
Full fuel	
I_{Roll}	390.9
I_{Pitch}	295.2
40 percent fuel	
I_R	382.2
I_P	289.2
SM	
40 percent fuel	
I_R	72
I_P	37.6
Zero percent fuel	
I_R	64.8
I_P	34.0

stability; spinup/spindown, axial and radial jets; sun and star sensors to determine spacecraft attitude and control the jets; jet control electronics; nutation dampers, and a pressurized monopropellant hydrazine system that supplies the reaction control jets. Figure 7 illustrates the layout of the reaction control subsystem, and Figure 8 is a block diagram of the flight control electronics.

The four spin jets are mounted in pairs at the edge of the solar panels and are used initially to spin up the spacecraft to 25 rpm for the mission cruise phase. The spin jets are required to perform only two more times: to spin down the spacecraft to 5 rpm prior to A/L separation, and then spin up the SM to 60 rpm for the flyby phase. An accelerometer (centrifugal) terminates the spinup operations and a centrifugal switch the spin down maneuver. The spinup speeds are based on a consideration of stability and jet control performance. A wide variation in these speeds is acceptable, the values given are only representative. The 5 rpm imparted to the A/L maintains its stability prior to entry, thereby eliminating the need for an attitude control system. The spin tolerances are not critical; therefore, a simple inexpensive spin control system suffices.

Attitude and spin axis direction velocity control is achieved through the use of four axial-directed jets located outboard the spacecraft on extendible arms (see Figure 7). The arms provide a large lever arm for the small, 5-pound thrust jets, and allow for jets thrusting in both axial directions without producing a significant jet exhaust gas impingement on the A/L. Attitude changes in the spin axis direction are produced by opposite jets, acting as a couple, which are pulsed in a desired average inertial direction, causing a transverse torque that precesses the angular momentum vector. The couple eliminates unwanted translational velocity components. Although only two jets are needed (one if couple requirement is waived), an additional set is provided for redundancy. The SM, operating open-loop, can maintain its attitude to within the 2.6 degrees (3σ) required for accurate pointing of the planar antenna after A/L separation.

The continuous firing of an axial jet(s) produces a velocity change along the spin axis without causing a change in spacecraft attitude. The use of an additional radial jet, which is "pulsed" along a direction through the vehicle center of gravity, with the axial system allows velocity corrections to be executed in any direction without requiring a spacecraft attitude change. Two radial jets (see Figure 7) are attached to the SM, one through the total spacecraft center of gravity and the other through that of the SM for use after A/L separation. The ability to perform velocity corrections without changing the vehicle attitude allows earth communications to be maintained, especially near Mars encounter, on the high gain antenna. A velocity correction accuracy of 4 to 5 percent of the total correction can be readily obtained with this system, based on previous experience with preflight calibration data and inflight temperature and pressure measurements of propellant systems. This accuracy satisfies the requirement of not significantly degrading trajectory control accuracy, as limited by orbit determination while performing the encounter minus 5-day velocity correction.

Orbit determination accuracies of from 60 to 70 kilometers (3σ), in Mars miss coordinates, will not be significantly degraded.

Control of the jets is effected by an electronics system which has been flight proven many times and is known, since its inception at Hughes in 1959, as the jet control electronics (JCE). The primary feature of the JCE is a synchronous controller which initiates jet pulses of any desired duration, spaced one spin period apart, starting at any desired point in the spin cycle relative to a plane containing the spin axis and spacecraft-sun line. This "start angle" remains constant during any particular maneuver, even though the plane is moving during the maneuver. The essence of this mode, known as a rhumb line precession, is that a given initial attitude and desired final attitude explicitly define the "start angle" for the jet pulses. The JCE synchronous mode, Figure 8, provides the spin synchronous pulse control for the axial jets precession, ΔP, and radial jet velocity, ΔV, modes. The JCE also provides for continuous firing of the axial jets for velocity corrections. The flexibility of the JCE is supplemented with a simple backup "quadrant mode" of operation in the event of a failure in the JCE phase lock counter.

Four sun sensors and a star sensor on the SM provide attitude information. In addition to spin axis attitude data, the sun sensor output pulses are a reference for initiating the JCE. The sensors basically provide two pulses with a time difference proportional to the angle between the spin axis and sun line (or star line). The sensors allow for spin axis attitude determination to within 0.5 degree. The attitude control system also utilizes mercury tube dampers to maintain reasonable nutation histories during maneuvers.

Table 5 summarizes the attitude and velocity control system performance for the spacecraft and SM by itself.

The propellant tanks, lines, thrusters, and associated hardware system are of all-welded construction to minimize weight and leakage. The complete system is tested and then installed into the SM as a single unit. Liquid and gas manifolds and all component attachments, except thrusters, use welded joints. High strength titanium is used for the tanks, lines, and gas and propellant fill valves, and the pressure transducers are of stainless steel. Where titanium-to-stainless steel joints are required, diffusion-bonded transition joints are used. A propellant mass fraction of 0.85 is typical as a result of design efforts to minimize tankage and fixed hardware weight. Catalytic hydrazine thruster designs are available from several suppliers, including Rocket Research, Walter Kidde, and Hamilton Standard, the latter thruster design having been successfully flown on ATS and subjected to a wide variety of spacecraft maneuvers over the past year.

The monopropellant hydrazine system has four 14.2-inch diameter tanks filled to 55 percent with 120 pounds of propellant. The propellant includes 11 pounds of contingency and satisfies the requirements of 65 meters per second for midcourse corrections, 100 meters per second for

TABLE 5. VELOCITY AND ATTITUDE CONTROL, FLYBY - VIDEO DESIGN

	Spacecraft	SM
ΔV per continuous axial jet firing per spin period (one jet)	0.210 fps/rev (25 rpm, 4.5-pound thrust)	0.123 fps/rev (60 rpm, 4.0-pound thrust)
Axial jet continuous firing time per foot per second ΔV	11.3 seconds/fps (one jet) 5.65 seconds/fps (two jets)	8.14 sec/fps (one jet) 4.07 sec/fps (two jets)
Weight of propellant per fps ΔV (axial jet)	0.245 lb/fps	0.081 lb/fps
Attitude change per pulse	1.63 deg/pulse(1) (90-degree pulse) 0.876 deg/pulse(1) (45-degree pulse)	1.39 deg (90-degree pulse) 0.754 deg (45-degree pulse)
Propellant weight per degree of attitude change	0.0154 lb/deg	0.0068 lb/deg
ΔV per 90-degree pulse of radial jet	0.0478 fps/pulse	0.0534 fps/pulse
Propellant weight per radial jet pulse, 90-degree pulse	0.014 lb/pulse	0.00476 lb/pulse
Number of radial jet pulses per fps	20.9 pulses/fps	18.7 pulses/fps
Propellant weight per fps (radial)	0.267 lb/fps	0.089 lb/fps
Time required per fps (radial jet)	49.3 seconds/fps (one jet)	18.9 seconds/fps (one jet)
Spacecraft ΔV per degree of attitude change	0 (nominally, using two axial jets)	0 (two jets produce pure couple)
Maximum nutation angle during reorientation	2 degrees	2 degrees

(1) Using two axial jets to produce pure couple

the deflection maneuver, and the remainder for attitude and spin control operations. Nitrogen gas at 125 to 275 psi. and centrifugal force expel the propellant. Table 6 lists the design characteristics of the reaction control system (RCS).

A typical arrangement of the RCS is illustrated in Figure 9. Shown are two independent units, each consisting of two tanks, five jets, and associated equipment. If an element in one fails in a specified sense, the mission may still be completed with the other unit. Nominal mission performance requires operation of both systems, since they each contain half of the required total propellant load of 120 pounds. The inclusion of a valve between the propellant lines of each unit, similar to that of Intelsat IV, would supply the total propellant to both units. Each tank contains a temperature transducer and each unit a pressure transducer, the data being necessary for utilization of the jet calibration curves.

The spin-up/despin jets operate in pairs to produce a pure couple. Note that a spin change requires both tank systems to produce the pure couple about the spin axis. This arrangement simplifies the plumbing, requiring only one feed line to each spin jet cluster. Should one of the RCS systems be lost, spinup or despin may still be accomplished with a single jet (one jet for spinup, one jet for despin), although the dynamic performance would be somewhat degraded. Loss of one of the RCS systems would not be a serious failure mode for spin changes, provided adequate propellant remained in the other system to effect despin and MSSM spinup.

The axial jets operate in oppositely directed pairs to produce a pure couple for re-orientation and in unidirectional pairs to produce axial ΔV. As in the spinup/despin case, almost double redundancy is available in the axial jets because of the specific combinations of failures required before the mission is seriously degraded.

Deployment of the axial jet clusters may be passively accomplished without the use of electrical signals, squibs, or actuation. This technique was successfully used on two ATS missions for deployment of eight VHF antennas. The flexible connection for the hyrazine line will not present any problem.

Control of the thrusters is effected solely through electrically operated valves, there being at least one valve per thruster. Series redundant valves could be used if desired, as on the Intelsat IV satellite. The four spin jets attached to the mission-critical propellant supply may possibly not be optimum from a reliability standpoint. Series redundant valves might be used on these jets, or the spin jets could be separately supplied by their own small propellant tanks for an additional weight of about 2 pounds. Other elements of the system, including pressure and temperature transducers, pressure charge valve (for nitrogen), and fill and drain valve, are items of existing, flight proven hardware and hardly need consideration in this preliminary study.

