NASA CR-66728-1

FINAL SUMMARY REPORT

STUDY OF A SOFT LANDER/SUPPORT MODULE FOR MARS MISSIONS

VOLUME I - SUMMARY

By Raymond S. Wiltshire and Hugh E. Craig

Prepared under Contract No. NAS1-7976 by
MARTIN MARIETTA CORPORATION
Denver, Colorado

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FOREWORD

This Final Summary Report for the Soft Lander/ Support Module study, a supplement to the "Study of Direct Versus Orbital Entry for Mars Missions" (NASA Contract NAS1-7976), is provided in accordance with Part III A.6 of the contract schedule as amended. This Final Summary Report is in three volumes as follows:

NASA CR-66728-1 Volume I, Summary;

NASA CR-66728-2 Volume II, Subsystem Studies;

NASA CR-66728-3 Volume III, Appendixes.

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SUMMARY

This report documents the study results accomplished in Modification 3 to the Direct Versus Orbital Entry for Mars Mission Study. The objective of this study was to determine the conceptual design of an attitude stabilized soft lander capsule and to obtain solutions in the areas of communication, data handling, cost, reliability,* weight, program implementation of long-lead items and the effective use of existing equipment. The Martin Marietta study is believed to be in complete response to the requirements established by Langley Research Center, National Aeronautics and Space Administration. Specifically the study has:

1) Investigated each subsystem to identify any critical long-lead items requiring FY 69 funds to support the Mars 1973 Soft Lander Mission;
2) Established the minimum costs of the autonomous soft lander/support module configuration that meets overall mission guideline constraints and emphasizes the availability for the 1973 launch opportunity;
3) Evaluated the applicability of current spacecraft hardware undergoing development to determine changes necessary to meet the mission requirements;
4) Performed mission analysis to confirm communications link analysis for cruise, entry, and postlanding mission phases;
5) Developed conceptual design of the attitude stabilized soft lander capsule with its separable support module, using the results from current tradeoff studies and analyses, along with those previously generated in the basic Mars Mission Mode Study.

Mission analyses were conducted considering the Titan IIIC launch vehicle performance, launch period selection, targeting capability, landing site variability, communication link geometry, and terminal phase.

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*Reliability studies were performed in a general sense, as intended by the contract statement of work.

Additional subsystem tradeoff studies and analyses were conducted to assess the various support module alternatives, e.g., (1) trailing vs flyby, (2) separation distance from the planet, (3) location of sun and star sensors, and (4) communication interfaces with capsule.

As a result of the above studies, a conceptual design of the soft lander/support module was derived wherein many subsystem components are not required to be sterilized, permitting use of much existing equipment.

The final task accomplished during this study was to determine the overall mission cost based on the preferred design and the study ground rules and constraints.

INTRODUCTION

This summary volume is presented in four major sections -- Mission Analysis, Configuration Studies, Subsystem Studies, and Conclusions.

The first section covers the mission analysis studies, including mission profile, Titan IIIC launch vehicle capability, flight operations, launch date/arrival date considerations, targeting, and communication link geometry. To achieve two-station coverage during entry and landing, a mean touchdown time of 6 hr universal time is selected. Targeting flexibility in terms of landing longitude is achieved by varying the Mars arrival date. The Titan IIIC payload capability resulting from this analysis is shown to provide a substantial margin for the 1973 Mars Mission.

The second section describes the preferred configuration resulting from the study that is an attitude stabilized soft lander using the direct entry mode with a flyby unsterilized support module. It is shown to soft land the 947 lb of useful landed weight; a 2568 lb total spacecraft with an 11-ft-diameter aeroshell is required.

Individual subsystems are discussed in the third section; study results are presented, the preferred design is described with alternatives identified, and the subsystems conclusions listed. The preferred configuration avoids several problems by not requiring the support module, cruise solar array, cruise ACS, and stellar sensors to be sterilized.

The fourth section presents the study conclusions and identifies areas that merit further effort.

The study ground rules and Science instruments are given in tables 1 and 2.

TABLE 1.- GROUND RULES

1)	Two spacecraft to be launched;
2)	Launch vehicle to be Titan IIIC;
3)	The mode of delivery shall be direct entry;
4)	The support module shall provide relay communication and not conduct scientific experiments;
5)	Minimum lander lifetime on the surface shall be 3 days, with a 90-day goal;
6)	Minimum data volume shall be 10^7 bits, with a goal of 10^8 bits;
7)	Entry data must be obtained independent of landing success;
8)	Entry data from the first lander must be available and analyzed before second lander entry is committed;
9)	The present three 85-ft and two 210-ft DSN facility antennas (Goldstone and Australia) will be available;
10)	Launch opportunity to be 30 days' duration;
11)	Minimum launch window to be 2 hr;
12)	Available launch azimuth range to be 45 thru 115°;
13)	The landing site shall be near equatorial at approximately ±20° latitude.

TABLE 2.- SCIENCE INSTRUMENTS

Objective	Instrument	Weight, lb	Data bits
Entry science (18 lb)			
Structure of atmosphere	Pressure	3	2×10^4
	Temperature	1	10^4
	Density	4	10^5
Composition	Mass spectrometer	8	6×10^4
Water vapor	Aluminum oxide hygrometer	2	5×10^3
Surface science (39 lb)			
Imagery	Facsimile	5	10^7
Atmospheric composition/organic compounds	Gas chromatograph-mass spectrometer/pyrolysis	16	10^5
Biology	?	8	10^3
	Soil sampler[a]	2	10^2
Atmospheric water	Aluminum oxide hygrometer	1	10^3
Subsurface water	Aluminum oxide hygrometer with probe	3	10^3
Atmospheric temperature	Platinum resistance thermometer	1	10^3
Atmospheric pressure	Capacitance diaphragm	1	10^3
Wind velocity	Cup anemometer	2	10^3

[a] Soil sampler is required to supply a sample to both gas chromatograph and biological instruments.

SYMBOLS AND ABBREVIATIONS

ACS	attitude control system
AFETR	Air Force Eastern Test Range
AMR-1	high-altitude radar altimeter
AMR-2	low-altitude radar altimeter
au	astronomical units
B_E	entry ballistic coefficient, M/C_DA, slugs/foot^2
bps	bits per second
CST	combined system test
C_3	Earth departure energy, (kilometers/second)^2
DAU	data automation unit
DLA	declination of departure asymptote, degrees
DSIF	Deep Space Instrumentation Facility
DSS	Deep Space Station
ETR	Eastern Test Range
FSK	frequency shift keying
G&C	guidance and control
GS-MS	gas chromatograph-mass spectrometer
h_E	entry altitude, kilometers
h_P	periapsis altitude, kilometers
h_{PD}	parachute deployment altitude, kilometers
h_t	terrain height above mean surface, kilometers
h_v	vernier ignition altitude, kilometers
IMU	inertial measurement unit
ITL	integrate-transfer-launch
LM	Lunar Module
LR	landing radar
M/C	Midcourse
ODP	orbit determination period
PCM	pulse code modulation

PSK	phase shift keying
R&D	research and development
S/C	spacecraft
SMAB	solid motor assembly building
t_c	coast time, hours
t_E	entry time, seconds
T/M	telemetry
t_P	time on parachute, seconds
t_{PL}	post land relay link time, seconds
t_v	vernier time, seconds
TWTA	traveling wave tube amplifier
$T_{M/C}$	time at which first M/C maneuver is made, days
UT	Universal time
VDA	valve drive amplifiers
V_{HE}	**Mars asymptotic velocity, kilometers/second**
VIB	vertical integration building
W	Weight, pound
α	absorptivity
α_c	lander antenna aspect angle, degrees
$\alpha_{S/M}$	support module relay antenna aspect angle, degrees
ΔV	velocity increment, meters/second
ΔV_{EJ}	capsule deflection velocity increment, meters/second
ΔV_{MIL}	midcourse correction velocity increment, meters/second
γ_E	inertial entry flightpath angle, degrees
ϵ	emissivity
σ	standard deviation
τ_{EJ}	deflection angle, degrees
φ_6	landing site longitude corresponding to dual station coverage, degree
ζ_E	angle measured at Mars between $\vec{V}_{HE}$ and $\oplus$ direction, degree

MISSION ANALYSIS

The mission analysis results presented below summarize the analyses performed for the soft lander/support module configuration concept. The overall mission sequence is shown in figure 1. This concept makes use of the Titan IIIC launch vehicle and an integrated spacecraft that both controls the flight to Mars and performs a soft landing. The spacecraft is supported by a spin-stabilized, flyby support module during the encounter and entry phase of the mission, which allows a high bit rate data return via a relay link (lander to support module to Earth). The initial postlanding communication link is also via the support module. Postlanding communication is obtained with a direct Mars surface to Earth link. A 3-day lander lifetime on the surface is assured with 90 days possible under favorable conditions.

The data below are first presented in terms of a typical design mission profile. This is followed by the more detailed parametric data that led to the selected mission profile. The parametric results include updated launch vehicle performance, a flight operations summary to establish timing requirements, launch period selection and targeting flexibility, midcourse correction ΔV requirements, and communication link geometry analysis. Much of the parametric results presented earlier (refs. 1 and 2) are also directly applicable.

1. DESIGN MISSION PROFILE

The design mission profile presented here is intended as an overall summary of the parametric analyses. It reflects constraints imposed by the desire to have a simple soft lander configuration with minimum total program cost. The overall performance capability summarized below has not been optimized. It represents the minimum that can be done with this concept. Areas where trade studies can improve the performance are indicated.

In addition to the study ground rules, several constraints are imposed to assure a meaningful mission. These are as follows:

1) Maximum landing longitude flexibility - 1/2 the planet for the first vehicle, total planet for the two vehicles;

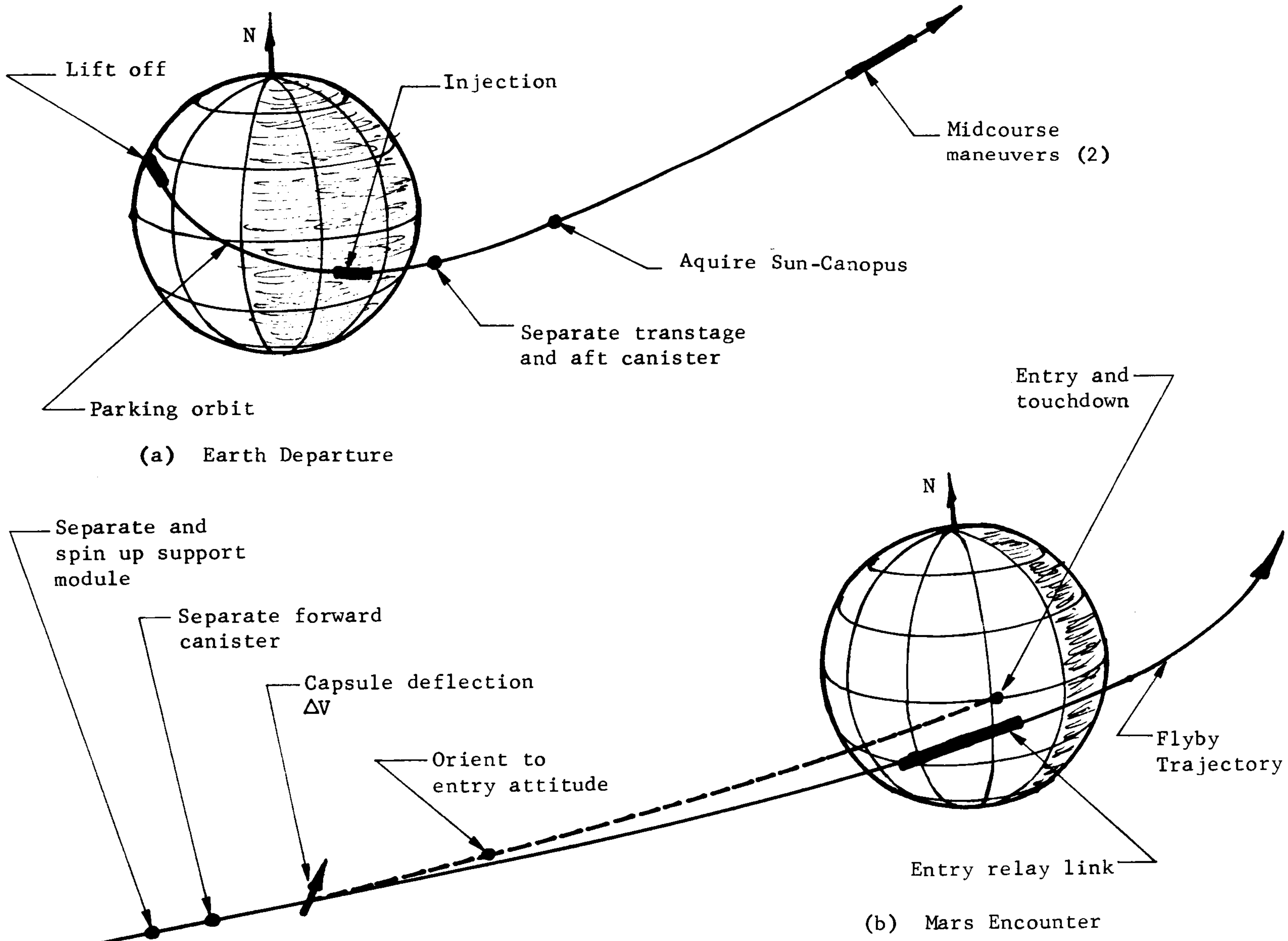


Figure 1.- Soft Lander/Support Module Mission Sequence

2) Maximum communication range at encounter of 1.4 au;

3) Minimum arrival date separation of 5 days between the two vehicles;

4) Minimum post-touchdown communication link time of 10 min.