TABLE 6. PROPOSED HYDRAZINE REACTION CONTROL SYSTEM DESIGN CHARACTERISTICS

Propellant	Mono propellant hydrazine
Pressurant	Gaseous nitrogen
Feed design	Blowdown pressurization and centrifugal force expulsion (spinup is accomplished by using propellant trapped in lines under pressure to supply centrifugal field on propellant in tanks during initial stages of spinup).
Pressure range	275 to 125 psi
Thrust range	5 to 3 pounds force
Specific impulse range	225 to 200 lbs-sec/lb.
Pulse width	125 milliseconds to 15 minutes
Total impulse delivered	$\approx$ 22,000 lb/sec.
Operating temperature range	40° to 120° F.
Propellant weight	120 pounds
Pressurant weight	2 pounds
Hardware weight	39 pounds
Tank outer diameter (4)	14.2 inches
Initial fill fraction	55 percent

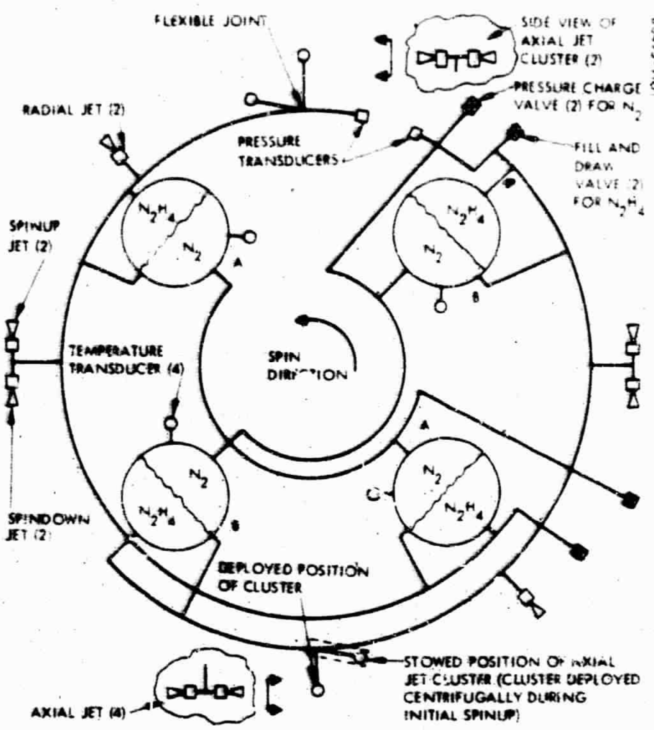

Figure 9. Typical Arrangement of Reaction Control System

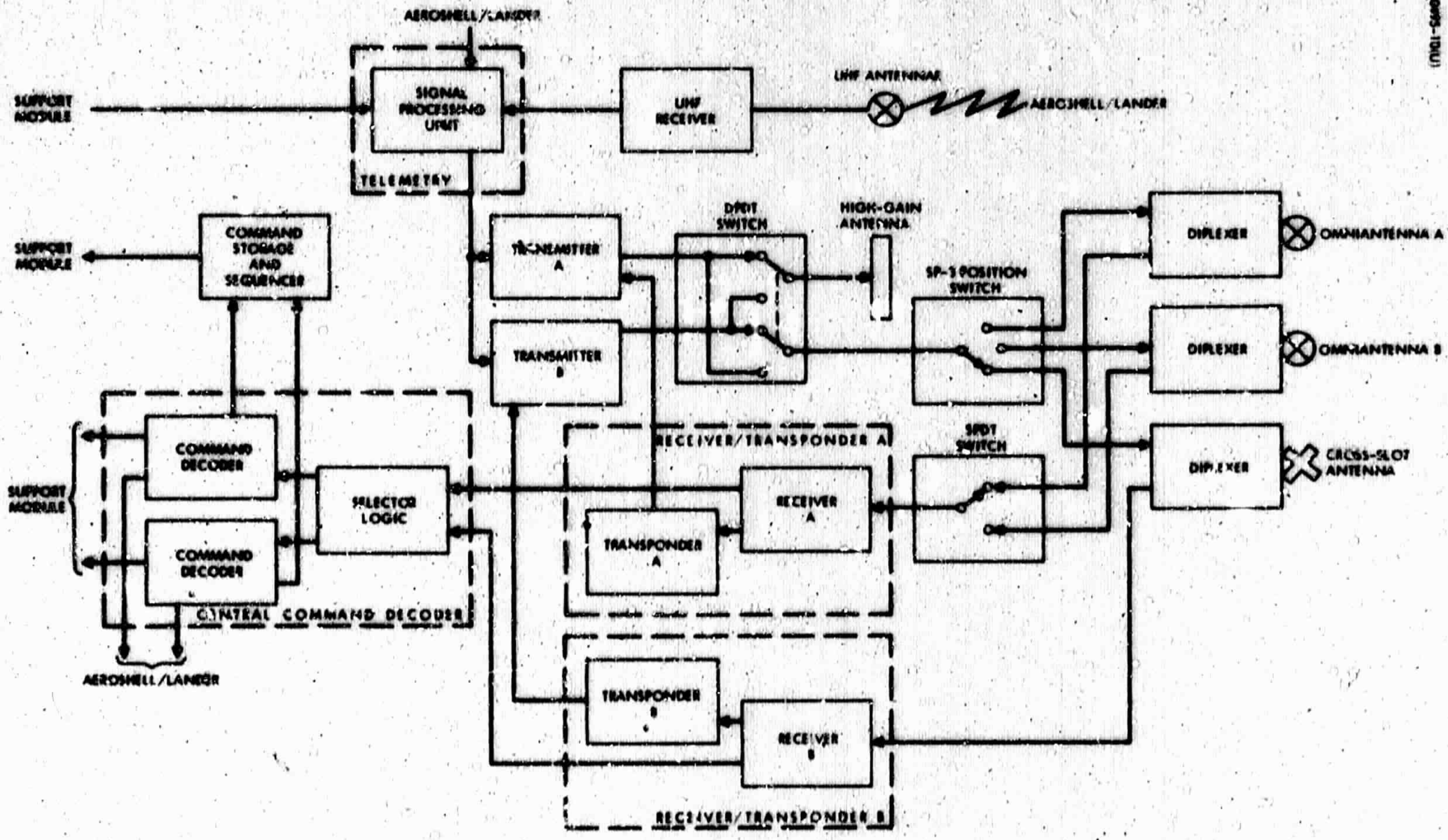

Figure 10. Telecommunications Subsystem

All of the flight control hardware, with the exception of the star sensor, will meet the sterilization requirements, based on past tests and flights under similar environments. A sterilizable photomultiplier tube has been developed by Electromechanical Research, Princeton, N. J., but it has not been tested and flown in a complete sensor unit.

TELECOMMUNICATIONS

The Flyby-Video Data MSSM telecommunications system, which includes not only the transmit and receive functions, but the command decoding, storage and sequencer, and signal processing (telemetry) aspects is based on the extensive S-band experience with the Surveyor program and telemetry/command/VHF know-how with earth communications satellites. Complete redundancy is provided for this vital subsystem by using parallel units and switching between antennas, receivers, and transmitters. Also worst case design techniques are incorporated to ensure link performance. Figure 10 is a block diagram of the SM telecommunications subsystem.

The S-band portion of the subsystem which provide , for maintenance of telemetry and commands between the SM and earth DSN (1973 configuration) consists of a 17.9-square-foot planar antenna having a gain of 29.2 db and a 5.2-degree beamwidth, two 40-watt traveling-wave tube (TWT) high power transmitters, two 3-watt solid state low power transmitters, two receiver/transponders, two backup semiomnidirectional turnstile antennas, and associated diplexers and switches.

The primary function of the high-gain planar antenna is to operate with a 40-watt transmitter and relay video data received from the A/L back to earth over a distance of 160×10^6 kilometers at Mars encounter. In addition, the planar antenna and a 3-watt transmitter are used from 90 days after earth injection to encounter for sending spacecraft data to earth. Since the planar antenna has a small beamwidth that is directed along the spacecraft spin axis, a series of four spacecraft attitude maneuvers must be made to keep the earth within the beamwidth. The video data can be transmitted at 17,000 bits per second, although only 16,000 is required for a 10-minute postlanding video period to obtain 10^7 bits. The planar antenna does not receive data.

Prior to using the planar antenna 90 days after earth injection, the cross slot antenna is used to send telemetry data from the spacecraft to earth. The cross slot is an integral part of the planar antenna and has a gain of at least -1 dB over a 120-degree beamwidth. The cross slot operates with a 3-watt transmitter during most of the mission except for brief high-power, 40-watt periods during midcourse maneuvers. The cross slot antenna is also used with a 40-watt transmitter during the A/L separation period because the planar antenna cannot be held in the earth direction. The cross slot will receive commands in conjunction with the S-band receiver and operate in a transponder loop with a receiver, transmitter, and planar antenna (when required) to produce range/range rate data for the orbit determination process.

TABLE 7. MSSM TELEMETRY LINK PERFORMANCE SUMMARY

Range (km x 10^6)	Down-Link Support Module to DSN (Approximately 2295 MHz) — Maximum Allowable Information Rate (bps) — Receive on 210-foot Antenna: Cross Slot Antenna, Gain, -1 dB, watts: 3	40	Omni-directional Antenna, Gain, -6 dB, watts: 3	40	High-Gain Antenna, Boresight, ±2.6 degrees, watts: 3	40	High-Gain Antenna, Boresight, ±5 degrees, watts: 3	40	Receive on 85-foot Antenna: Cross Slot Antenna, Gain, -1 dB, watts: 3	40	Omni-directional Antenna, Gain, -6 dB, watts: 3	40	High-Gain Antenna, Boresight, ±2.6 degrees, watts: 3	40	High-Gain Antenna, Boresight ±5 degrees, watts: 3	40
10	520	-	200	2.58k	-	-	-	-	150	790	10.7	250	-	-	-	-
40	319	470	4	160	-	-	-	-	0.9	43	0	7.3	1.95k	-	276	-
80	1.3	130	0	33	5.12k	-	735	9.6k	0	4	0	0	490	6.5k	63	915
120	0	53	0	10	2.27k	-	321	4.26k	0	0	0	0	217	2.88k	23	407
160	0	23	0	1.8	1.[illegible]k	17.0k	181	2.4k	0	0	0	0	122	1.62k	9.1	228
175	0	18	0	0	1.07k	14.2k	151	2.0k	0	0	0	0	102	1.35k	6.1	190
200	0	12	0	0	820	10.85k	116	1.53k	0	0	0	0	73	1.08k	2.6	146

Up-Link DSN to Support Module (Approximately 2113 MHz) — Maximum Allowable Command Rate (bps) — Transmit on 210-foot Antenna (400 kw RF): Cross-Slot Antenna (Gain, -1 dB): Range (km x 10^6)	Bit Rate (bps)	Omnidirectional Antenna (Gain, -6 dB): Range (km x 10^6)	Bit Rate (bps)	Transmit on 85-foot Antenna (100 kw RF): Cross-Slot Antenna (Gain, -1 dB): Range (km x 10^6)	Bit Rate (bps)	Omnidirectional Antenna (Gain, -6 dB): Range (km x 10^6)	Bit Rate (bps)
160	1,500	160	475	160	3.3	92.3	3
200	960	200	304	200	1.2	116.3	1

Two semiomnidirectional turnstile antennas located on the SM extendable arms provide complete spherical coverage, with a gain greater than -9 dB, and would be used in the event of a failure in the attitude control system or cross slot area. The antennas are similar to those used on Surveyor.

The switches allow any transmitter to drive any of the antennas, and either semiomnidirectional antenna to be connected to one of the receivers. The other receiver is permanently attached to the cross slot antenna. In the event the signal coming from one receiver disappears, logic will switch in the other receiver.

Table 7 summarizes the SM S-band communications link performance. Performance is given for the 210- and 85-foot DSN systems.

The UHF, 400-MHz portion of the telecommunications subsystem is designed to only receive entry and video data from the A/L. Sixteen half-wavelength dipole antennas are mounted about the solar panel perimeter and transfer the A/L signal to a UHF passive filter/receiver which in turn feeds the data into the S-band system for transmission to earth. The A/L is assumed to use a 50-watt transmitter and cross slot antenna with 0 db at 70 degrees off the antenna beam center. The UHF link will support 16,000 bits per second at a maximum range of 5000 kilometers, which is associated with a flyby trajectory having at least 10 minutes of video transmission time.

The S-band modulation is PCM/PSK/PM and the UHF choice is direct carrier FM, PCM/FSK. The S-band system assumes a commandable modulation index and the use of convolutional-coding to optimize the SM system.

The system includes two command decoders which provide for 168 SM commands and necessary A/L commands. Commands are also stored in the sequencer prior to A/L separation for purposes of having a completely automatic sequence during this phase of the mission. The sequencer stores 30 commands.

The signal processing unit accepts and converts for transmission SM and A/L telemetry data. A total of 166 data channels can be processed. Included in the channels are the star sensor intensity and time data required to determine spacecraft attitude.

The telecommunication equipment weight and sizes are based on presently available items such as the 20-watt TWTs (used in parallel to produce 40 watts) or those being developed, such as the solid-state 3-watt transmitters. In addition, the sterilization requirement can be readily accommodated into the telecommunications system design based on similar requirements for Surveyor S-band and earth satellite equipment.

POWER

The energy sources for the system are solar panels and rechargeable silver zinc batteries. Except for brief periods of high power operation, the solar panels will supply the operational loads associated with the cruise phase of the mission. The 37.4 square feet of panel area is based on the requirement to sustain total system loads, 85 watts, at Mars aphelion and under the anticipated sun/spacecraft orientation. The steady-state cruise mode power requirements are listed in Table 8. The solar panel design is based upon a solar cell conversion factor of 3.30 watts per square foot, normal incidence angle at Mars aphelion, and an area utilization factor of 90 percent. The power conversion figure includes radiation degradation of 15 percent and any voltage mismatch. However, the solar panel is designed to result in a minimum mismatch under the near-Mars condition of solar intensity and panel temperature. A maximum solar incidence angle of 40 degrees from normal at Mars encounter was dictated as a result of trajectory analysis. It was stated as a requirement that 22 watts of unregulated power will be drawn continuously for the A/L battery chargers, thermal control, and sequencer. The power calculations are based upon a continuous drain of 25 watts by the A/L.

Battery capacity requirements are dictated by the system loads during the encounter minus 12 hours through the flyby phase of the mission. A detailed command/time sequence was generated for this phase and used to generate a power and energy histogram. The results appear in Table 9. The results are conservative because the sequence was started at 5 rather than 12 hours before encounter, based on earlier studies. The reduction in battery size and weight using 12 hours is insignificant because only a very low charging rate, 1.7 watt-hours per hour, is available during the additional time gained. To enhance system reliability, two batteries are used, each sized such that the mission can be completed on a single battery if necessary. Additional safety is provided by the use of isolating diodes between each battery and the bus (see Figure 11). In this way, failure of a single battery during the mission will disconnect it from the line, thereby preventing a rapid discharge of the remaining battery.

In determining battery size, a minimum period of 1.5 hours between the end of the SM deflection maneuver and the start of flyby, a battery charging rate of 6 percent and a discharge limited to 80 percent of capacity is assumed. Two batteries with a capacity of 283 watt-hours each (total of 566 watt-hours), satisfy the requirements. Power calculations indicate that the maximum current drain on a battery will be on the order of 5 amperes, assuming single battery operation. Since the capacity of each battery is approximately 12 ampere-hours, the maximum discharge rate will be approximately 40 percent of capacity. Although detailed functional performance characteristics of sterilizable batteries are not presently known, the discharge rate indicated appears to be approximately 50 percent of the anticipated maximum discharge capability.

TABLE 8. CRUISE MODE STEADY-STATE LOADS

Unit	Power Required, watts: Unregulated	Power Required, watts: Regulated
Transmitter (low power, 3 watts)		
Assume 15 percent efficient	----	20.0
Flight control	----	10.0
Telemetry and command	----	6.5
S-band rece vers	3.0	----
Aeroshell/lander	25.0	----
SM thermal control	5.0	----
Battery charge regulator*	1.5	----
Boost regulator		
Assume 75 percent efficient	9.0	----
Subtotals	43.5	36.5
Totals	80.0	
Contingency and high power operation	5.0	
Total	85.0	
Solar panel power to be supplied	85.0	

*Assumes that the battery will be disconnected from the bus (after being fully charged) for most of transit.

The energy-to-weight ratio for a sterilizable silver zinc battery of this capacity is approximately 20 watt-hours per pound; therefore, the total battery weight is 28.3 pounds. Although no fully operational sterilizable batteries exist, the development of fully sterilizable (prolonged soak at 135° C) silver zinc primary and secondary batteries is in an advanced stage. The studies underway at Jet Propulsion Laboratories of battery cells provided by Electrical Storage Battery, Inc., are well documented.

TABLE 9. ENERGY REQUIREMENTS FOR TERMINAL SEQUENCE

Unit	Power, watts		Pre-Encounter Phase		Fly-By Phase	
	Regulated	Unregulated	Time, minutes	Energy, watt-hrs	Time,[a] minutes	Energy, watt-hrs
Transmitter						
40 watts RF, 25 percent efficient	160.0		63.3	168.7	53.7	143.3
3 watts RF, 15 percent efficient	20.0		120.8	40.3	-	-
Flight control						
Cruise mode	10.0		38.0	6.3	53.7	8.9
Thrusting mode	15.0		146.0	36.5	-	-
Telemetry and command decoder	6.5		184.1	19.9	53.7	5.8
Command storage and sequencer	3.0		184.1	9.2	-	-
UHF receivers	2.5		183.9	7.7	53.7	2.2
S-band receivers		3.0	184.1	9.2	53.7	2.7
Battery charge regulator		4.5	184.1	13.8	53.7	4.0
Boost regulator						
P_{reg} = 187 watts (87 percent efficient)		24.3	26.0	10.5	-	-
P_{reg} = 182 watts (87 percent efficient)		23.6	57.3	14.7	-	-
P_{reg} = 46.5 watts (77.5 percent efficient)		10.4	120.0	20.8	-	-
P_{reg} = 41.5 watts (77 percent efficient)		9.5	0.8	0.1	-	-
P_{reg} = 179 watts (87 percent efficient)		23.3	-	-	53.7	20.8
Thermal control		5.0	184.1	15.3	53.7	4.5
Subtotals				373.0		192.2
Solar panel		85.0	184.1	260.5	53.7	76.0
Battery energy required				112.5		116.2

[a]Times are based on 30 minutes total visibility after lander touchdown.

Overcharging the batteries will be prevented by the battery charger/booster which will contain a charge current limiting function. It is anticipated at this time to essentially disconnect the batteries from the bus after being fully charged. Although a thorough detailed knowledge of sterilizable silver zinc battery characteristics does not presently exist, it is felt that a long period of float charging could prove detrimental. When the solar panel output is insufficient to sustain system loads, the batteries will automatically be connected to the bus. This procedure is similar to that followed on the Mariner spacecraft.

The boost regulator provides regulated power to the transmitters, UHF receivers, and the command storage and sequencer, as well as the flight control and telemetry subsystems. The concept of a central regulator was chosen primarily on the basis of minimizing overall cost, weight, and thermal and electrical dissipations. Additional advantage of the existing redundancy of the transmitter can be realized by having additional separate regulators with current limiting for each transmitter. In this way, a short circuited transmitter could be removed from the bus and the system could continue operating on the remaining transmitter. However, there would be attendant penalties in the areas previously mentioned.

The steady state regulated power requirement for cruise mode operation is 36.5 watts. Assuming a regulated bus voltage of 30 ± 1 percent, the boost regulator output current is approximately 1.2 amperes. The maximum regulated load requirement is expected to be 6 or 7 amperes. It is estimated that an efficiency of 75 percent is reasonable at the lower output, while 85 to 90 percent can be achieved at the higher output.

Although not a part of the SM power subsystem, two A/L battery chargers are incorporated into the SM design.

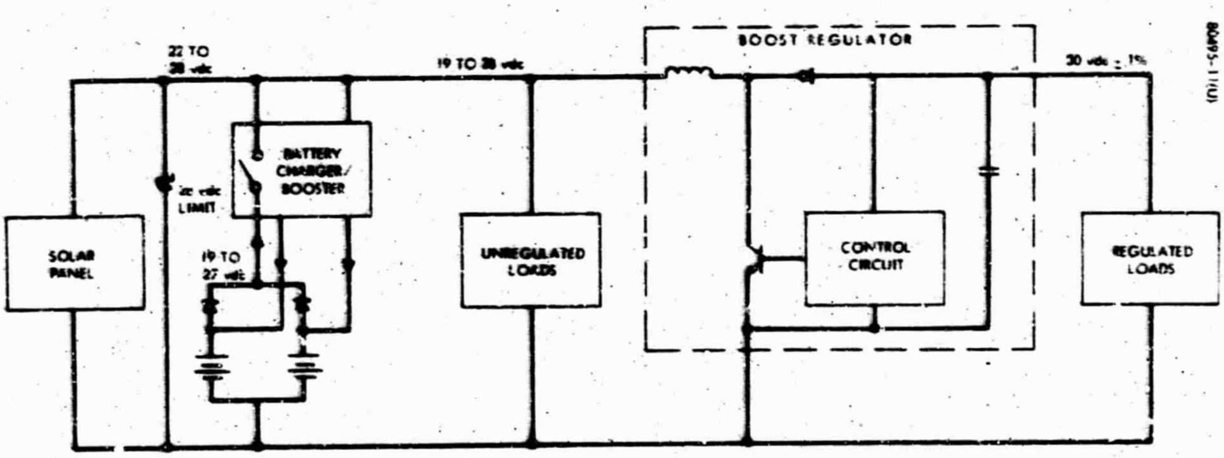

Figure 11. Power Subsystem Functional Block Diagram

6. FLYBY DESIGN - ENTRY DATA

The requirement to relay only entry data, at a rate of 2000 bits per second, and not video data at 16,000 bits per second, relaxes the specifications on the telecommunications subsystem, thereby resulting in a few differences between the Flyby - Entry and Video Data MSSM designs and mission profile. The mission profile modifications are a fewer number of spacecraft attitude corrections to maintain the planar antenna directed at the earth and, obviously, no video transmission sequence. A smaller number of attitude corrections are possible because the lower gain (22.2 dB) planar antenna with its larger beamwidth (11.5 degrees) will keep the earth in view for longer periods of time.