The poorly defined entry environment suggests that the first lander be targeted to an area that appears smooth, is scientifically interesting, and has a low general terrain height above the mean planet surface. These requirements should be satisfied by the longitude flexibility of item 1) above. The maximum communication range is limited to ensure a data volume of 10^7 bits over the first 3 days on the surface. The 5-day arrival separation prevents overlap between data receipt and command functions and allows quick-look evaluation of data before committing the second vehicle. The 10-min initial link time allows for transmittal of a site survey picture before communication is lost with the support module.

The study ground rules include the requirement for a 30-day launch period. A more detailed launch operations analysis has indicated that two launch vehicles can be launched by a single crew, two-pad operation with a launch period of 20 days or a period of 28 days for two crews and one pad. The 28-day period is sufficiently close to 30 days so that no separate discussion is attempted. The minimum energy 30-day launch period from 13 July 1973 to 12 August 1973 requires an Earth-departure energy, C_3, of 16.5 $(km/sec)^2$. The 20- and 30-day launch period cases will require C_3 of 15.7 and 16.7 $(km/sec)^2$, respectively, when an additional 0.2 $(km/sec)^2$ is included for targeting flexibility. Respective first launch days are 19 July 1973 (20 days) and 16 July 1973 (30 days). The summary performance presented here is based on both the 15.7 and 16.5 $(km/sec)^2$ cases. For all of the above, a minimum 2-hr/day window is assured by limiting the declination of the departure asymptote, DLA, to 35°.

The Titan IIIC capability is summarized in table 3 for these two cases.

The total midcourse correction, as reflected in table 3, is 30 m/sec. This can be applied in at least two corrections. The design periapsis altitude of the support module flyby trajectory is 2500 km. This results in a 10-min minimum postlanding link time with worst-case atmosphere and periapsis altitude dispersion.

TABLE 3.- TITAN IIIC CAPABILITY

		20-day	30-day
C_3, $(km/sec)^2$		15.7	16.5
W_{Inject}, total injected spacecraft weight, lb		2787	2707
$W_{Encounter}$, lb, W_{Inject} minus		2566	2487
	Aft canister and payload adapter	179	179
	Midcourse correction, propellant (ΔV = 30 m/sec)	35	34
	ACS gas	7	7
W_{Entry}, lb,	$W_{Encounter}$ minus	2072	1996
	Support module	126	126
	Forward canister	296	296
	Deflection propellant (ΔV = 75 m/sec)	72	69

Note: 1. Launch azimuth - 114°.
2. 3σ performance.
3. 12.5-ft-diam payload fairing (shroud).

Support module separation and spinup occur about 17 hr before Mars encounter. This is followed, about 90-sec later, by cruise module separation. At 16.5 hr before entry, the lander is given an impulse of up to 75 mps (depending on periapsis altitude dispersion) to deflect it from the flyby trajectory of the support module to the required entry trajectory.

The lander enters the Martian atmosphere at an altitude of 800 000 ft, a design velocity (for terminal phase analysis) of 21 000 fps and a nominal flightpath angle of -21°. The design entry weight of 1860 lb gives an entry ballistic coefficient of 0.375 slug/ft^2. The parachute is deployed by an altitude trigger at 11 500 ft above the terrain (Mach 2 in the minimum atmosphere of ref. 3). The vernier is ignited by an altitude trigger at 5300 ft above the terrain; the parachute has reached a flightpath angle of -70° in the minimum atmosphere at this altitude. The maximum terrain height capability is 4700 ft above the mean surface. Preliminary study results indicate that parachute size may be varied to give a higher terrain capability with a small weight penalty. This is recommended as a future trade study. The design time sequence from entry to touchdown in the minimum and maximum atmosphere is given in table 4. The minimum time from the end of transmission blackout to touchdown allows all entry data to be relayed before touchdown. With the maximum time to touchdown and a 3σ dispersion in periapsis altitude, postlanding link time is 10 min, sufficient for transmittal of a site survey picture to the support module for relay to Earth.

TABLE 4.- TIME SEQUENCE, ENTRY TO TOUCHDOWN

	Minimum atmosphere, $\gamma_E = -24°$	Maximum atmosphere, $\gamma_E = -18°$
Entry (800 000 ft)	0	0
End of blackout (10 000 fps)	105	94
Chute deployment	139	336
Vernier ignition	165	386
Touchdown		
From entry	208	436
From end of blackout	103	342

Note: 1. Entry velocity, V_E = 21 000 fps.

2. Entry ballistic coefficient, B_E = 0.375 slug/ft^2.

For the first 3 days of life, the lander has a direct link to Earth, capable of 353 bps. This is adequate to return the required 10^7 bits of data. Operation on batteries and solar cells will allow 90 days of weather-station data gathering and transmission capability; favorable solar radiation for the solar cells will allow more data-gathering and transmission time.

2. TITAN IIIC CAPABILITY

Previous estimates of the Titan IIIC payload capability for the 1973 Mars mission have been based on generalized payload/velocity data. As a result of increased interest in the Titan IIIC as the launch vehicle for the 1973 mission, a more detailed analysis of its capability has been made. This analysis is based on digital computer runs and includes the following:

1) A more detailed estimate of payload fairing weight applicable to the soft lander/support module concept;
2) Boost trajectory shaping within airload and aeroheating constraints;
3) Jettison of the design payload fairing weight at the correct time in the trajectory;
4) Launch azimuth of 114°;
5) Optimization of the propellant loading of upper stages and Earth park orbit eccentricity to meet African impact (Stage II) range safety constraints;
6) Determination of gravity losses encountered leaving park orbit;
7) An updated estimate of required launch vehicle margin.

This discussion will include a brief description of the analysis and results of the Titan IIIC payload study. More detail may be found in reference 4.

To accommodate the soft lander with its 11-ft-diameter aeroshell, a 12-ft-diameter payload fairing (shroud) is required. A sketch of a fairing with an outside diameter of 12.5 ft, used for this analysis, is shown in figure 2. The fairing weight is 2300 lb. Because the fairing is jettisoned at 280 sec after liftoff (during Stage II flight), the effect on payload capability is in the order of 7 to 8% of the fairing weight. The sensitivity of payload weight to fairing weight is shown in figure 3 as a function of C_3. The sensitivity of payload weight to fairing diameter is approximately -25 lb/ft.

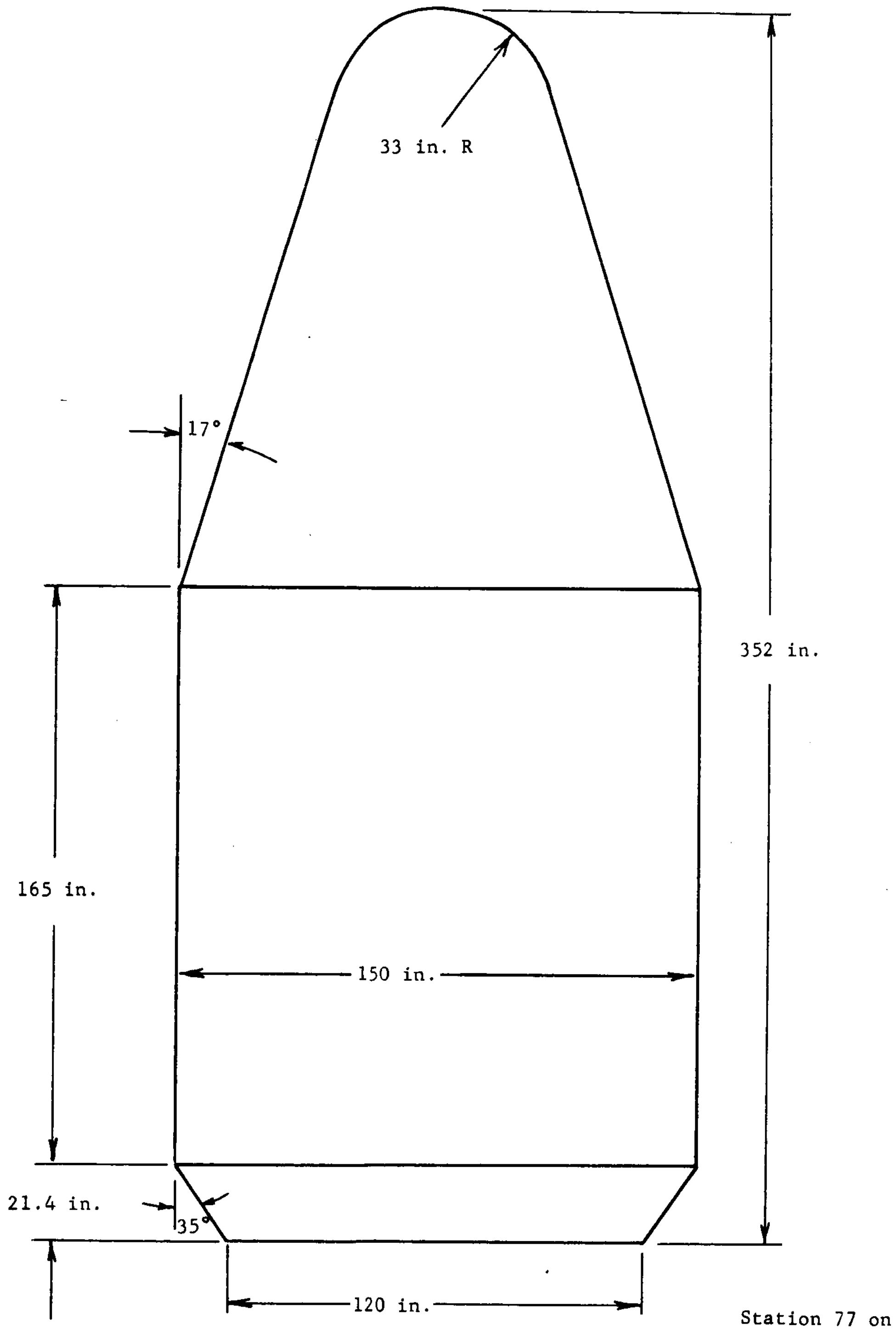


Figure 2.- Payload Fairing

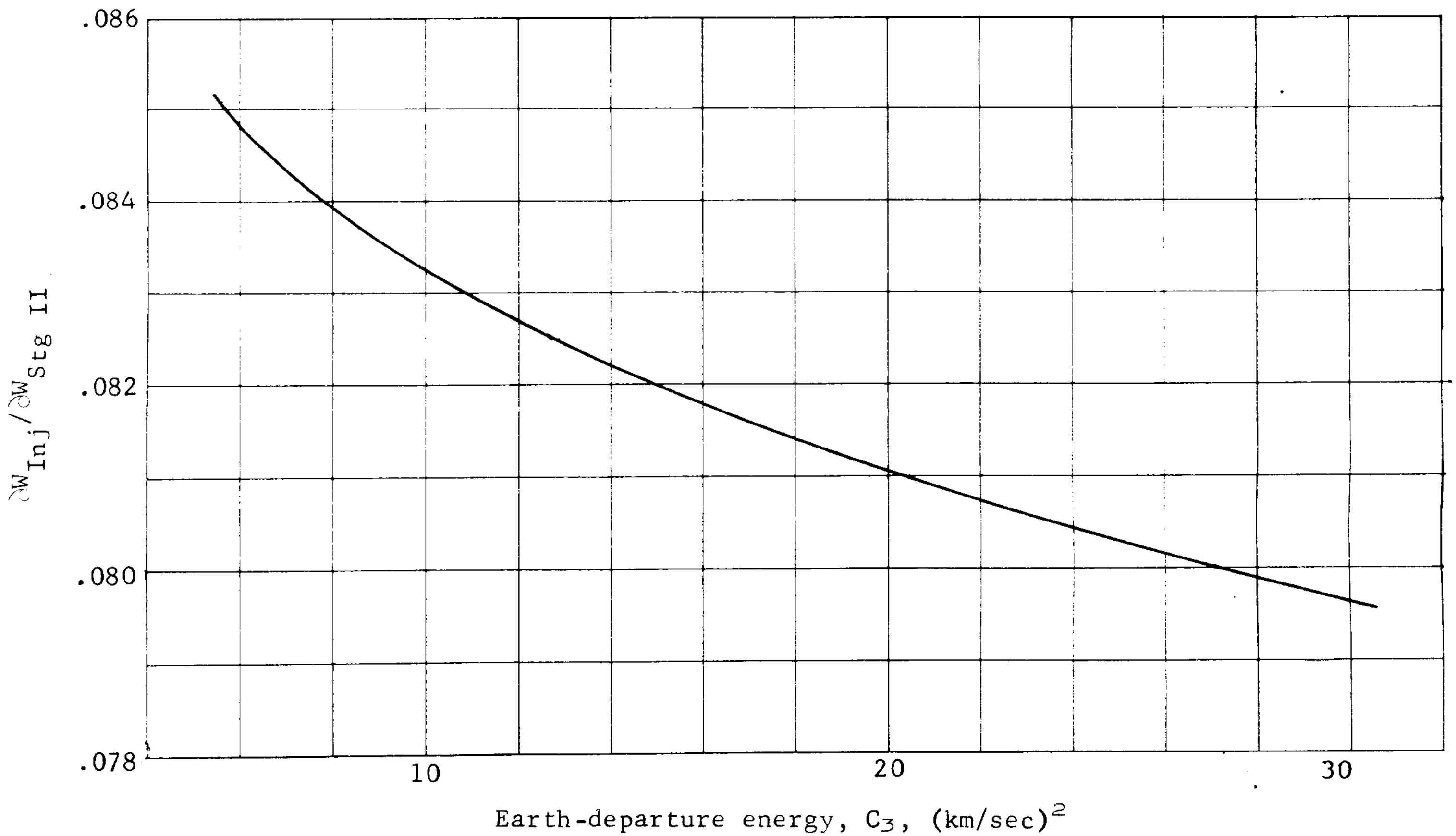


Figure 3.- Sensitivity to Weight Jettisoned during Stage II Flight

To avoid Stage II impact on Africa, a "nearside/farside" analysis is made. That is, upper stage (Stage II and Transtage) propellant loading is adjusted so that a maximum performing vehicle will not impact east of the near-African coast or a minimum performing vehicle will not impact west of the far-African coast. This analysis is combined with a study of the resulting park orbit eccentricity and the required orbit coast time to maximize payload. The result of this analysis indicates that a Stage II farside impact and a 90x230-n-mi park orbit are optimum in the C_3 range of 15 to 17 km^2/sec^2. An average coast time of 30 min or 120° is used. Payload sensitivity to coast angle is shown in figure 4.

Finally, a preliminary study of trajectory shaping upon leaving park orbit is undertaken to minimize injection (into heliocentric transfer orbit) gravity losses. Although considerably more work must be done here, the preliminary injection gravity loss is 70 fps. (This is 20 fps greater than the original estimate of 50 fps used in earlier studies.)

The Titan IIIC payload capability resulting from this analysis is shown in figure 5 as a function of C_3. Three curves are shown corresponding to various times in the transfer trajectory as discussed earlier. The injected weight is the total Titan IIIC capability less fairing penalty. The encounter weight is injected weight less payload adapter, aft sterilization canister lid, ACS gas, and midcourse correction propellant $\left(\Delta V_{M/C} = 30 \text{ m/sec}\right)$. Entry weight is encounter weight less support module, forward canister and associated equipment, and deflection propellant $\left(\Delta V_{EJ} = 75 \text{ m/sec}\right)$. Figure 5 indicates C_3 values of 15.7 and 16.7 km^2/sec^2 corresponding to 20- and 30-day launch periods, respectively, as discussed in the targeting analysis. In addition, the maximum allowable C_3 of 17.4 km^2/sec^2 for the current design weight is shown. A sequential weight summary is shown in table 5 along with the design weights and margins. The associated values of C_3 shown in the table are discussed in the targeting analysis.

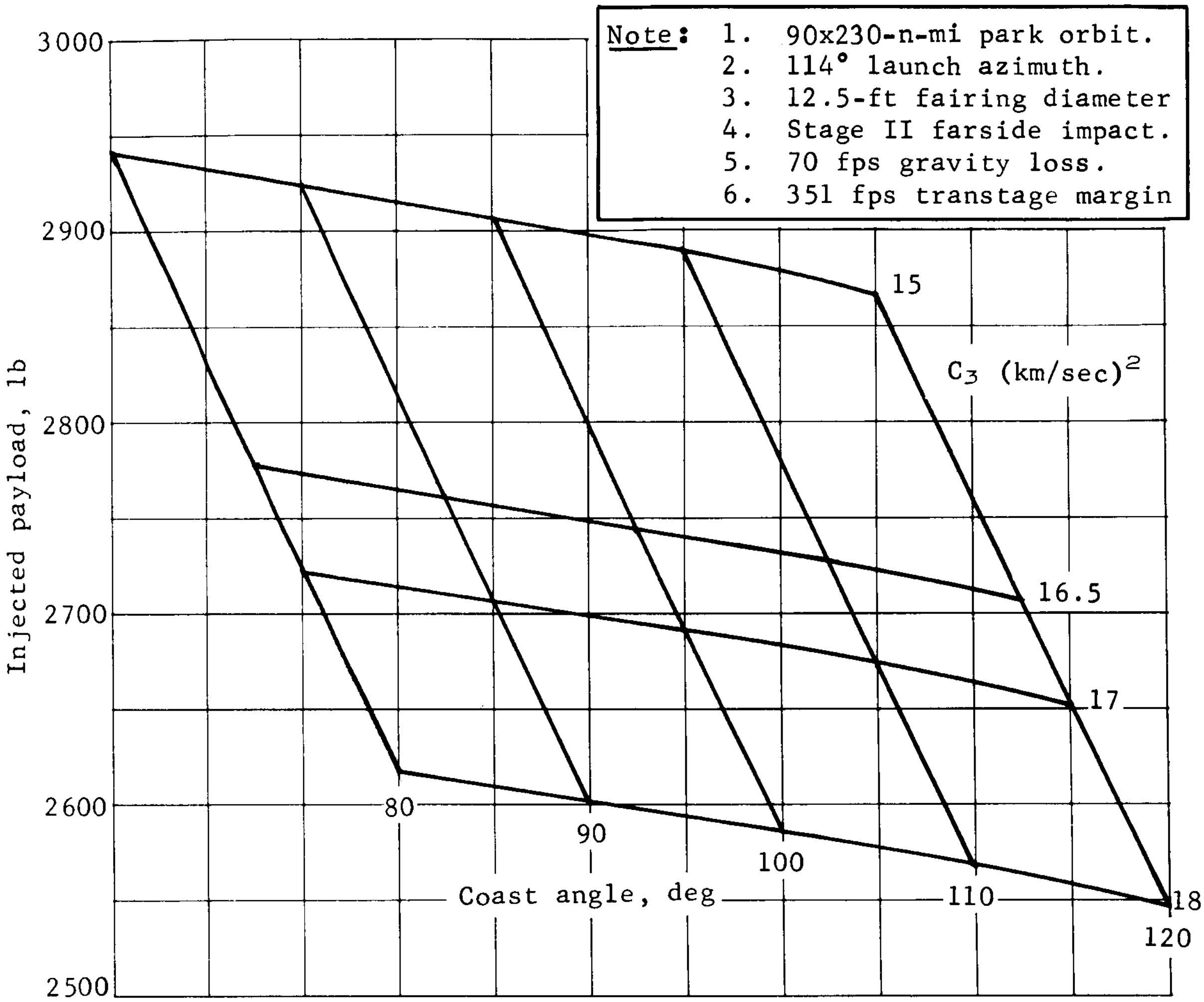


Figure 4.- Titan IIIC Sensitivity to Park Orbit Coast Angle

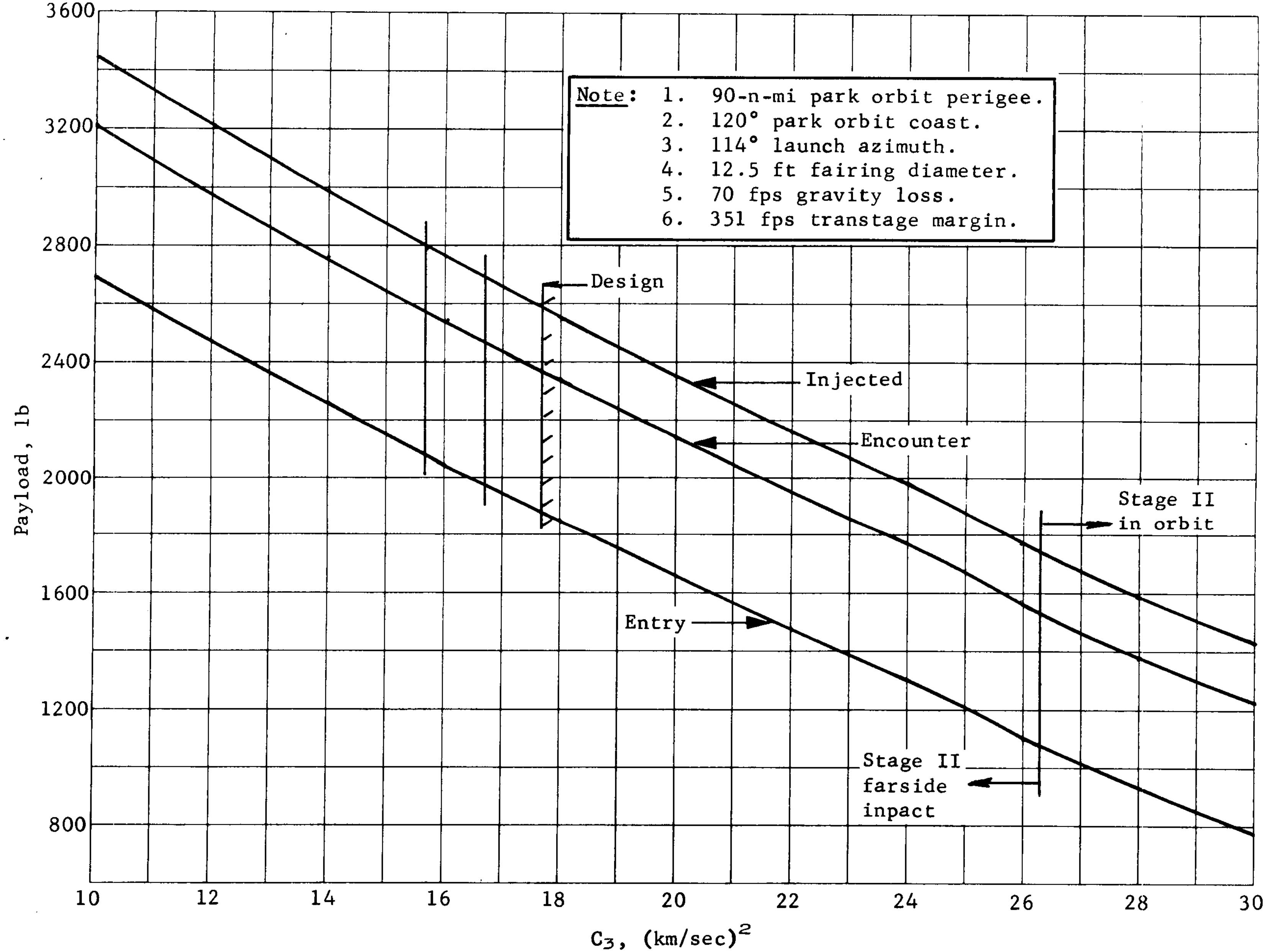


Figure 5.- Titan IIIC Capability

TABLE 5.- SEQUENTIAL WEIGHT SUMMARY

	Titan IIIC performance			Design	+ Margin		
C_3, (km/sec)^2	15.7	16.5	16.7		15.7	16.5	16.7
W_{Inj}, lb	2787	2707	2687	2568	219	139	119
$\Delta W_{Can,\ Ad}$, lb	179	179	179				
$\Delta W_{M/C}$ [a]	35	34	34				
ΔW_{ACS}, lb	7	7	7				
$W_{Encounter}$, lb	2566	2487	2467	2350	216	137	117
$\Delta W_{S/M}$, lb	126	126	126				
ΔW_{Can}, lb	296	296	296				
ΔW_{Def} [b]	72	69	-69				
W_{Entry}, lb	2072	1996	1976	1859	213	137	117

[a] $\Delta V_{M/C}$ = 30 mps

[b] ΔV_{EJ} = 75 mps

W_{Inj} = total injected spacecraft weight.

$\Delta W_{Can,\ Ad}$ = combined weight of aft sterilization canister and payload adapter.

$\Delta W_{M/C}$ = midcourse correction propellant weight.

ΔW_{ACS} = ACS gas weight.

$W_{Encounter} = W_{Inj} = \Delta W_{Can,\ Ad} - \Delta W_{M/C} - \Delta W_{ACS}$.

$\Delta W_{S/M}$ = support module weight.

ΔW_{Can} = forward sterilization canister with mounted equipment.

ΔW_{Def} = deflection propellant weight

$W_{Entry} = W_{Encounter} - \Delta W_{S/M} - \Delta W_{Can} - \Delta W_{Def}$.

It should be pointed out that the payload capability presented herein is based on a baseline follow-on vehicle with 3σ minimum performance level. As shown in table 5, this results in a minimum of more than 100 lb of margin over the soft lander design weight. An additional 100 lb may be added by going to a 20-day launch period. The use of a 3σ performance probability gives a payload capability 275 lb less than the use of nominal performance. Changes that have been incorporated for Vehicle 17 and up, and actual weighing of that vehicle indicate a payload capability some 200 lb greater than the baseline quoted herein. There are other sources of additional payload capability (margin) that should be considered. Reduction of Stage I ullage volume will provide a payload increase of about 160 lb. Further, the 114° launch azimuth is required for a daily launch window of 2 hr only when the declination of the departure asymptote (DLA) is 35°. The targeting analysis indicates that a lower declination (arrival date/longitude dependent) may be used throughout most of the launch period. A 2-hr maximum window (or less, considering the Titan IIIC launch-on-time record) would then allow a more easterly maximum launch azimuth. For a launch azimuth of 100°, the accompanying increase in payload would be about 100 lb. These and other ground rule-type changes result in additional 360 lb payload capability.

It is apparent then that a performance margin of two or three times the value shown here may be achieved on a no cost basis by ground rule changes. The further possibility exists that one or more suggested design changes may be approved by the 1973 time period. In view of all of this, it is evident that a substantial margin may be assumed for the Titan IIIC for the 1973 Mars mission.

3. FLIGHT OPERATIONS SUMMARY: TIMING REQUIREMENTS

The purpose of this section is to summarize the gross timing analyses associated with overall flight operations. The discussion covers the mission profile from launch to post-touchdown; the reasons for selecting overriding timing requirements are discussed in detail.