The Flyby - Entry Data SM inboard profile is shown in Figure 12. A 23-inch square planar antenna is mounted against the louvers which are attached to the equipment shelf. The antenna and surrounding frustrum cover provide sufficient surface area for radiation of thermal energy transferred from the equipment shelf via the louvers. The frustrum cover is coated with a high emissivity material, such as the quartz mirror used on Surveyor, to enhance radiation. Since lower wattage transmitters (20 watts) are used with consequently less thermal output, they are mounted to the equipment shelf and only the louvers between the shelf and antenna are required to regulate the compartment temperature. The cross slot can be mounted at the center of the planar, but is mounted at the side for this SM configuration.

The weights which constitute the total Titan III-C launch weight for the Flyby - Entry Data SM design are listed in Table 10. The SM weight of 521 pounds, which includes 47 pounds of contingency, plus the A/L, adapter, and canister weights, yields a total launch weight of 1989 pounds. The difference between the Titan III-C payload weight capability and the actual launch weight added to the 47 pounds results in a total SM, adapter, and canister weight contingency of 138 pounds.

Table 11 lists the basic subsystem weights of the SM.

Table 12 lists the moment of inertia properties for several SM loading conditions. As in the case of the Video Data design, the data indicates passive stability about spin axis.

TABLE 10. MSSM - LAUNCH WEIGHT DISTRIBUTION, FLYBY-ENTRY DESIGN

Item		Weight, pounds
Aeroshell/lander		1100.0
SM weight (includes 47 pounds contingency)		521.0
SM/booster adapter		117.2
Structure	77.2	
Umbilical	35.0	
Pyrotechnic separation logic	5.0	
Aft canister		181.8
Structure	125.8	
Pressure and vent.	50.0	
Separation harness	3.0	
Forward canister		69.0
Launch weight		1989.0
Titan payload	2080	
Total contingency for SM, adapter and canister		138

The attitude-velocity control and propulsion subsystem differ from the Video Data design in that smaller tanks (13.4-inch diameter) will hold the lighter propellant load, and dynamic performance is changed slightly. Since the performance data change is minor, all data presented in Section 5 under Attitude and Velocity Control/Propulsion, is directly applicable to the Entry Data design with the implication that a conservative performance is specified.

The reduction in bit rate to 2000 bits per second results in a planar array design having 22.2 dB of gain, 11.5 degrees of beamwidth, and a 23 by 23 inch surface area. A transmitter producing 20 watts of output power is used with the planar to perform the entry data relay function. The link performance data is listed in Table 13. All other aspects of the S-band system are identical to those of the Video Data design; therefore, the results of Section 5 under Telecommunications apply.

The UHF link specifies an A/L 35-watt transmitter for the Entry Data design case. This power will more than sustain the 2000 bit per second rate for the same SM UHF system and geometry specified in Section 5 under Telecommunications. These results can be used to change the A/L, SM, or flyby trajectory design to the overall advantage of the Mars 1973 program.

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TABLE 11. MSSM SUBSYSTEM WEIGHT ALLOTMENT, FLYBY DESIGN – ENTRY DATA

Item	Weight, pounds	
RF Subsystem		51.5
3-watt transmitter (2)	22.0	
20-watt transmitter (2)		
High-gain antenna (1)	4.0	
UHF antennas (16)	3.0	
Turnstile antenna (2)	1.0	
S-band receiver (2)	8.0	
UHF receiver (2)	4.7	
RF switches (3)	1.5	
Coaxial cables and attach clips	5.5	
Cross slot omni	0.5	
Diplexer (3)	1.3	
Signal Processing		30.7
Central command decoder (2)	5.7	
Central signal processor	10.0	
Command storage and sequencer (S/M)	5.0	
Sequencer (A/L)	10.	
Propulsion		146.0
5-pound thrusters (10)	11.0	
13.9-inch diameter tanks (4)	18.0	
Valves, lines, filters, and pressurant	8.0	
Propellant/pressurant (2 pounds)	109.0	
Flight Control		30.0
Flight control electronics	17.0	
Star sensors (1)	5.1	
Sun sensors (4)	1.0	
Local cables, connectors, and clips	3.4	
Nutation damper	2.5	
Accelerometer and acceleration switch	1.0	
Structure		116.4
Upper ring	11.7	
Upper cylinder	21.1	
Central ring	11.6	
Lower frustum	4.2	
Equipment shelf	21.0	
Thermal blankets and tie downs	4.5	
Lower thermal blanket support	6.5	

TABLE 11. MSSM SUBSYSTEM WEIGHT ALLOTMENT, FLYBY DESIGN - ENTRY DATA - Concluded

Item	Weight, pounds	
Harness	10.0	
Solar panel support tubes	7.0	
Balance weights	4.0	
Thermal control louver (5)	13.6	
Radial jet supports (2)	1.2	
Mechanisms		14.8
Separation mechanism	11.6	
SM/lander 4.8		
SM/adapter 6.8		
Axial jet deployment (2)	3.2	
Electrical Power		84.1
Solar panel (substrate, cells, interconnections)	36.8	
Boost regulator	8.5	
Battery charge regulator (1 S/M, 2 A/L)	10.5	
Battery (2) 40 percent discharge	28.3	
SM weight	473.5	
Contingency	47.5	
Total	521.0	

The Flyby - Entry Data design power system is identical to that of the Video Data design. It is recognized that the Entry Data SM requires less battery capacity because a 20-watt, instead of 40-watt, transmitter is used for a shorter period of time (entry data period only). Less capacity implies lower battery weight, but since the reduction is minor and even less important to the total SM weight, the difference is of no consequence for purposes of this study.

TABLE 12. MOMENTS OF INERTIA, FLYBY-ENTRY DESIGN

Vehicle State	Slug $-$ ft^2
A/L and SM	
Full fuel	
I_{Roll}	380
I_{Pitch}	287
40 percent fuel	
I_R	374
I_P	283
SM	
40 percent fuel	
I_R	67
I_P	35
Zero percent fuel	
I_R	61
I_P	32

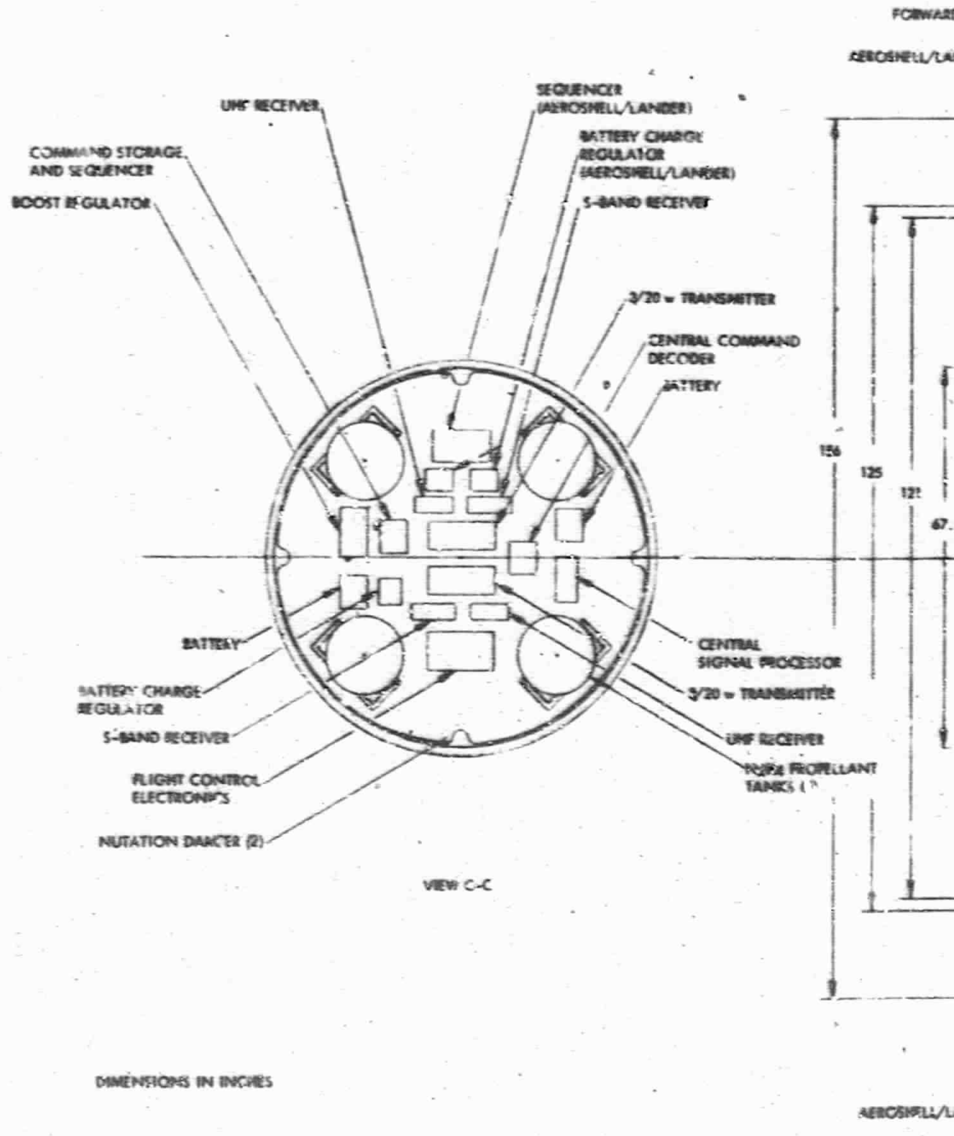
TABLE 13. MSSM TELEMETRY LINK PERFORMANCE SUMMARY (210-FOOT ANTENNA)

Mission Phase	Range ($\times 10^6$ km)	Maximum Allowable Information Rate, bps					
S-band Link to DSIF (Approximately 2295 MHz)							
	Antenna Gain and RF Output Power Conditions	Low-gain Antennas[1] (-6dB Worst Case)			High-Gain Antenna (22.17 dB On-Axis)		
		3 watts RF	20 watts	40 watts	3 watts	20 watts	40 watts
Transit I	≤ 10	240	1600				
First midcourse	10	240	1600				
Transit II	≤ 100		43[2] / 7	98[2] / 23	640	4280	
Second midcourse	100		43[2] / 7	98[2] / 23	640	4280	
Transit III	≤160		11[2]	31[2] / 3.7	250	2000	
Descent	160				250	2000	4000
2000 bps UHF Descent Relay Link (Approximately 400 MHz)							
Descent	6100 km	-4.2 dB MSSM worst case gain, 35 watts RF, 0 dB worst case lander gain					
Command Up Link (Approximately 2113 MHz)							
DSN-to-MSSM up link	160	-9 dB worst case MSSM gain, 100 kW,[3] DSIF RF power, 3 bps -1 dB worst case MSSM gain, 34 bps					

[1] Either cross-slot or boom-mounted omnidirectional antennas, over the earth-directed hemisphere.