Launch

Launch timing is a function of launch site activities, launch vehicle holds, payload holds, and weather considerations. The result of the launch timing analysis leads directly to launch period determination for the two vehicles to be launched for the

1973 Mars mission. A preliminary analysis indicates that a maximum launch delay of six days may be encountered for each launch vehicle. This value results from a root-sum-square of the values tabulated below.

Replace spacecraft with spare	3 days
Correct launch vehicle malfunction	3 days
Weather	5 days
rss	6 days

The payload replacement requirement resulting from detection of a malfunction during checkout is self-explanatory. The delay due to launch vehicle malfunction detected during checkout or countdown reflects the longest such delay encountered during Titan IIIC launch experience to date. The 5-day weather delay for high winds and/or other disturbances is an estimate based on Titan IIIC launch experience. Launch experience would indicate 1 or 2 days is an adequate weather hold estimate. However, the estimate is made longer because of the possibility of tropical storm or hurricane activity because the launch period approaches the hurricane season.

The delay between launch of the first vehicle and launch of the second is a function of launch pad operation and required prelaunch activities. Using a single launch pad with multiple crews (2 shifts + overtime) a 15-day turnaround (launch to launch) may be achieved. This includes 3 days for pad refurbishment and 12 days for pad operations. Pad operations include launch vehicle erection, checkout, combined systems test (CST), propellant loading, countdown, and launch. An additional day is allowed for payload installation, bringing the total delay to 16 days. The resulting maximum launch period including 6 days delay for each vehicle plus 16 days between launches is 28 days as shown in figure 6. For application to other analyses, this is extended to 30 days to agree with the study ground rule specifying a 30-day launch period.

Shorter launch periods may be achieved by using both of the Titan launch pads at the AFETR. A shorter launch period increases the launch vehicle payload capability by decreasing the value of C_3 required, as discussed above. A greater degree of targeting flexibility may also be achieved with a shorter launch period as discussed below. Assuming two launch pads with a single crew

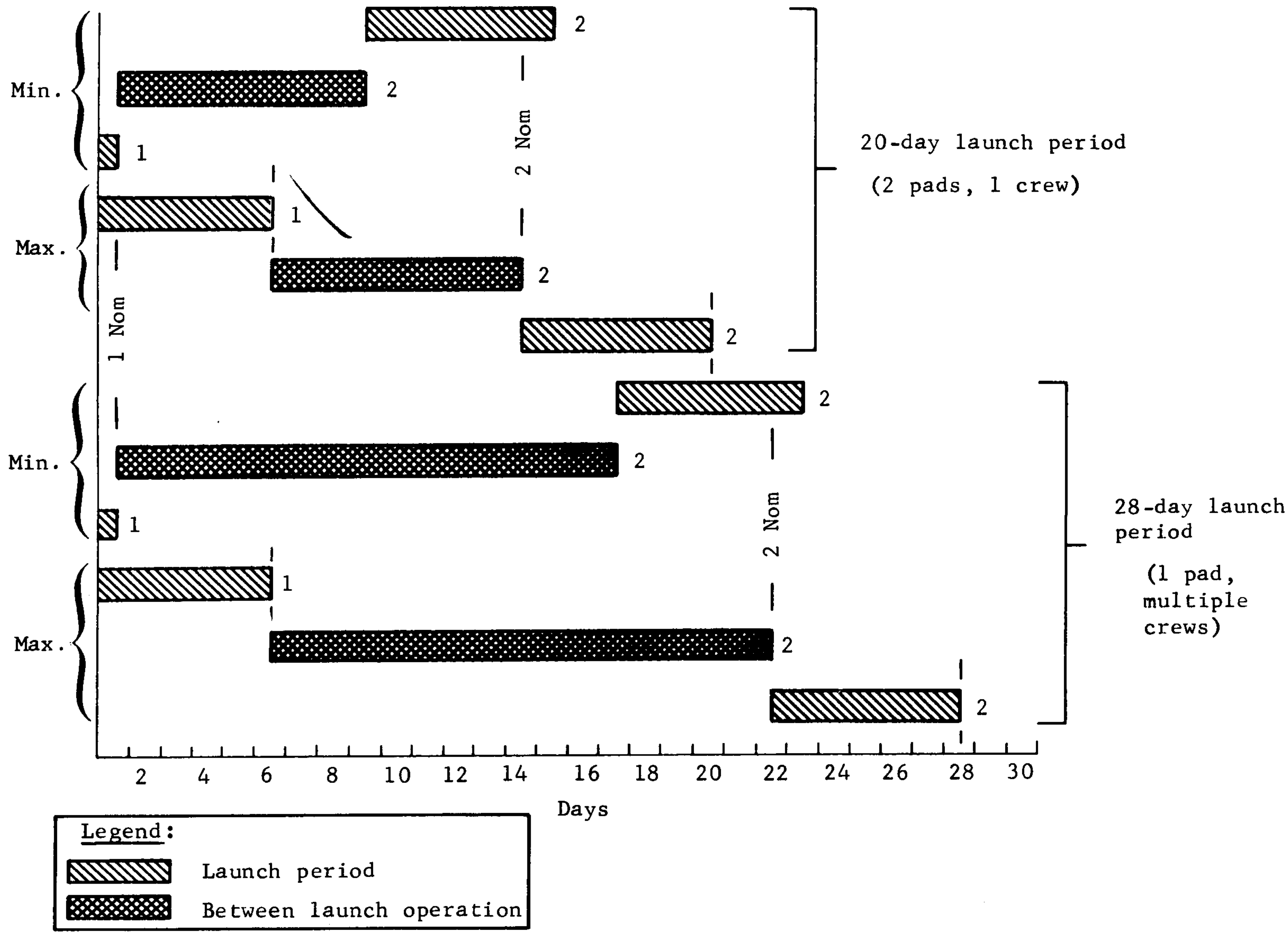


Figure 6.- Launch Periods

(8 hr + overtime), the delay between launches is reduced to 8 days. Table 6 describes the overall sequence in detail, using the Titan IIIC ITL facility at ETR. The following definitions apply to this discussion:

ITL - integrate-transfer launch;

VIB - Vertical Integration Building;

SMAB - Solid Motor Assembly Building;

CST - combined system test.

TABLE 6.- TITAN IIIC LAUNCH SEQUENCE, 2 PADS, 1 CREW

Time	Launch vehicle 1	Launch vehicle 2
9 weeks	VIB operations; SMAB operations; pad operations thru CST	
9 weeks		VIB operations; SMAB operations; Pad operations; Launch (first S/C)
1 day	Install spacecraft	
7 days	Pad operations, CST launch (second S/C)	

The resulting overall launch period is 20 days as shown in figure 6.

It is interesting to note that once a decision has been made to use two launch pads, the delay between two launches may be reduced even further. With two pads and multiple crews, a delay of 2 hr, launch to launch, may be achieved. This assumes range and mission support is available. This provides great flexibility in mission planning for the relatively low cost of another crew.

Planet Approach

The planet approach flight operations are based on the following assumptions:

1) Primary command operations be over Goldstone;
2) Entry phase and initial postlanding link will have dual-station coverage (Goldstone and Booroomba, Australia);
3) Two station coverage (two-way, two station noncoherent mode) tracking data before the last orbit determination program (ODP) run before calculation of the deflection maneuver.

The last two-way tracking point occurs about 46 hr before entry as shown in figure 7. The ODP run is completed 4 hr later. A complete trajectory run is obtained within the next hour. A 4-hr allowance is included for computer downtime or physical plant failure. This is the first of three time contingencies that are included in the approach operational sequence. The deflection maneuver calculations are completed 4 hr later. Three hours are allotted for the management decision process to transmit the calculated deflection maneuver. The command is formatted and ready to transmit to Goldstone 1 hr later. An 8-hr contingency for deep space station (DSS) or communication link failure is included. The command is received by the spacecraft at least 22 hr before lander entry. The command is transmitted back for verification and is received by the ground 10 min later. If the command is not verified, a 4-hr time period exists for retransmitting and verifying twice. The backup commands must be sent over Booroomba. The enable command is received by the spacecraft 10 min after command verification. The spacecraft is then oriented toward Earth for support module separation about 1 hr after the enable command is received by the spacecraft. The lander is then reoriented for the deflection maneuver and the deflection ΔV imparted 16.5 hr before lander entry. The lander is reoriented to a nominal zero angle of attack at entry. Lander entry occurs approximately 20 min before closest approach of the support module. Relay communications and the entry science are initiated about 1 hr before lander entry.

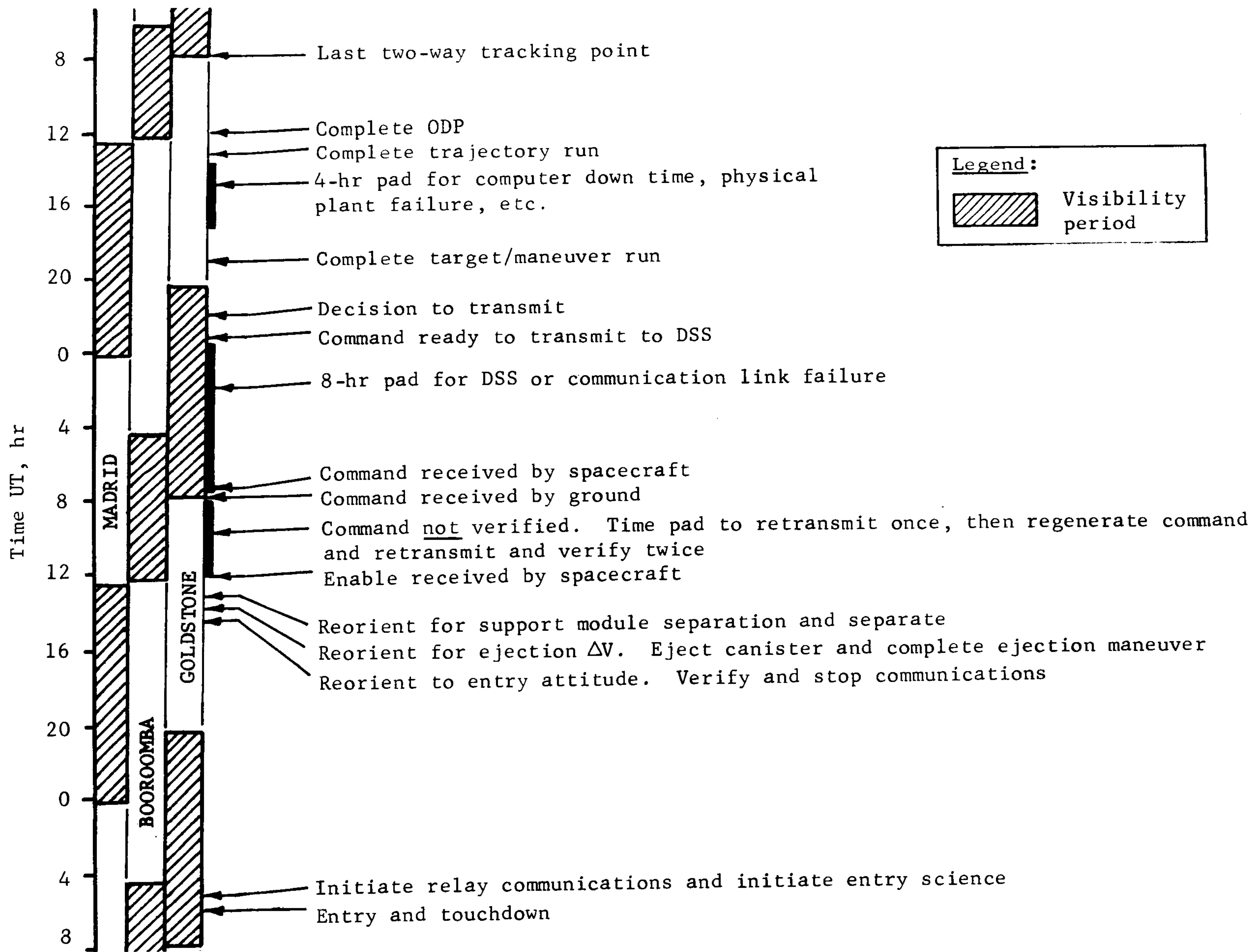


Figure 7.- Approach Operational Sequence and DSIF Visibility

Initial Post Landing Relay Link

The primary requirement during the initial postlanding link is to survey the landing site to determine the landing orientation as well as satisfactory locations for subsurface soil probe deployment. The survey is a picture by the facsimile camera with 90° (azimuth) by 25° (elevation) field of view with 0.1° resolution. The picture requires 1.4×10^6 data bits. With a data rate of 2.4×10^3 bps. this requires a transmission time of 9.7 min. The picture can be initiated approximately 30 sec after touchdown. The relay communication link geometry described later ensures that the landing site survey is accomplished under worst-case conditions of atmosphere, ground elevation mask, and periapsis altitude dispersion. If more postland time is available, through a more complex antenna design on the support module, or higher nominal periapsis altitude, the following two pictures are desirable. The first is a panoramic scan up to the horizon, a 360x10° field of view, with a 0.1° resolution. This requires 2.16×10^6 data bits or a transmission time of 15 min. The second picture is a detail within the landing site survey, 4x3.24° with a 0.01° resolution. This requires 7.77×10^5 data bits or a transmission time of 5.4 min.

Postlanding Direct Link for First Three Days

The postlanding operations for the first lander and the beginning of approach operations for the second lander are summarized in figure 8. The postlanding operations for the second lander are identical to those of the first. The first direct link to Earth occurs about 16 hr after touchdown. The length of the link is a function of antenna design, landing site latitude, and encounter date as described later. The required length of the link is at least 3.1 hr in order to obtain 10^7 bits in 3 days with an antenna design half power beamwidth of 46°. The data are received by the Goldstone station at approximately 0 hr UT. Note that dual-station coverage with Madrid is possible if a 210-ft dish is available at this station. All postlanding data must be received by a 210-ft dish. To simplify operations on the ground, the second spacecraft does not start command operations until after the third direct link of the first lander. The earliest time for the start of the second spacecraft approach operations (last two-way tracking point) is about 74 hr after touchdown of the first lander. The time from the start of approach operations to touchdown is about 46 hr. The shortest time interval between touchdown of the first and second lander is, thus, 5 days.

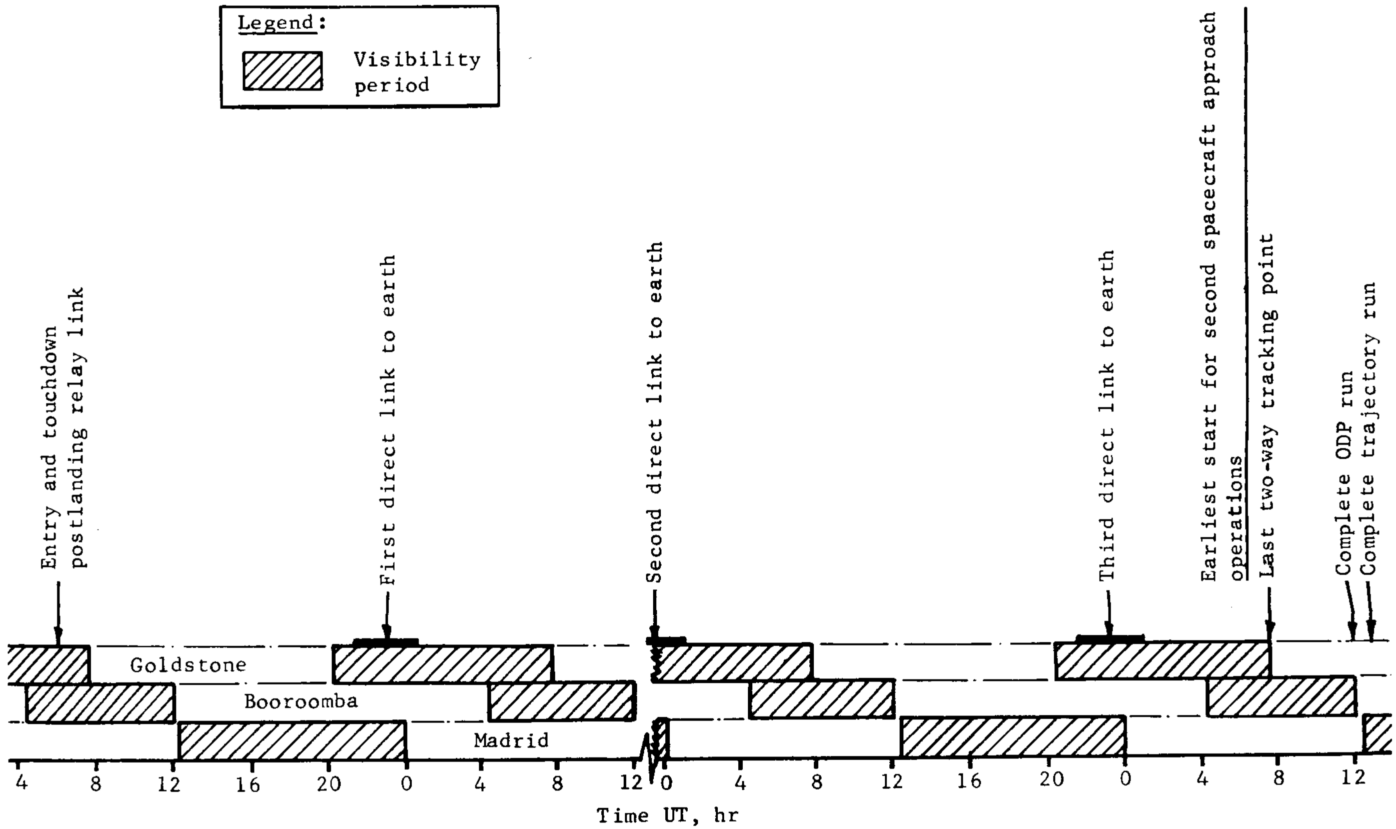


Figure 8.- Lander 1 Postland Operations; Begin Lander 2 Operations

4. TARGETING AND LAUNCH PERIOD SELECTION

A major objective of the Mars '73 mission is to land at a scientifically interesting location. Landing site latitudes between $\pm 20°$ are dictated in the statement of work. The landing site longitude has not been selected and probably will not be until after the Mariner '69 mission, or possibly the Mariner '71. In a mission planning sense as much landing site longitude flexibility as possible should be available consistent with other mission constraints.

The landing site longitude is a function of touchdown time of day, universal time, and encounter date, and is shown in figure 9. It is also dependent on the orientation of the hyperbolic excess velocity vector at Mars, $\vec{V}_{HE}$, and the entry flightpath angle, γ_E. The data shown correspond to a γ_E of -21°. Because the orientation in inertial space of the approach hyperbolic excess velocity vector at Mars does not vary much over the launch period, the landing site is also relatively fixed in inertial space. The slope of the constant longitude lines is due to the difference in the rotation rates of Earth and Mars about their respective axes. In 24 hr, Mars rotates 350.89° or 9.11° less than Earth. Therefore, if touchdown occurs at the same time of day, but one day later, the longitude of the landing site increases by 9.11°. On a given encounter date, the longitude of the landing site increases by 14.62° per hour increase in the touchdown time of day. This is equivalent to the rotation rate of Mars.

Superimposed on figure 9 are the coverage times of the three DSIF stations, Goldstone, Madrid, and Booroomba, Australia. The coverages assume a 15° elevation mask for all the stations. The ground rules for this study state that 210-ft dishes are to be assumed for Goldstone and Booroomba. An 85-ft dish is assumed at Madrid. The real-time playback of entry data and postlanding data for the first 3 days must occur over Goldstone or Booroomba. It is highly desirable to have dual-station coverage for the entry data and initial postlanding data (entry data are not stored). The time of day that corresponds to dual-coverage for Goldstone and Booroomba is about 6 hr UT $\pm$ 1.6 hr. The landing site longitude corresponding to a touchdown time of 6 hr, φ_6, is then directly a function of encounter date. On any given encounter date, the possible longitude variation about φ_6 that still ensures dual-station coverage for the entry data and initial postlanding link data is about $\pm 20°$. If a third 210-ft dish at Madrid were available, the touchdown time could be about 22 hr UT $\pm$ 2.0 hr with dual-coverage over Goldstone and Madrid.

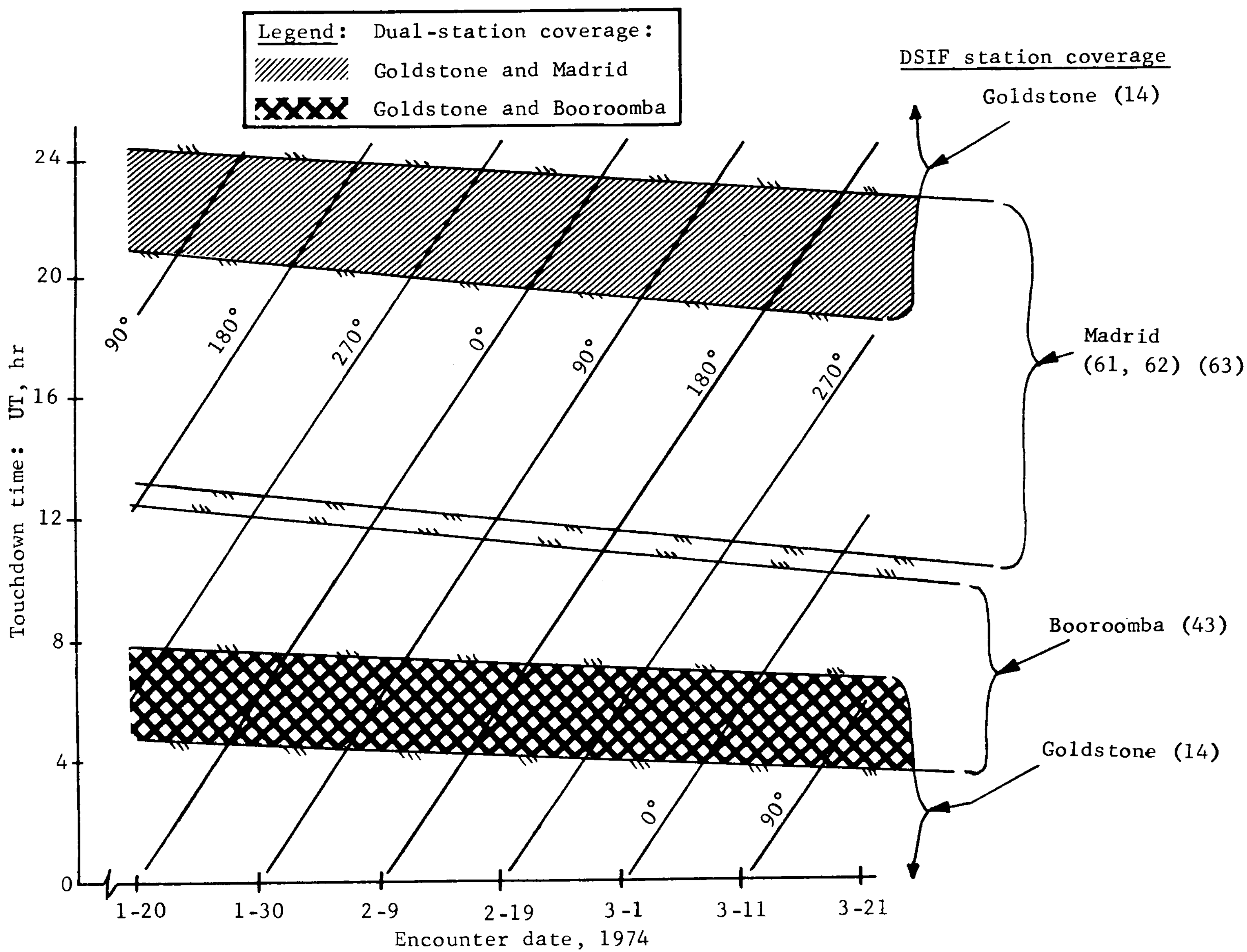
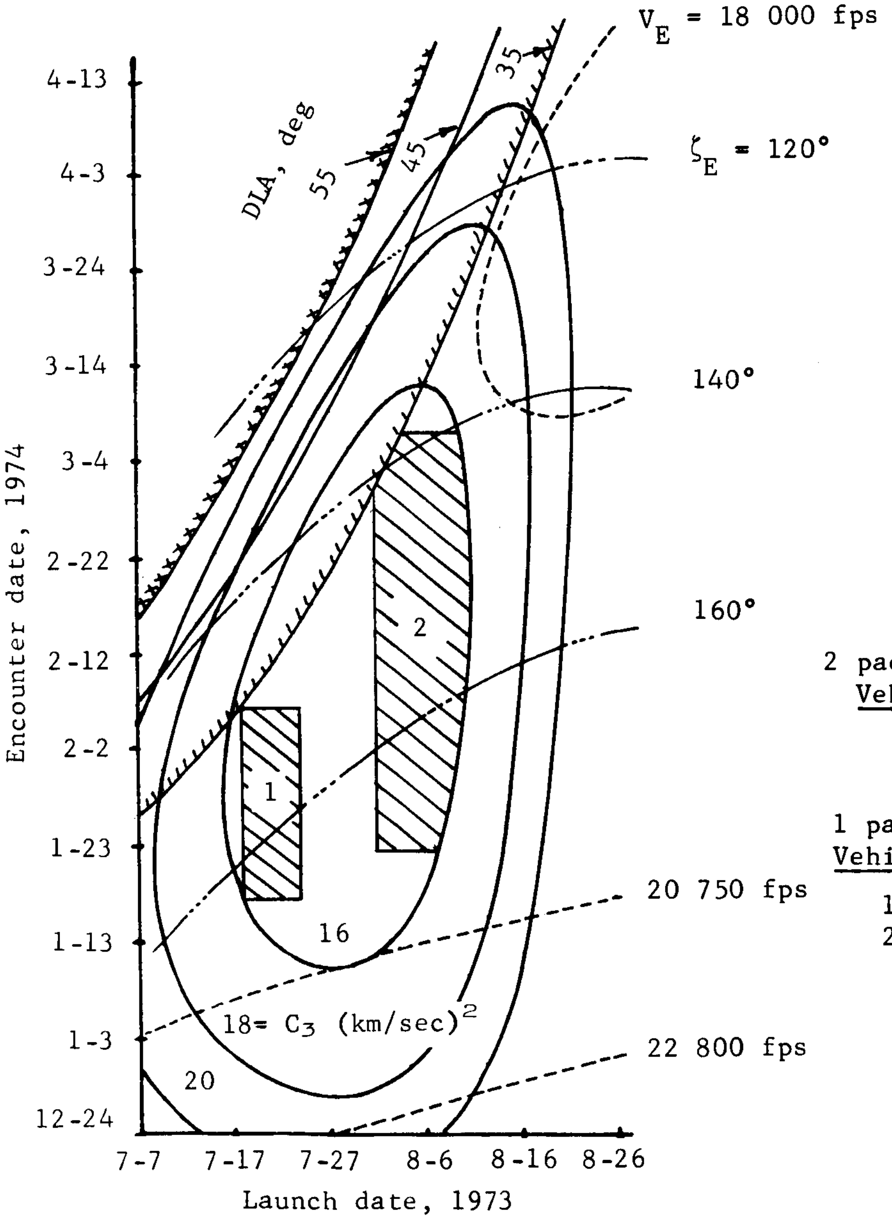


Figure 9.- Landing Site Longitude Selection

The length of the launch period and the amount of longitude flexibility desired for each launch vehicle determine the launch dates and encounter dates available for mission planning. The launch dates/encounter dates must also be selected so that the launch energy required is within the Titan IIIC capability with margin. A launch period of 30 days is ground ruled in the statement of work. This is the period in which both vehicles must be launched. A discussion of a 20- and 28-day launch period based on Titan IIIC experience and the number of launch pads available is given in the Flight Operations Summary. Another constraint that must be considered is the need for at least a 2-hr daily launch window. This requires the DLA to be less than 35° to keep the launch azimuth less than the range safety limitation of 114°.