[2] Assumes that gain of cross-slot antenna is always ≥ -1 dB in look angle cone.

[3] 400 kW will significantly increase command bit rate.



VIEW C-C

DIMENSIONS IN INCHES

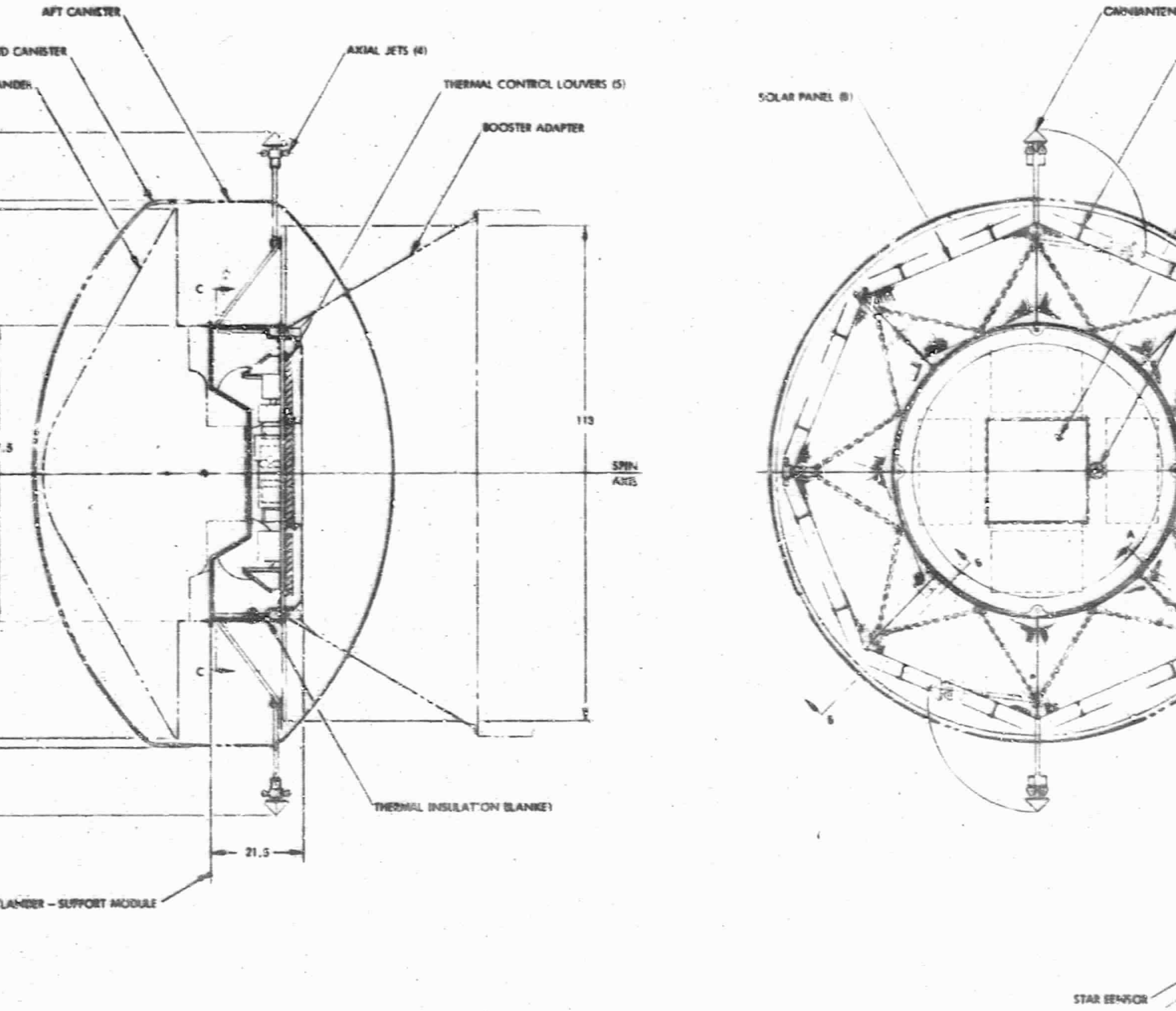
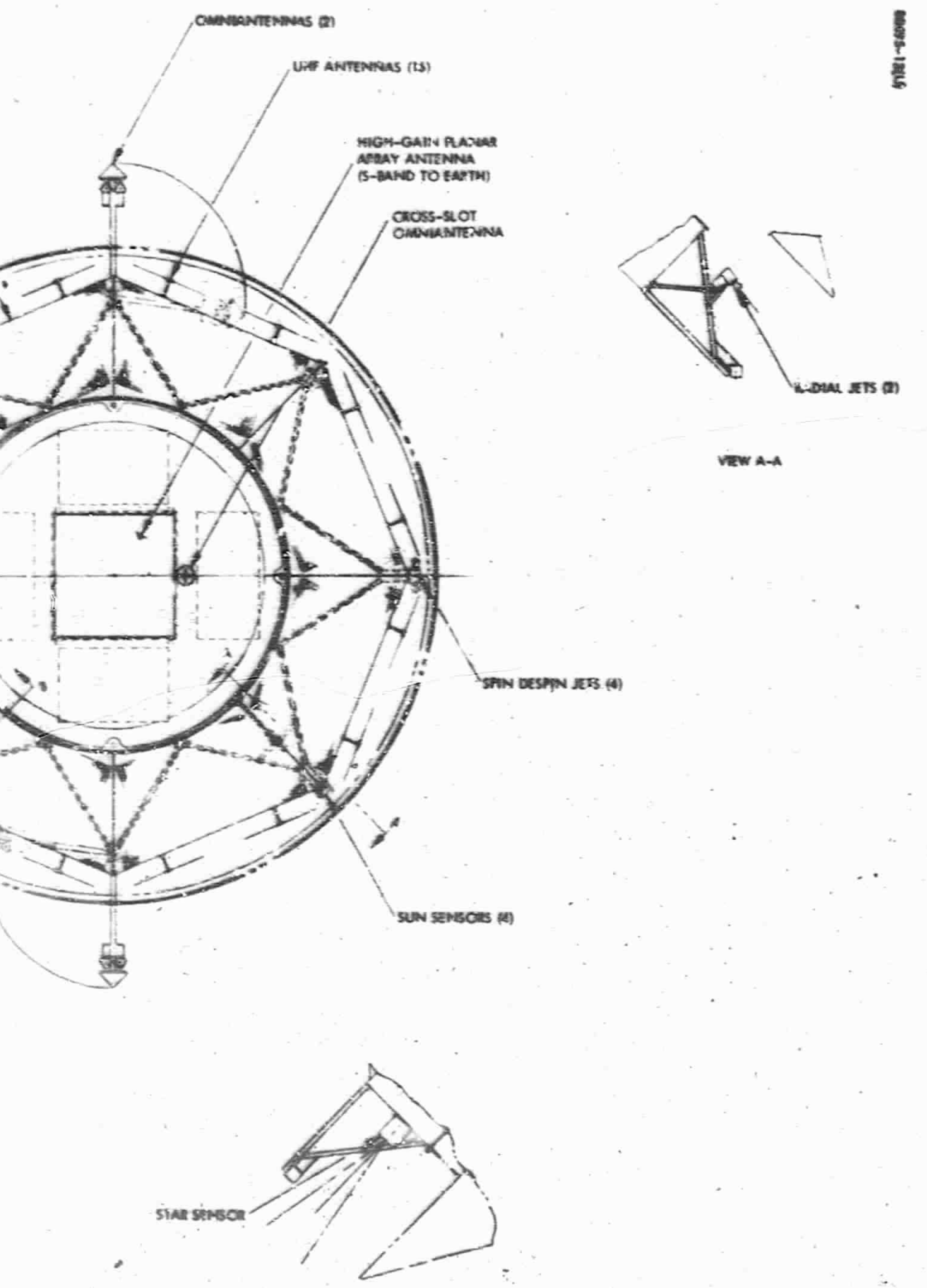


Figure 12. Flyby – Entr



re 12. Flyby – Entry Data SM Inboard Profile

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7. ORBITER DESIGN

The Orbiter MSSM design, in addition to relaying entry and video data, must perform a braking maneuver that places the SM into a synchronous (24.6-hour period) Mars orbit. The SM passes over the lander each day and relays data to earth for the lifetime of the lander, which is at least 3 days. This mission objective, plus the use of the Flyby Mode of delivery which eliminates the need to sterilize the SM, results in an encounter phase mission profile and SM design that significantly differs from the Flyby SM designs.

Figure 13 illustrates the encounter and orbital phase of the Orbiter SM mission. Twelve hours before the relay function begins, the SM performs an attitude rotation to align the A/L for a deflection maneuver. Since the Flyby Mode of delivery is used, the A/L must perform a deflection maneuver to attain a landing trajectory and a subsequent attitude rotation and despin operation to satisfy entry requirements. Then the SM is reoriented to its cruise position, separates the aft canister, and completes the relay phase in a manner similar to that of the Flyby SM design.

The insertion into Mars orbit is subordinate to the transmission of video data to earth. The insertion events will not begin until 10^7 bits of video data have been relayed or, even more desirable, until the A/L to SM communication link is lost due to the range increasing during flyby. A period of 3 to 5 minutes is required to align the SM, followed immediately with a high-energy propulsion system thrusting period. The SM is then reoriented to point the planar antenna at the earth and relay data from the A/L approximately once every 24.6 hours for 3 days. An orbital trim maneuver to compensate for excessive insertion errors and periodic velocity/attitude adjustments may be required during the Orbiter lifetime.

Two Orbiter SM designs have been briefly investigated, one using a solid propellant and the other a liquid propellant orbital insertion system. The solid propellant Orbiter SM inboard profile is shown in Figure 14. As in the case of the Flyby SM designs, the Orbiter SM is mounted between the adapter and A/L, but is outside of the canisters because the sterilization requirement has been waived. The cylindrical section has been lengthened to 38 inches to accommodate the solid propellant engine located at the center of the SM. A short conical frustrum transfers loads from the engine to the cylinder. A smaller cylindrical section surrounds the solid engine and

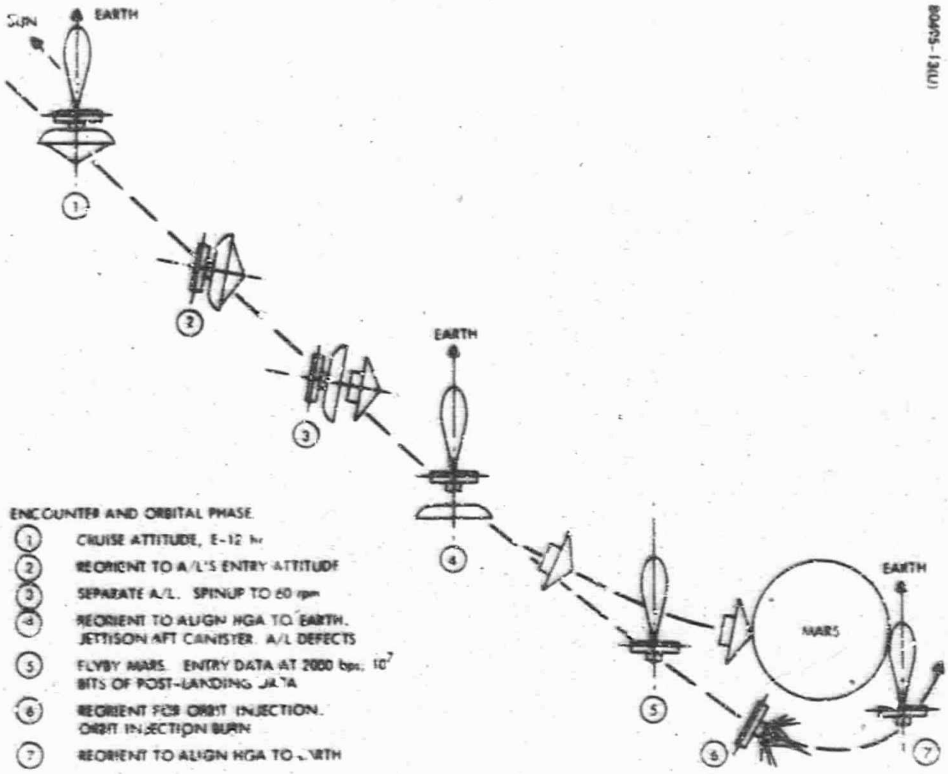
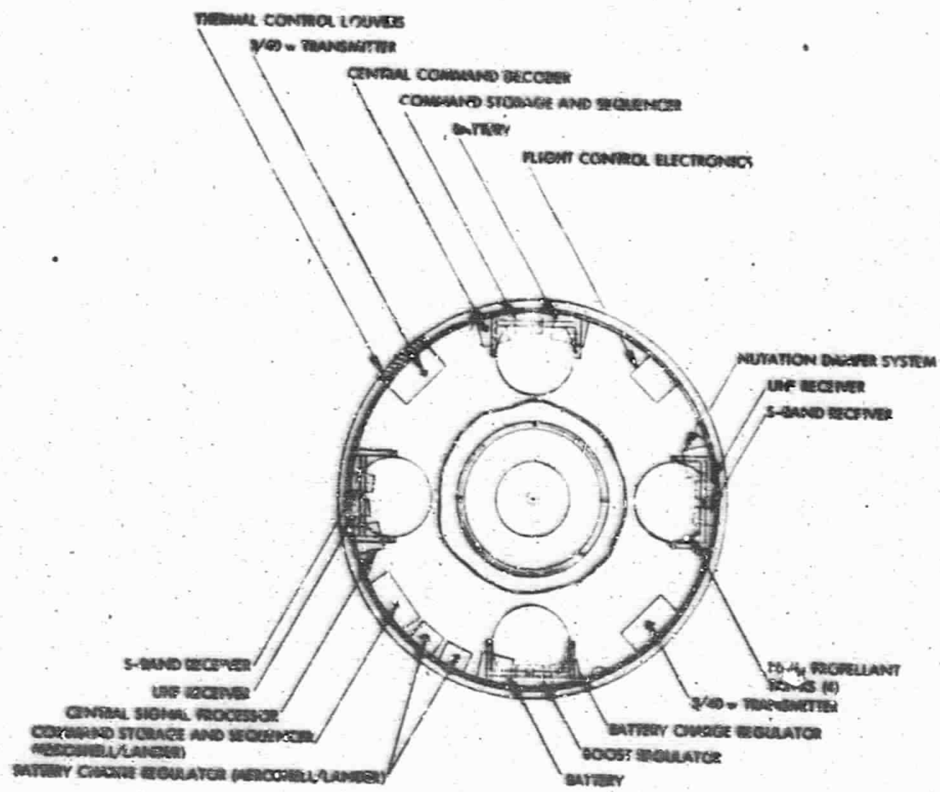


Figure 13. Mission Profile, Encounter and Orbital Phase



VIEW B-B

DIMENSIONS IN INCH

51A

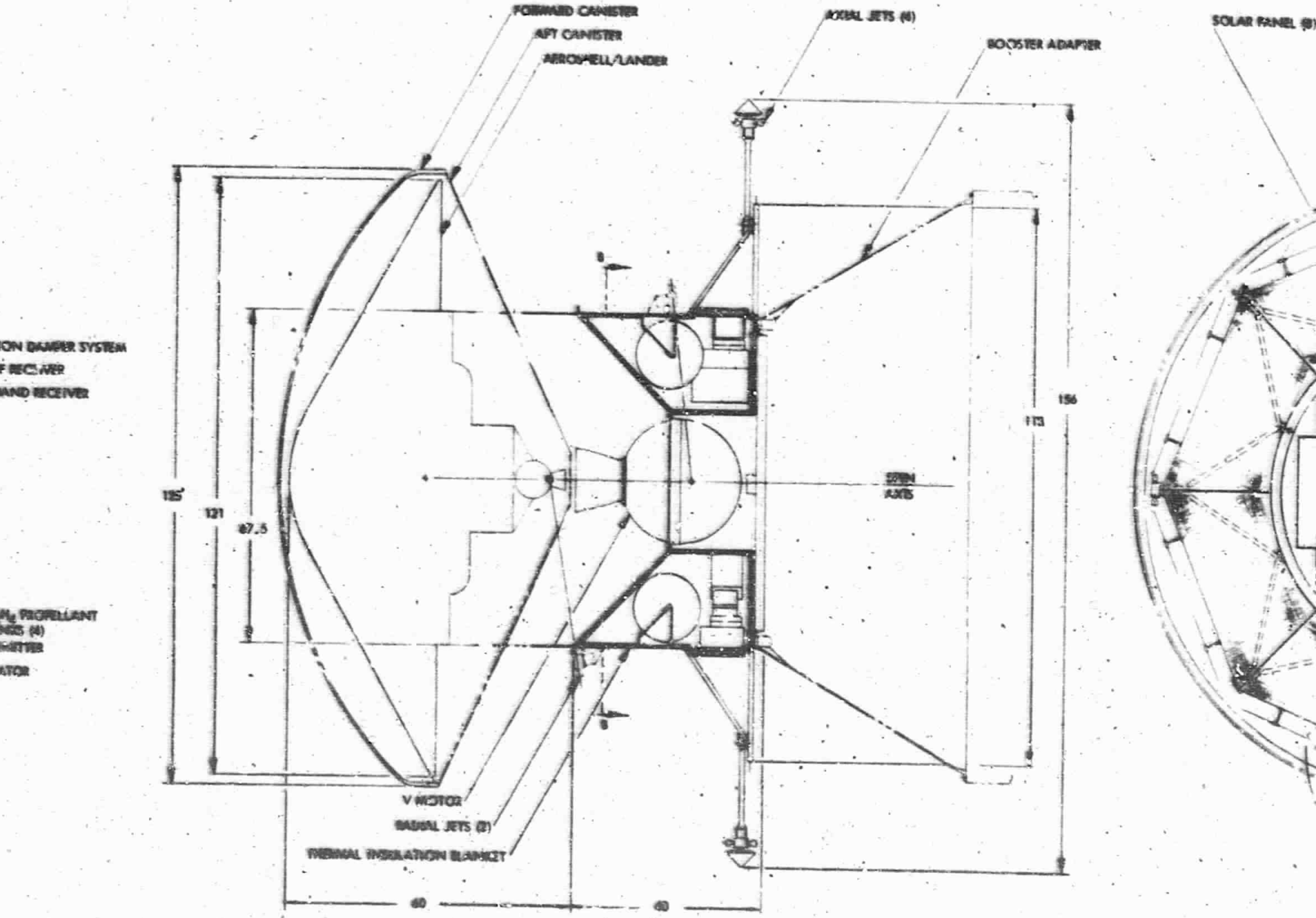

FORWARD CANISTER
AFT CANISTER
AEROSHELL/LANDER
AXIAL JETS (4)
BOOSTER ADAPTER
SOLAR PANEL (8)
SPIN AXIS
V MOTOR
RADIAL JETS (2)
THERMAL INSULATION BLANKET
115
121
87.5
113
156
60
60