An example of how the available launch dates/encounter dates are determined is shown in figure 10 for a launch energy, C_3, of 16 km^2/sec^2 and a launch period of 20 days. The width of the launch period for vehicle 1 is 6 days. The height (range of encounter dates) for vehicle 1 is selected to allow coverage of half the planet, i.e., 180° in longitude. This selection is arbitrary but reasonable because a scientifically interesting landing site can surely be found in this range of longitudes. The allowable range of encounter dates for mission planning (longitude selection) is from Jan. 17 to Feb. 6, 1974. This corresponds to longitudes between about -110 and 70°. Once a longitude is selected for vehicle 1, it is held constant for the 6-day launch period. Vehicle 2 can be launched 8 days after vehicle 1. From a mission planning viewpoint, therefore, the first day of the launch period for vehicle 2 is day 14 of the launch period. During actual mission operations, however, if vehicle 1 is launched on day 1, vehicle 2 can be launched on day 9. The earliest allowable encounter date for vehicle 2 is about 5 days after vehicle 1 encounter. Thus, command operations for vehicle 2 begin after 3 days of planetary life for vehicle 1. The width of the launch period for vehicle 2 is required to be at least 6 days. The allowable range of encounter dates depends on the longitude selected for vehicle 1, and range between Jan. 22 and Mar. 7, 1974. The actual launch period available for vehicle 2 is as long as 10 days for some of the encounter dates and possibly as long as 18 days if vehicle 1 goes off on day 1. The latest encounter date is constrained to be Mar. 1, 1974, to keep the distance at arrival less than 1.40 au (for communication purposes). If the longitude for vehicle 1 is selected to be -110°, the earliest allowable encounter date, the allowable longitudes for vehicle 2 are -45 to 270°, a 315° variation. If the



2 pads, 1 crew

Vehicle	Launch days
1	1-6
2	14-20

1 pad, Multiple crews

Vehicle	Launch days
1	1-6
2	22-28

Figure 10.- Mars 1973 (Type I) Launch Period

longitude for vehicle 2 is selected to be 70°, latest allowable encounter date, the allowable longitudes for vehicle 2 are 135 to 270°, a 135° variation. As with vehicle 1, once a longitude is selected for vehicle 2 it remains constant over its launch period. It is obvious that a 28-day launch period cannot be obtained with a C_3 of 16 km^2/sec^2.

The longitude of vehicle 2 can be changed somewhat based on the data returned from vehicle 1 concerning the Martian environment. If the atmosphere is found to be much more dense than the assumed minimum atmosphere, steeper entry flightpath angles are allowable, which change the preselected landing site longitude. The entry flightpath angle is determined by the lander deflection maneuver, 16.5 hr before entry, which is calculated on the ground and sent to the lander by Goldstone.

The allowable longitude selection for vehicles 1 and 2 is summarized as a function of C_3 in figure 11. The longitude variation for vehicle 1 is, by definition, 180° independent of C_3. The minimum allowable C_3 for a 20-day launch period is 15.5 km^2/sec^2. This allows only one specific longitude for vehicle 2, about 135°. The C_3 must be increased about 0.2 km^2/sec^2 above the minimum, to 15.7 km^2/sec^2, to obtain reasonable longitude flexibility with vehicle 2. Similarly, for a 30-day launch period, the minimum C_3 is 16.5 km^2/sec^2 so that to get longitude flexibility with vehicle 2, a C_3 of 16.7 km^2/sec^2 is required. The flexibility shown in figure 11 for vehicle 2 corresponds to the earliest possible encounter date for vehicle 1.

The choice of landing site latitude is dictated by the post-landing direct link and is discussed below.

5. MIDCOURSE ANALYSIS

The sizing of the midcourse correction requirement is an important design consideration from a weight standpoint as well as ACS design. The size of the first midcourse correction depends primarily on the errors in position and velocity after injection onto the trans-Mars trajectory. These errors depend on the launch vehicle and its guidance system, the parking orbit coast time, and the injection velocity required. This analysis assumes a Titan IIIC launch vehicle, a 30-min coast time and an injection velocity of 38 500 fps (C_3 = 16.5 km^2/sec^2). Another consideration in the size of the first midcourse correction is the time at which it occurs.

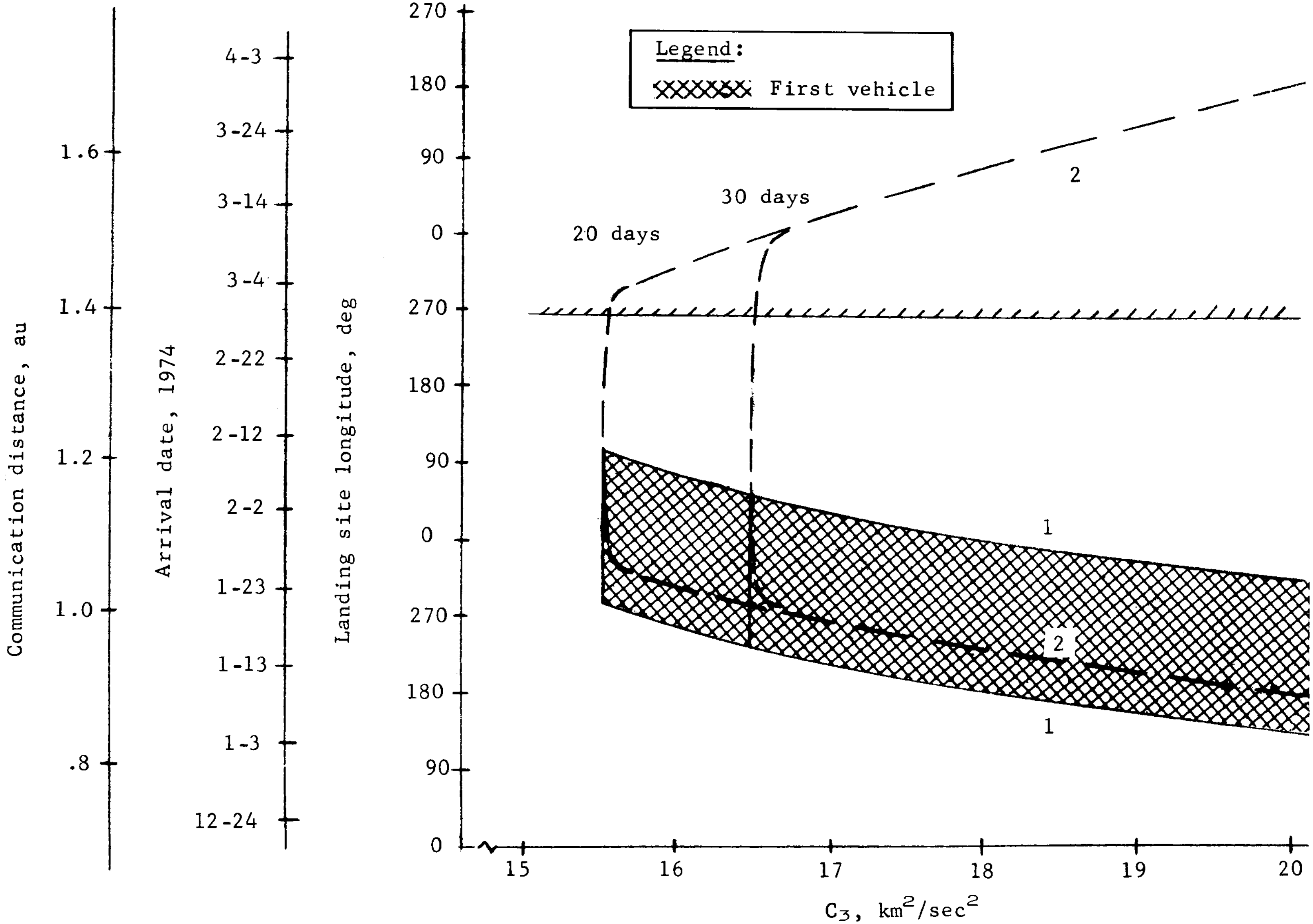


Figure 11.- Launch Energy Requirements

The results are based on analytic two-body sensitivities but check well with n-body results (within 10%). The error at injection, injection covariance matrix Q_{INJ}, is propagated to the Earth's sphere of influence, Q_{SPH}. The covariance matrix is then converted to a heliocentric reference frame, propagated to Mars, and converted to position errors in the impact parameter plane $(\hat{B} \cdot \hat{T}, \hat{B} \cdot \hat{R})$ and normal to it $(\hat{S})$. The position errors at Mars are then mapped back to the time at which the maneuver is made to obtain the covariance matrix of the midcourse correction, $CV_{M/C}$, required to null the error at Mars. The time of flight error is nulled with the S component which is in the direction of the hyperbolic excess velocity vector at Mars. The required fuel loading for the first midcourse correction is based on the mean value, μ_R, and standard derivation, σ_R, of a Rayleigh distribution that, in turn, is based on the rms value of $CV_{M/C}$ as explained in reference 5. The fuel loading is $\mu_R + 3\sigma_R$.

The normalized fuel loading required if the first midcourse maneuver is made at the Earth's sphere of influence is shown in figure 12. It is assumed that the injection covariance matrix is rather insensitive to injection velocity (C_3) in the range of interest $(14.5 < C_3 < 17.0)$, so that the same Q_{INJ} can be used for all dates. The variation of fuel loading with launch date/encounter date is small. The effect of the time at which the correction is made, $T_{M/C}$, is shown in figure 13 for a typical launch date/encounter date. The $\Delta V_{M/C}$ doubles about 43 days after leaving the Earth's sphere of influence.

An additional fuel allotment is necessary if the aim point at Mars is biased at Earth injection. This is done to ensure that the spacecraft will not enter the Mars atmosphere with greater probability than 10^{-4} even if no midcourse maneuver is performed. This also ensures that the transtage (Titan IIIC injection stage) will not enter the Mars atmosphere if its retro maneuver fails. A 1.0-m/sec velocity correction in the most sensitive direction can change the position in the impact parameter plane by about 25 000 km. An additional 5-m/sec is adequate for injection biasing. The second midcourse correction is sized to null out the errors at the target due to errors in the execution of the first midcourse correction. The error in the magnitude of the first midcourse correction is less than 0.1 m/sec. The fuel loading for the second midcourse correction is less than 1.0 m/sec. A fuel loading of 30 m/sec is sufficient for both midcourse corrections and injection biasing.