Figure 1

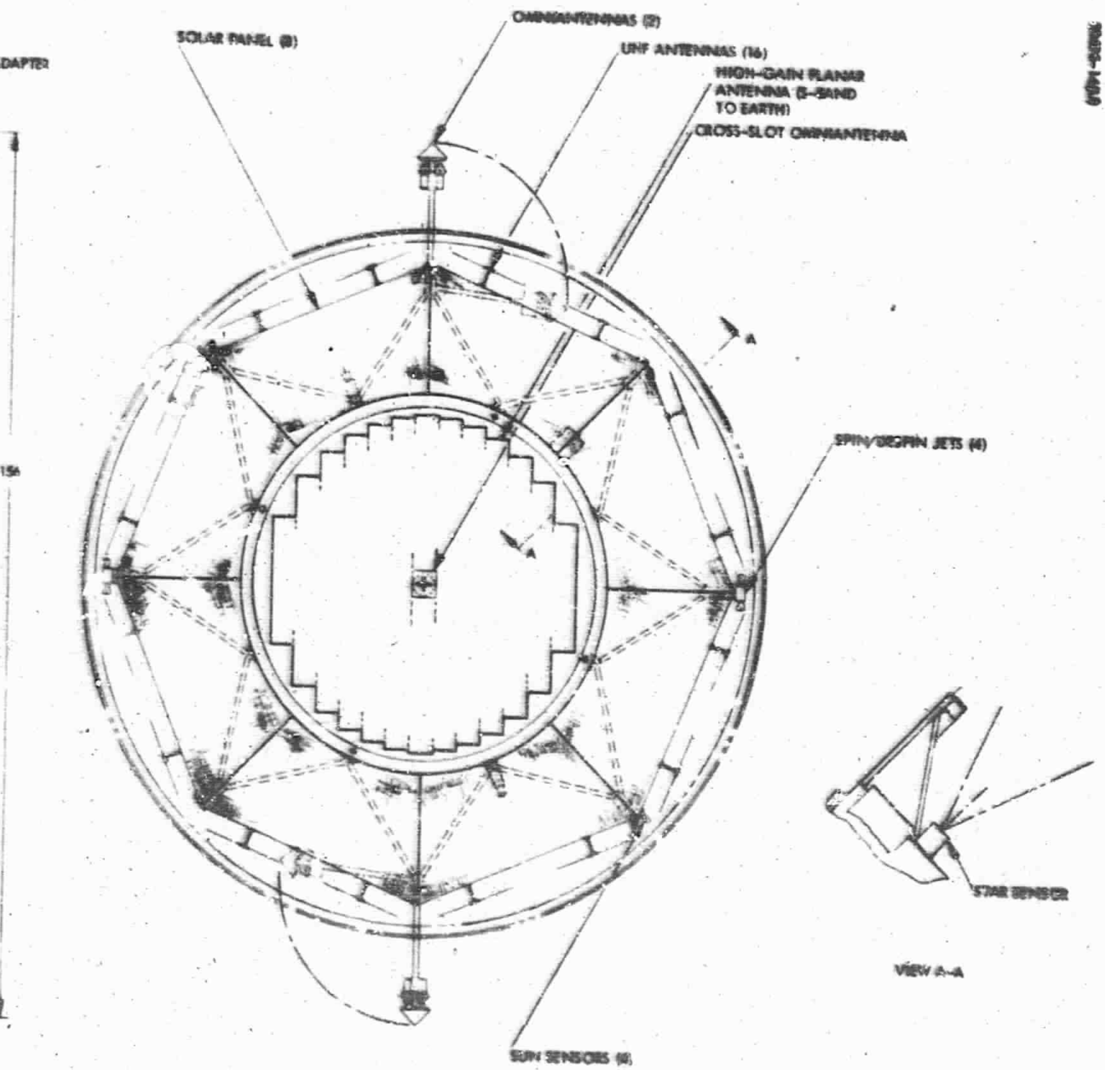


Figure 14. Solid Propellant Orbiter SM Inboard Profile

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provides planar antenna and thermal blanket support plus thermal isolation for the equipment compartment and engine. The engine also requires a thermal blanket and heaters.

The equipment inside the SM is mounted to the outer cylinder wall which contains all the thermal louvers used to control the compartment temperature. This arrangement might prove attractive enough for incorporation into the Flyby designs. Aside from slight mounting differences of the planar antenna and radial jets, all other gross characteristics of the Orbiter design are identical to those of the Flyby - Video Data SM design.

A breakdown of the launch weight for the Orbiter system is presented in Table 14. The A/L weight of 1186 pounds, includes 86 pounds for a deflection system and interface structure. The orbiter SM weight is 1120 pounds, including 90 pounds for contingencies. The inclusion of 344 pounds for the adapter and canisters yields a launch weight of 2650 pounds, which is identical to the Titan III-C payload capability. Therefore, 90 pounds is available for SM, adapter, and canister contingencies, assuming a fixed A/L weight.

The basic Orbiter SM weights are listed in Table 15. The planar antenna and structural weights do not reflect exactly the configuration shown in Figure 14. Twenty-two pounds has been added to accommodate a slightly larger antenna and solar panel outer diameter. This change allows Orbiter SM communications for the greater distances associated with this design.

Table 16 presents the SM moments of inertia and indicates the spacecraft is not passively stable about the spin axis. A simple active nutation damper is included in the design to maintain stability about the spin axis.

The general description of the flight and reaction jet control subsystem presented in Section 5 is applicable to the Orbiter SM design, but the performance characteristics are different, as noted in Table 17. The solid engine weighs 468 pounds and can impart 5630 fps to the SM.

The telecommunications subsystem requires a planar antenna with a gain of 30 dB (21.2 square feet) and this results in link performance slightly better than that given in Section 5. With the exception of a slightly different automatic sequencer, the remainder of the telecommunications subsystem described in Section 5 is valid for the Orbiter design.

The power subsystem of the Orbiter SM is functionally identical to that described in Section 5 under Power. The batteries are 11 pounds lighter because sterilization is not required.

An alternate Orbiter SM design using a liquid braking system was considered with an inboard profile as shown in Figure 15. A weight breakdown is given in Table 18. Due to study limitations, the liquid system study was not pursued in depth. This system is envisioned using a monopropellant for all pre-orbiting maneuvers and then adding an oxidizer to produce a high specific impulse for the orbital insertion phase. Two fuel and two oxidizer tanks, plus a gas bottle for the pressurant, are required. In addition to the low thrust jets, two 200-pound thrust engines are mounted outside the SM cylindrical section. This SM design allows for a closer attachment to the A/L and possibly a more stable spinning spacecraft. The flexibility of the liquid design may warrant further studies.

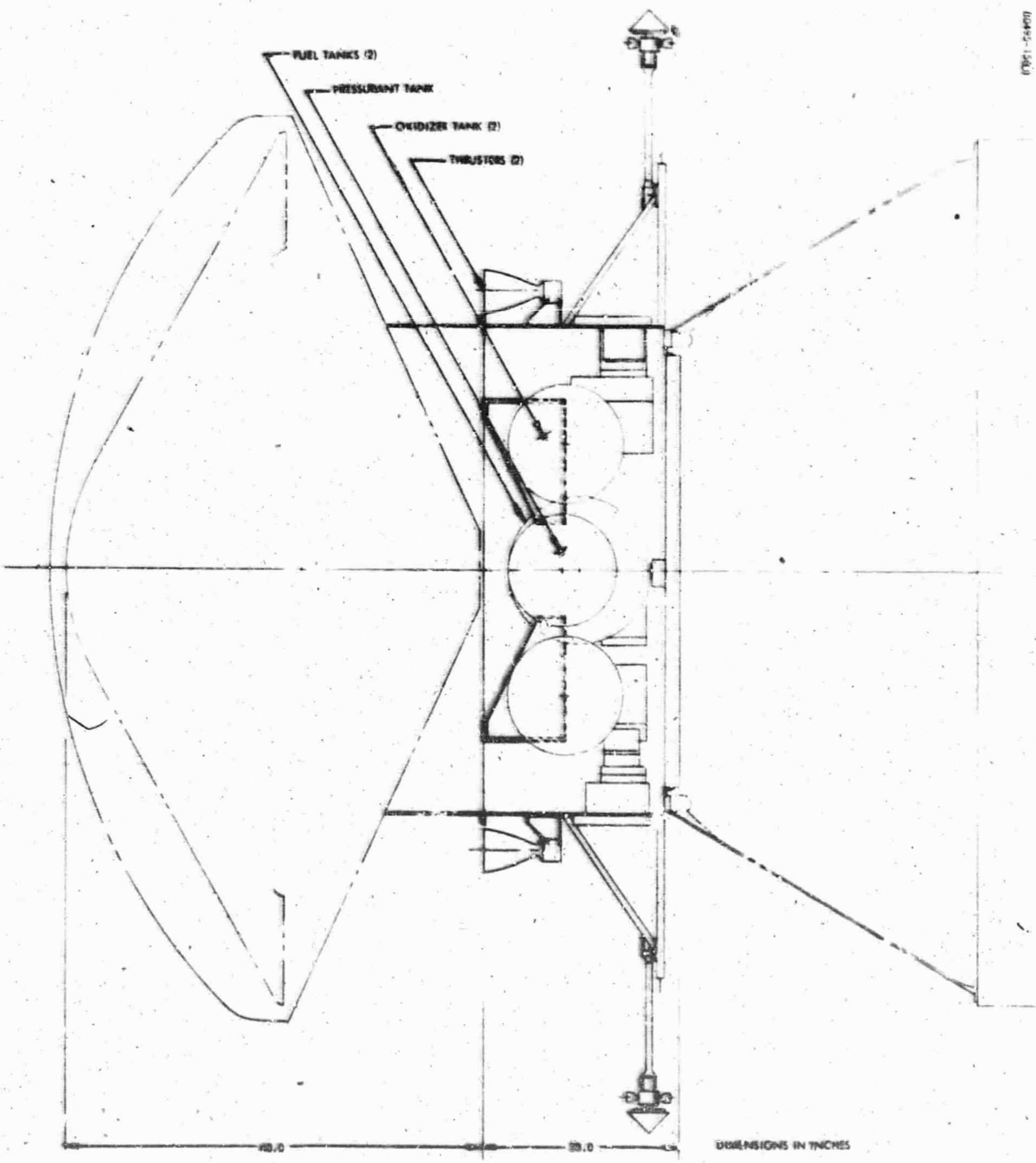

Figure 15. Liquid Braking System Orbiter SM Design

TABLE 14. ORBITER (SOLID) LAUNCH WEIGHT DISTRIBUTION

Item	Weight, pounds	
Aeroshell/lander		1186
SM weight (including contingency of 90 pounds)		1120
SM/booster adapter		125
Structure	85.0	
Umbilical	35.0	
Pyrotechnic separation logic	5.0	
Aft canister		150
Structure	97.0	
Pressure and vent	50.0	
Separation harness	3.0	
Forward canister		69
Launch weight		2650
Titan payload	2650	
Total contingency for SM, adapter, and canisters		90

TABLE 15. ORBITER (SOLID) SUBSYSTEM WEIGHT ALLOTMENT

Item		Weight, pounds	
RF Subsystem			83.3
3-watt transmitter (2)		41.2	
40-watt transmitter (2)			
High-gain antenna (1) (21.2 ft^2)		16.6	
UHF antennas (16)		3.0	
Turnstile antenna (2)		1.0	
S-band receiver (2)		8.0	
UHF receiver (2)		4.7	
RF switches (3)		1.5	
Coaxial cables and attach clips		5.5	
Cross slot omni		0.5	
Diplexer (3)		1.3	
Signal Processing			30.7
Central command decoder (2)		5.7	
Central signal processor		10.0	
Command storage and sequencer (S/M)		5.0	
Sequencer (A/L)		10.0	
Propulsion			632.7
5-pound thrusters (10)		11.0	
14.2-inch diameter tanks (4)		20.7	
Valves, lines, filters, and pressurant		8.0	
N_2H_4 propellant		115.0	
Solid propellant		430.0	
Case and inerts		48.0	
Flight Control			37.5
Flight control electronics		17.0	
Star sensors (1)		5.1	
Sun sensors (4)		1.0	
Local cables, connectors, and clips		3.4	
Nutation damper (2) (active)		10.0	
Accelerometer and acceleration switch		1.0	
Structure			149.2
Upper ring		12.7	
Upper cylinder		4.0	
Engine frustum and ring		27.4	
Inter cylinder and ring		5.1	
Lower ring		9.2	
Thermal blankets and tie downs		5.5	
Thermal blanket support		6.1	
Harness		15.0	
Solar panel support tubes		7.0	
Balance weights		4.0	
Thermal control louver (15 square feet)		12.0	
Radial jet supports (2)		1.2	
Equipment support brackets and hardware		4.0	
Mechanisms			14.8
Separation mechanism		11.6	
SM/lander	4.8		
SM/adapter	6.8		
Axial jet deployment (2)		3.2	
Electrical Power			81.8
Solar panel (substrate, cells, interconnections)		45.8	
Boost regulator		8.5	
Battery charge regulator (1 S/M 2 A/L)		10.5	
Battery (2) 40 percent discharge		17.0	
Subtotal			1030.0
Contingency			90.0
Total			1120.0

TABLE 16. ORBITER (SOLID) MOMENTS OF INERTIA

Vehicle State	Slug-ft^2
A/L and Orbiter	
Full Fuel	
I_{Roll}	471
I_{Pitch}	730
Midcourse, fuel removed	
I_R	454
I_P	720
Orbiter alone at separation	
I_R	84
I_P	58.3
Orbiter-liquid and solid fuel used	
I_R	67
I_P	50.0

TABLE 17. ORBITER VELOCITY AND ATTITUDE CONTROL

	Support Module Only After Orbit Injection	Support Module Only Before Orbit Injection	Support Module Attached to Aeroshell/Lander
ΔV per continuous axial jet firing per spin period (one jet)	0.266 fps/revolution (60 rpm, 4.5 pounds thrust)	0.138 fps/revolution (60 rpm, 4.5 pounds thrust)	0.14 fps/revolution (25 rpm, 4.5 pounds thrust)
Axial jet continuous firing time per fps ΔV	3.73 sec/fps (one jet) 1.87 sec/fps (two jets)	7.19 sec/fps (one jet) 3.59 sec/fps (two jets)	16.9 sec/fps (one jet) 8.45 sec/fps (two jets)
Weight of propellant per fps ΔV	0.0809 lb/fps	0.176 lb/fps	0.365 lb/fps
Attitude change per pulse	5.86 degree/pulse[1] (90-degree pulse) 3.14 degree/pulse[1] (45-degree pulse)	3.14 degree/pulse[1] (90-degree pulse) 1.69 degree/pulse[1] (45-degree pulse)	1.56 degree/pulse[1] (90-degree pulse) 0.842 degree/pulse[1] (45-degree pulse)
Propellant weight per degree of attitude change	0.0043 lb/degree	0.008 lb/degree	0.016 lb/degree
ΔV per 90-degree pulse of radial jet	0.06 fps/pulse	0.0315 fps/pulse	0.032 fps/pulse
Propellant weight per radial jet pulse (90-degree pulse)	0.00483 lb/pulse	0.00483 lb/pulse	0.014 lb/pulse
Number of radial jet pulses per fps	16.5 pulses/fps	31.8 pulses/fps	31.4 pulses/fps
Propellant weight per fps (radial)	0.093 lb/fps	0.179 lb/fps	0.40 lb/fps
Time required per fps (radial jet)	16.5 sec/fps (one jet) 8.25 sec/fps (two jets)	31.8 sec/fps (one jet) 15.9 sec/fps (two jets)	74 sec/fps (one jet) 37.sec/fps (two jets)
Spacecraft ΔV per degree of attitude change	0 (nominally, using two axial jets)	0 (nominally, using two axial jets)	0 (nominally, using two axial jets)
Maximum nutation angle during reorientation	1.5 degrees	1.5 degrees	2.0 degrees

[1] Using two axial jets to produce pure couple.

TABLE 18. ORBITER (LIQUID) SUBSYSTEM WEIGHT ALLOTMENT

Item	Weight, pounds	
RF Subsystem		83.3
3-watt transmitter (2)	41.2	
40-watt transmitter (2)		
High-gain antenna (1)	16.6	
UHF antennas (16)	3.0	
Turnstile antenna (2)	1.0	
S-band receiver (2)	8.0	
UHF receiver (2)	4.7	
RF switches (3)	1.5	
Coaxial cables and attach clips	5.5	
Cross slot omni	0.5	
Diplexer (3)	1.3	
Signal Processing		30.7
Central command decoder (2)	5.7	
Central signal processor	10.0	
Command storage and sequencer (S/M)	5.0	
Sequencer (A/L)	10.0	
Propulsion		609.0
5-pound thrusters (10)	11.0	
Tanks (2 oxidizer, 2 fuel, 1 pressurant), regulator, valves, lines, filters, and pressurant	61.7	
200-pound thrusters (2)	536.3	
Midcourse and attitude		
ΔV_R		
N_2H_4 fuel		
N_2O_4 oxidizer		
Flight Control		37.5
Flight control electronics	17.0	
Star sensors (1)	5.1	
Sun sensors (4)	1.0	
Local cables, connectors, and clips	3.4	
Nutation damper (2) (active)	10.0	
Accelerometer and acceleration switch	1.0	
Structure		146.1
Upper ring	12.7	
Upper cylinder	28.0	
Engine and tank support structure	32.0	
Inner support structure		
Lower ring	9.2	
Thermal blankets and tie downs	6.0	
Thermal blanket support	15.0	
Harness	15.0	
Solar panel support tubes	7.0	
Balance weights	4.0	

TABLE 18. ORBITER (LIQUID) SUBSYSTEM WEIGHT ALLOTMENT - Concluded

Item		Weight, pounds	
Thermal control louver (15 square feet)		12.0	
Radial jet supports (2)		1.2	
Equipment support brackets and hardware		4.0	
Mechanisms			14.8
Separation mechanism		11.6	
SM/lander	4.8		
SM/adapter	6.8		
Axial jet deployment (2)		3.2	
Electrical Power			81.8
Solar panel (substrate, cells, interconnections)		45.8	
Boost regulator		8.5	
Battery charge regulator (1 S/M 2 A/L)		10.5	
Battery(2) 40 percent discharge		17.0	
Subtotal			1003.2
Contingency			98.1
Total			1101.3

8. PROGRAM PLAN

An MSSM development plan and the key program milestones are shown in Figure 16. The program assumes that three flight SMs, one of which is a backup, will be delivered to a prime contractor (capsule integrator) in late 1972 to be mated with landing vehicles and launched toward Mars in mid-1973. The program begins with a phase C effort to define system and subsystem specifications and procure long lead time items such as sterilizable star sensors and batteries. Phase D begins in May 1970 with the start of unit design, development of five models, and operational support equipment. The models, one of which is delivered to the capsule integrator for total structural test, are all completed early in the program so the test results are incorporated into the flight drawings and initial proof test models. After a final drawing release, the flight units plus spares are assembled into complete SMs, tested, and delivered to the capsule integrator. The support equipment is available as the SMs are ready, and are delivered with the SMs. An adequate support level is maintained through launch. This program plan, with minor changes, is applicable to all MSSM designs considered for the Mars 1973 Program.

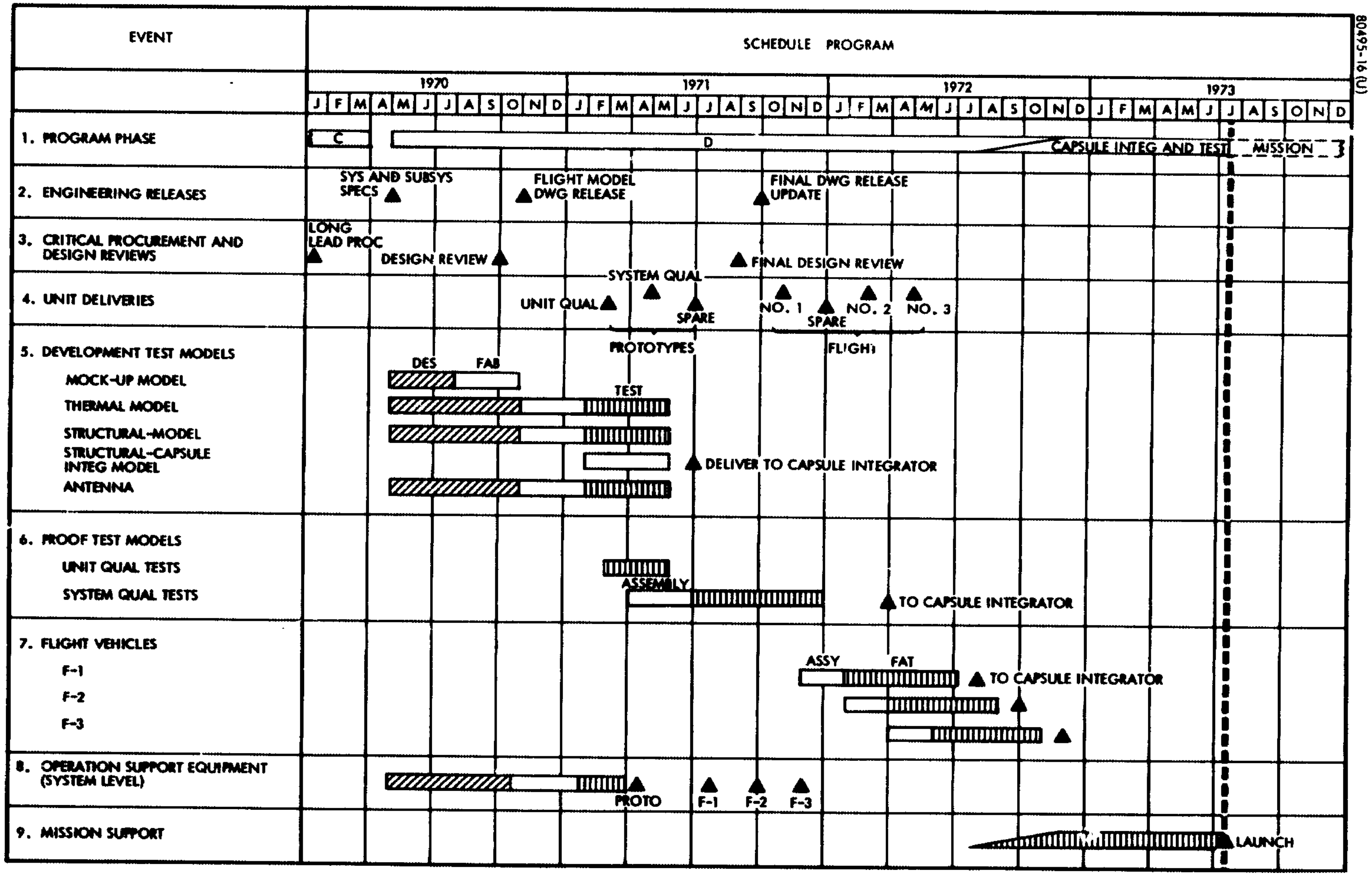

Figure 16. MSSM Program Schedule