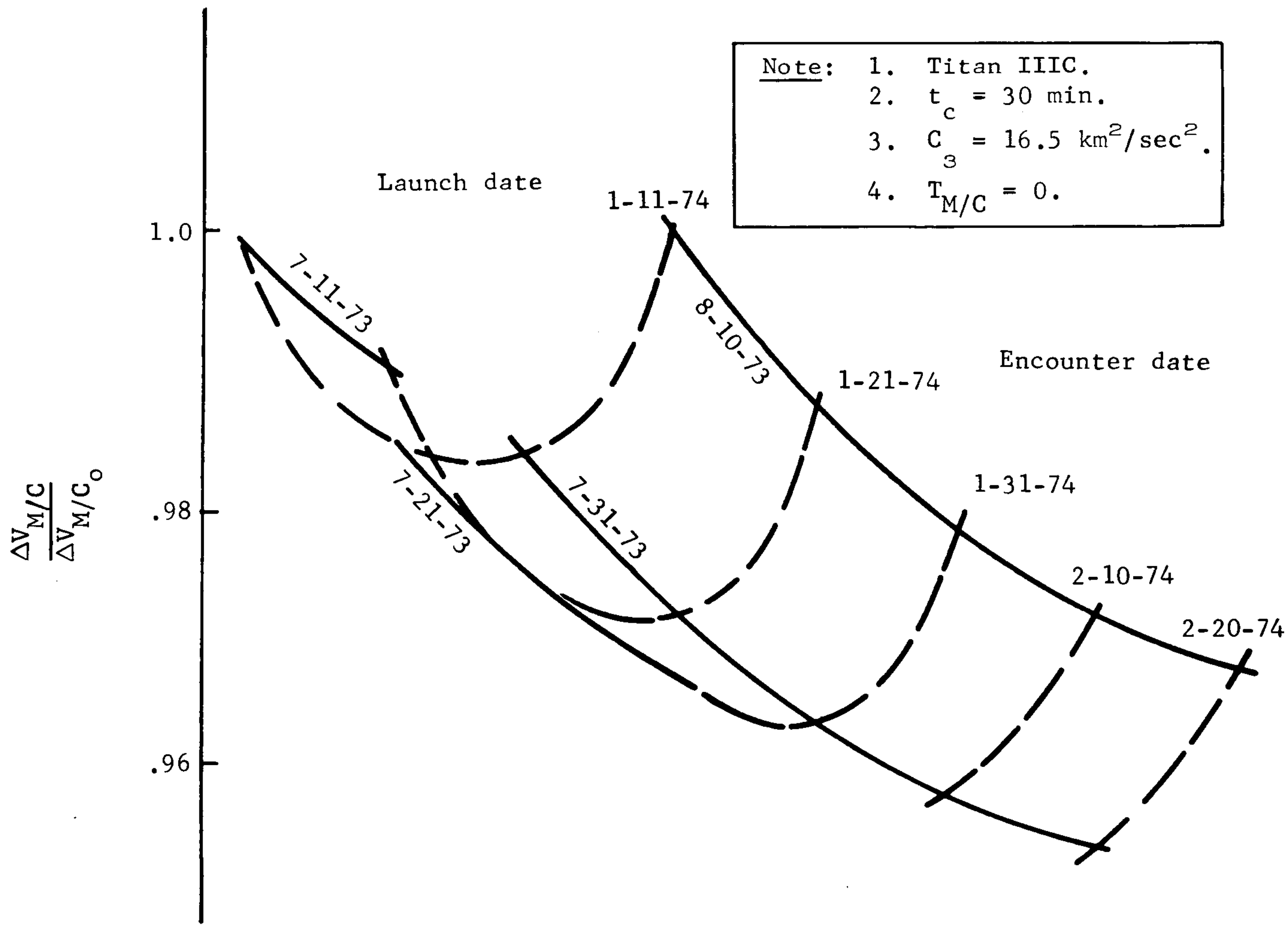


Figure 12.- Midcourse ΔV Dependence on Launch Date/Encounter Date

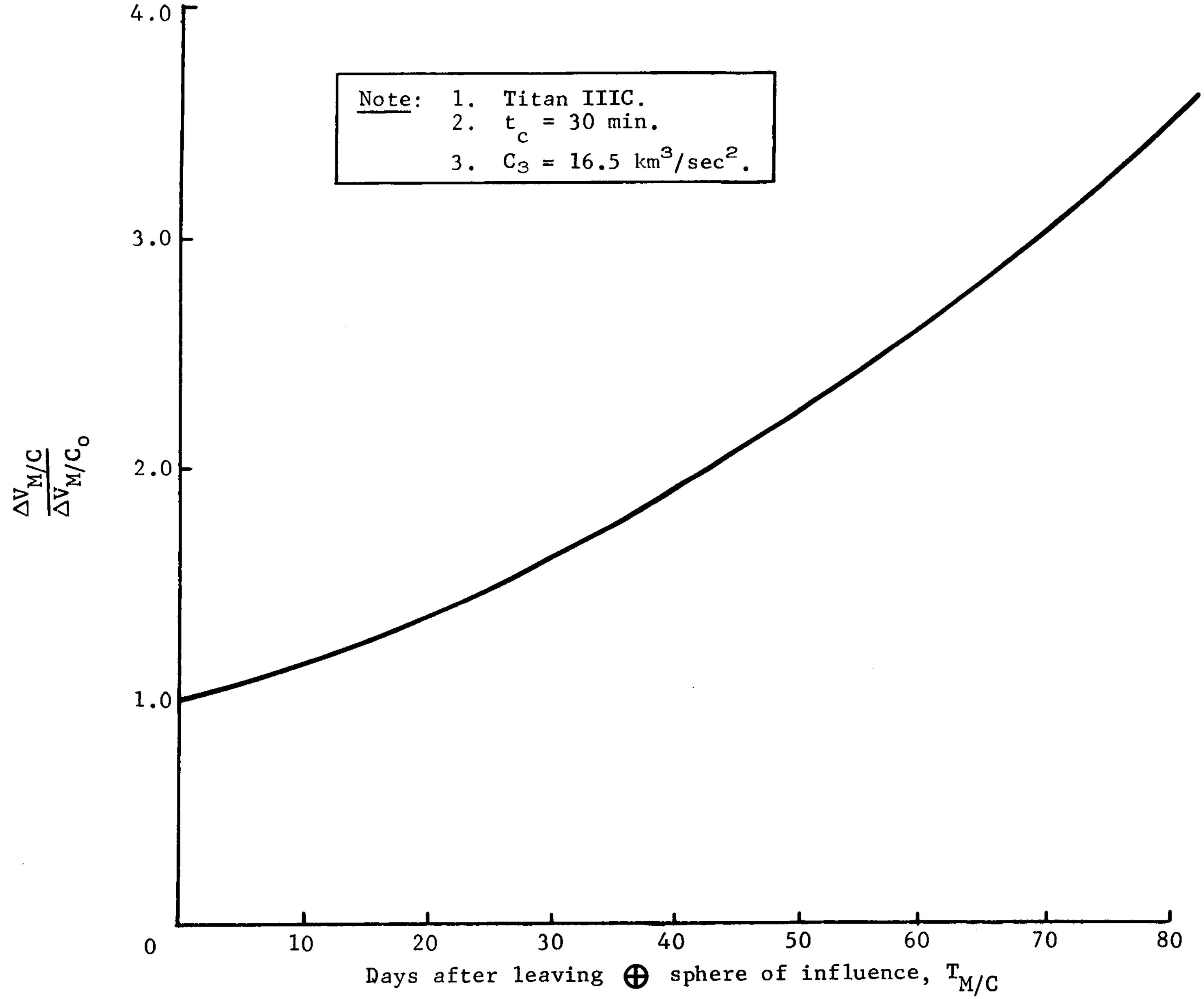


Figure 13.- Midcourse ΔV Dependence on Time of Midcourse Maneuver

6. APPROACH AND RELAY COMMUNICATION LINK GEOMETRY

The maximum allowable entry flightpath angle, γ_E, for a given entry ballistic coefficient is heavily dependent on the assumed Martian atmosphere. The maximum allowable entry flightpath angle in the minimum atmosphere is about -24° for an entry ballistic coefficient of 0.375 slugs/ft^2. It was shown in reference 1 that if projected DSN tracking capability is assumed, a nominal entry flightpath angle of -21° may be selected. The $\pm 3\sigma$ dispersion about the nominal is 3°. The projected DSN capability assumed corresponds to a 3σ navigation uncertainty in periapsis altitude at the time of lander deflection of about 110 km.

Typical approach geometry showing both the support module flyby trajectory and the lander trajectory is shown in figure 14. The view is in the trajectory plane of both the support module and lander as seen from the general direction of the Martian north pole. The support module is oriented toward the Earth, separated, and spun up some 17 hr before lander entry. The lander is reoriented for deflection and a ΔV of about 75 m/sec imparted with the ACS. The ΔV is given in a direction that speeds the lander up so that, at the time of entry, the support module lags sufficiently to ensure at least 10 min of postland data over the relay link. The science requirement for a landing site survey is thus satisfied. Three support module flyby trajectories are shown to illustrate the effect of flyby periapsis altitude. The angle between the hyperbolic excess velocity vector and the direction to Earth, ζ_E, is 160°. Contours of constant ζ_E are shown in figure 10 and a ζ_E of 160° is seen to correspond to an early encounter date. The lander entry point corresponds to a γ_E of -21° and a V_{HE} of 3.6 km/sec (entry velocity of about 20 000 fps). The lander entry point is uprange about 1.5° for every degree increase in γ_E. The entry point is approximately 5° further downrange for a V_{HE} of 3.0 km/sec. The downrange angle traversed during entry is about 12° (ref. 1).

A 22° elevation mask is shown at touchdown. The latest environmental criteria document (ref. 3) states that 98% of the landing sites on Mars have slopes less than 22°.

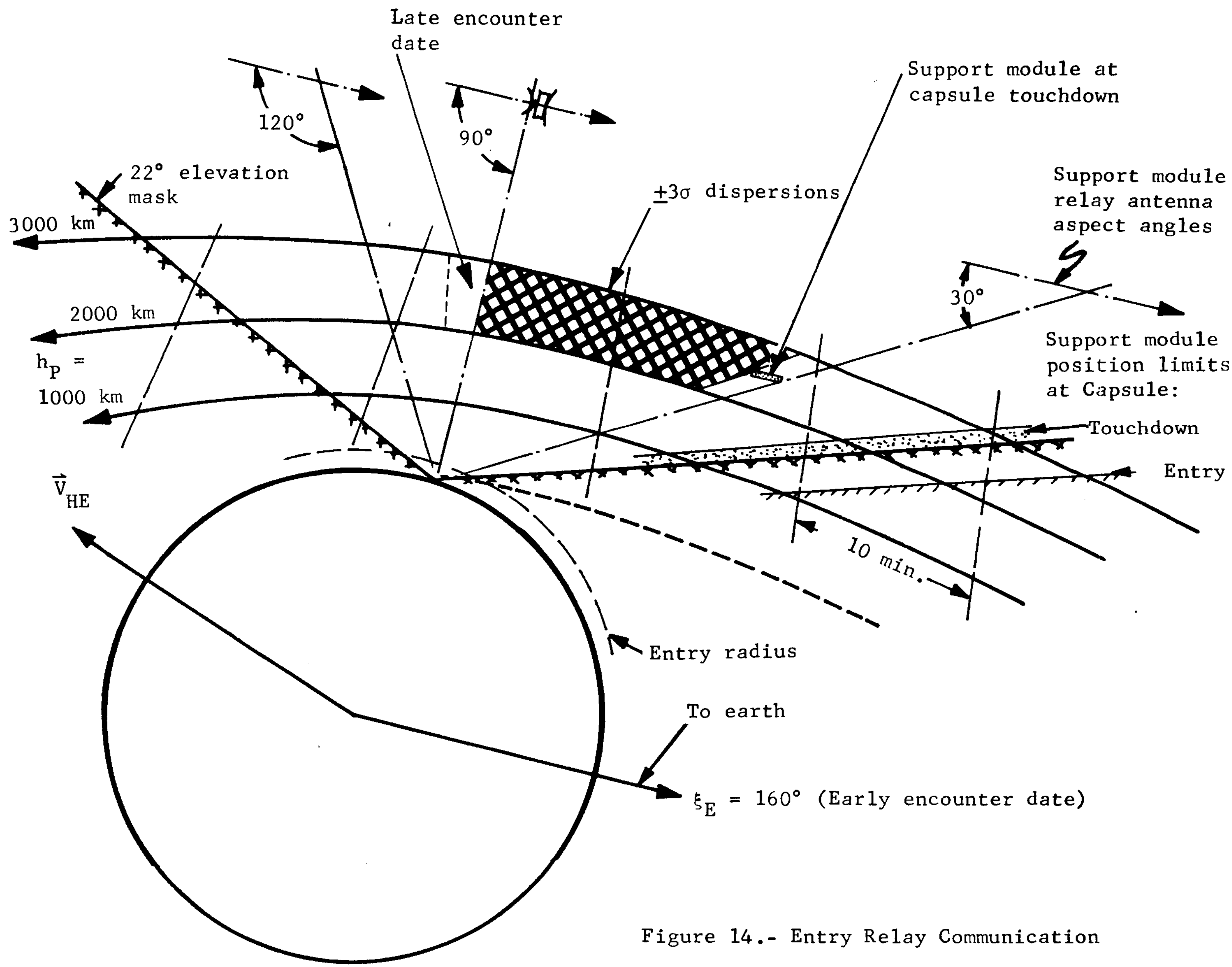


Figure 14.- Entry Relay Communication

Early in the study, a sterilized trailer mode for the support module was investigated. The trailer was separated from the lander by springs and trailed the lander to approximately the same entry point. This mode of operation is only desirable for γ_E steeper than about -40° due to the 22° elevation mask. The trailer mode was not considered further.

The maximum postland relay link time, t_{PL}, occurs when the support module at lander touchdown in the minimum atmosphere (min. entry time) is at the right-hand elevation mask as shown in figure 14. Time marks 10 min apart are shown to illustrate the length of postland relay link time available. The t_{PL} is seen to increase as flyby periapsis altitude increases. At the same time, however, communication ranges over the relay link are increasing with periapsis altitude degrading the system performance of the link. The length of t_{PL} is designed for entry in the maximum atmosphere. The entry times as a function of atmosphere, ballistic coefficient, and γ_E are summarized in figure 15. The range is from about 3.3 to 7.2 min for a γ_E of -21° ($\pm$3° 3σ) and a ballistic coefficient of 0.375.

Even in a failure mode where the parachute fails to deploy, the minimum time from the end of blackout to impact (γ_E = -24°, min. atmosphere) is 51 sec. This allows time for both relay data streams (time lag of 46 sec due to blackout) to be transmitted.

The uhf relay link on the support module is located opposite the high-gain direct link antenna. The locus of positions of the support module when its relay antenna aspect angles, $\alpha_{S/M}$, are 30, 90, and 120° are shown (fig. 14). The baseline design restricts the $\alpha_{S/M}$ to be less than 90°, which is slightly past overhead of the touchdown point for a ζ_E of 160°. This, of course, reduces the t_{PL} due to cutoff before the 22° elevation mask on the left. The position of the support module for $\alpha_{S/M}$ = 90° moves 10° further downrange for every 10° decrease in ζ_E. The ζ_E for late encounter dates can be as low as 140°. A trade study is necessary to investigate the possibility of increasing t_{PL} ($\alpha_{S/M} > 90°$) through antenna design on the support module.

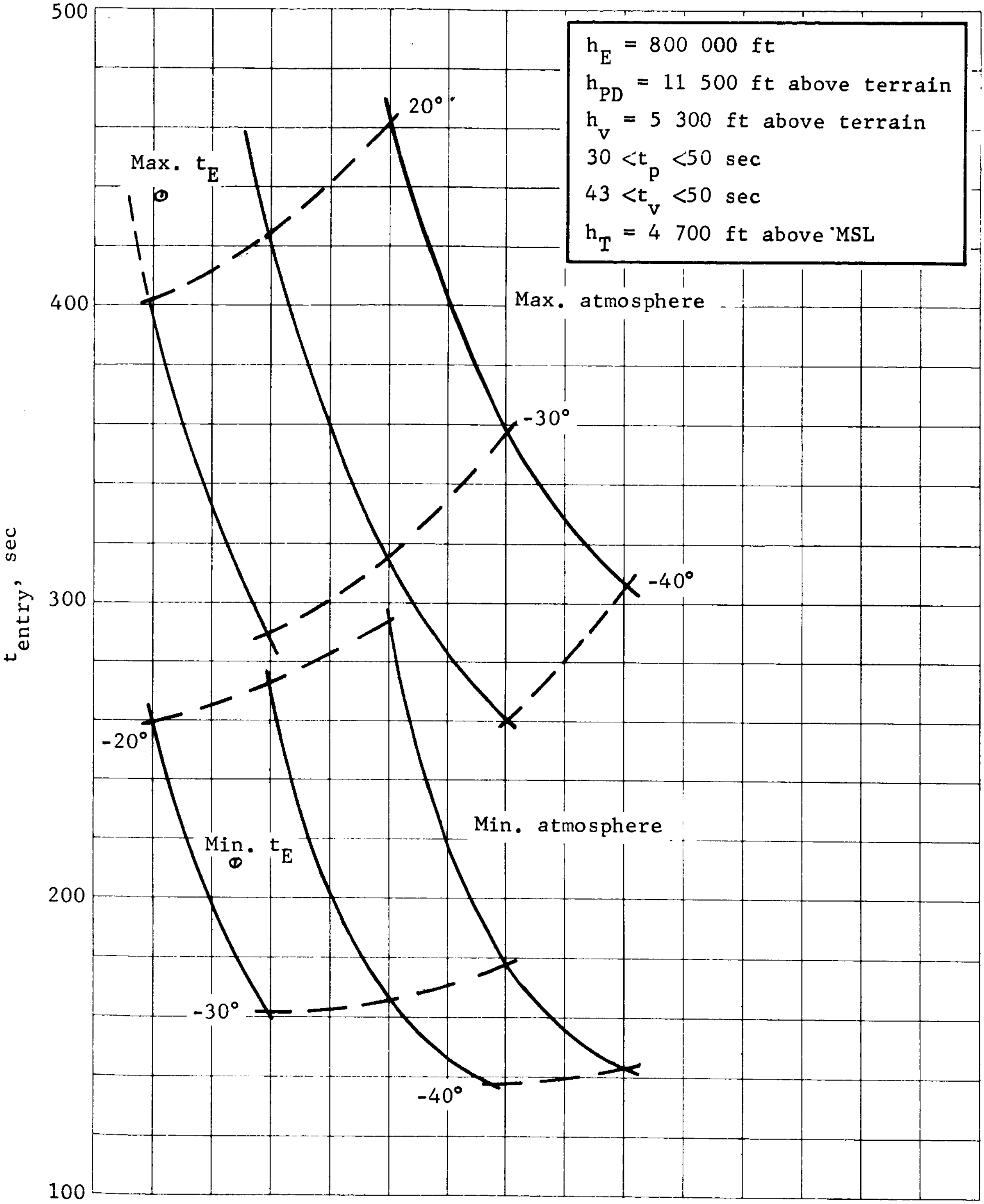


Figure 15.- Time from Entry to Touchdown

The baseline design restricts the lander antenna aspect angle, α_C, during entry to be less than 55° (or up to 77° for landing on a 22° slope). This corresponds to a support module elevation angle at touchdown of 35°. The location of the support module at capsule touchdown corresponding to a support module elevation angle at touchdown of 35° is shown in figure 14. The t_{PL} is again reduced from the theoretical maximum corresponding to the 22° elevation mask to either side of the touchdown point.

Even with these restrictions imposed to make a conservative design for the relay link, the design goal of at least 10 min postland relay link time is obtainable in the maximum atmosphere. A nominal flyby periapsis altitude of 2500 km is required. At the time of the calculation of the second midcourse correction, the navigation plus maneuver uncertainty results in a conservative 3σ error in periapsis altitude of 300 to 500 km. The error in periapsis altitude due to execution errors of the second midcourse maneuver is small in comparison to those caused by navigation uncertainty. If the nominal periapsis altitude is 2500 km, a conservative estimate of the 3σ dispersion is 500 km. Even if the predicted value of h_P turns out to be the -3σ value of 2000 km, slightly more than 10 min is obtained in the maximum atmosphere. The baseline geometry as a function of predicted h_P is shown crosshatched in figure 14.

The ΔV required for lander deflection is shown in figure 16 as a function of periapsis altitude and capsule antenna aspect angle design for entry (or support module elevation angle at touchdown). The data shown are for a γ_E of -21° and a V_{HE} of 3.6 km/sec. The ΔV_{EJ} increases for a steeper γ_E and decreases for a lower V_{HE}. A complete set of data that illustrate the variation is given in reference 1. The ΔV_{EJ} shown is for a coast time, time from deflection to lander entry, of 16.5 hr. This corresponds to distances from the center of Mars between 160 000 and 225 000 km for V_{HE} between 2.4 and 3.6 km/sec.

The ΔV_{EJ} is very nearly inversely proportional to coast time and can be easily estimated for any other coast time. Also crossplotted on the carpet plot is the t_{PL}, corresponding to a ζ_E of 160° and a maximum $\alpha_{S/M}$ of 90°. For a given desired t_{PL} the ΔV_{EJ} is nearly independent of h_P or α_C. The baseline

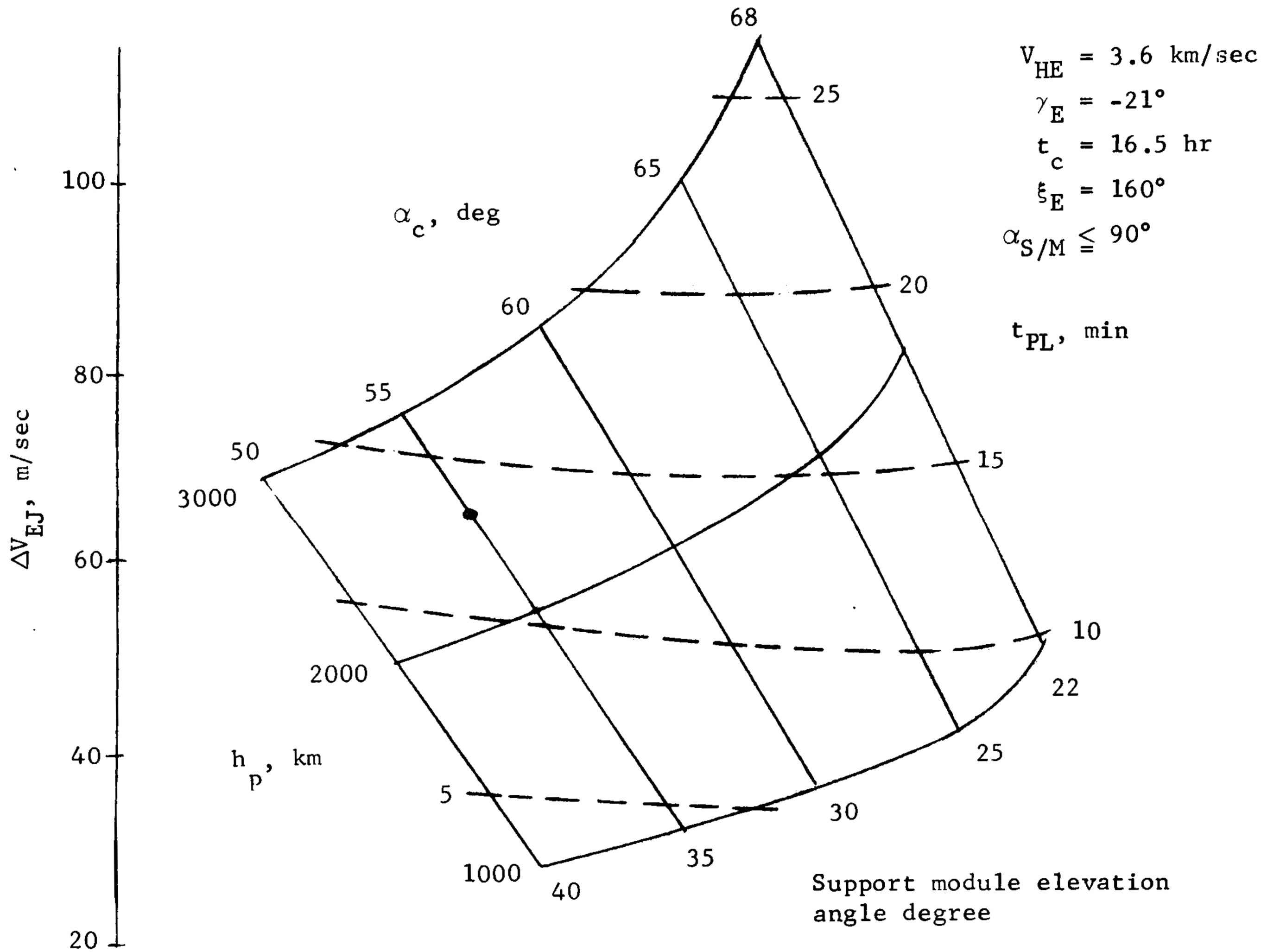


Figure 16.- Ejection Requirements

design aims for an h_P of 2500 km. Based on the predicted values of h_P at the time of lander deflection, the ΔV_{EJ} and deflection angle, τ_{EJ}, are adjusted so that the elevation angle of the support module at touchdown in the minimum atmosphere is 35°. The ΔV_{EJ} required varies between 55 and 75 m/sec. The deflection angle, angle between the velocity vector before deflection and the ΔV_{EJ}, is approximately -40° (40° down). As discussed in reference 1, deflection angles this high result in small dispersions in γ_E due to pointing errors at deflection (a 3σ error of 0.35 deg for a 3σ pointing error of 0.5°). The lander must be reoriented through an angle of about 30° after deflection to obtain a nominal zero angle of attack at entry. The t_{PL} correspondint to the reference maneuver strategy varies between 10.5 and 16 min, depending on the periapsis altitude dispersion. The ΔV_{EJ} may be minimized if the t_{PL} desired is held constant, independent of the best estimate of h_P at the time of deflection. This requires either a higher nominal periapsis altitude or a larger α_C design. A trade study is required to find the optimum maneuver strategy.

7. POSTLANDING DIRECT LINK

The primary postlanding mission lasts for about 3 days, three direct link transmission periods to Earth. The requirement is for a total of 10^7 bits of data during the three links. The S-band antenna is gravity-oriented (local vertical) after touchdown. The total number of bits of data is dependent on the bit rate and length of time of Earth visibility. The bit rate is a function of the Earth-to-Mars communication distance at encounter, transmitter power, and antenna half-power beamwidth (or equivalently antenna gain). For a 30-day launch period with a C_3 of 16.7 km^2/sec^2, the range of possible encounter dates is Jan. 14 to Mar. 1, 1974. The corresponding communication distance varies between 0.96 and 1.4 au (1 au is 149 597 890 km). The transmitter power selected is 50 W. The design antenna half-power beamwidth is 46°.

The Earth viewtime is directly a function of landing site latitude, encounter date, and antenna half-power beamwidth. The Earth viewtime within a given half-power beamwidth is a maximum when the latitude of the landing site is equal to the latitude of the sub-Earth point. For the range of arrival dates discussed above, the latitude of sub-Earth varies between -20 and -12°. The Earth viewtime is shown in figure 17 as a function of date and landing site latitude for a half-power beamwidth of 46°. The maximum viewtime increases by an hour for a half-power beamwidth of 60°. The landing site latitude is ground ruled to be between ±20°. Based on the reference antenna design and the latest encounter date, at least 3.1 hr of viewtime is required for each of the first three direct links to obtain the total of 10^7 bits. Of this time, 1/2 hr is allowed for carrier lockup so that 2.6 hr is available for data transmittal. To satisfy this viewtime constraint, the landing site latitude must be selected to be near the latitude of the sub-Earth point, i.e., southerly latitudes.

Consideration must also be given to the 90-day lifetime weather station mode of operation that requires about 5000 bits of data per day. With a design bit rate of 8-1/3 bps, this requires an Earth viewtime of 30 min (including time for carrier lockup). The landing site latitude required to satisfy both the 3-day mission and the 90-day weather-station mission are summarized in figure 18 as a function of encounter date. To satisfy the 3-day mission, the landing site latitudes must be between the boundaries corresponding to 3.1 hr. The latitudes corresponding to 0.5 hr of viewtime are also shown as a function of date. The lower boundary on latitude as a function of encounter date is found by translating the lower 0.5-hr boundary to the left by 90 days. The earliest encounter date for a 30-day launch period is Jan. 14, 1974, as shown in figure 11. The allowable landing site latitudes for this encounter date are -24 to -12.5°. The latest allowable encounter date is Mar. 1, 1974. The allowable latitudes are -12.5 to -6.5°.

If the S-band antenna were oriented normal to the spin axis of Mars with the same half-power beamwidth, 46°, landing site latitudes between ±45° are permissible. However, the required encounter dates are from Apr. 1 to May 1, 1974, which are not permissible due to the large communication distance to the Earth. Increasing the half-power beamwidth for a gravity-oriented antenna to 60° does not yield the required number of bits of data as discussed in the Telecommunications Section of volume II.

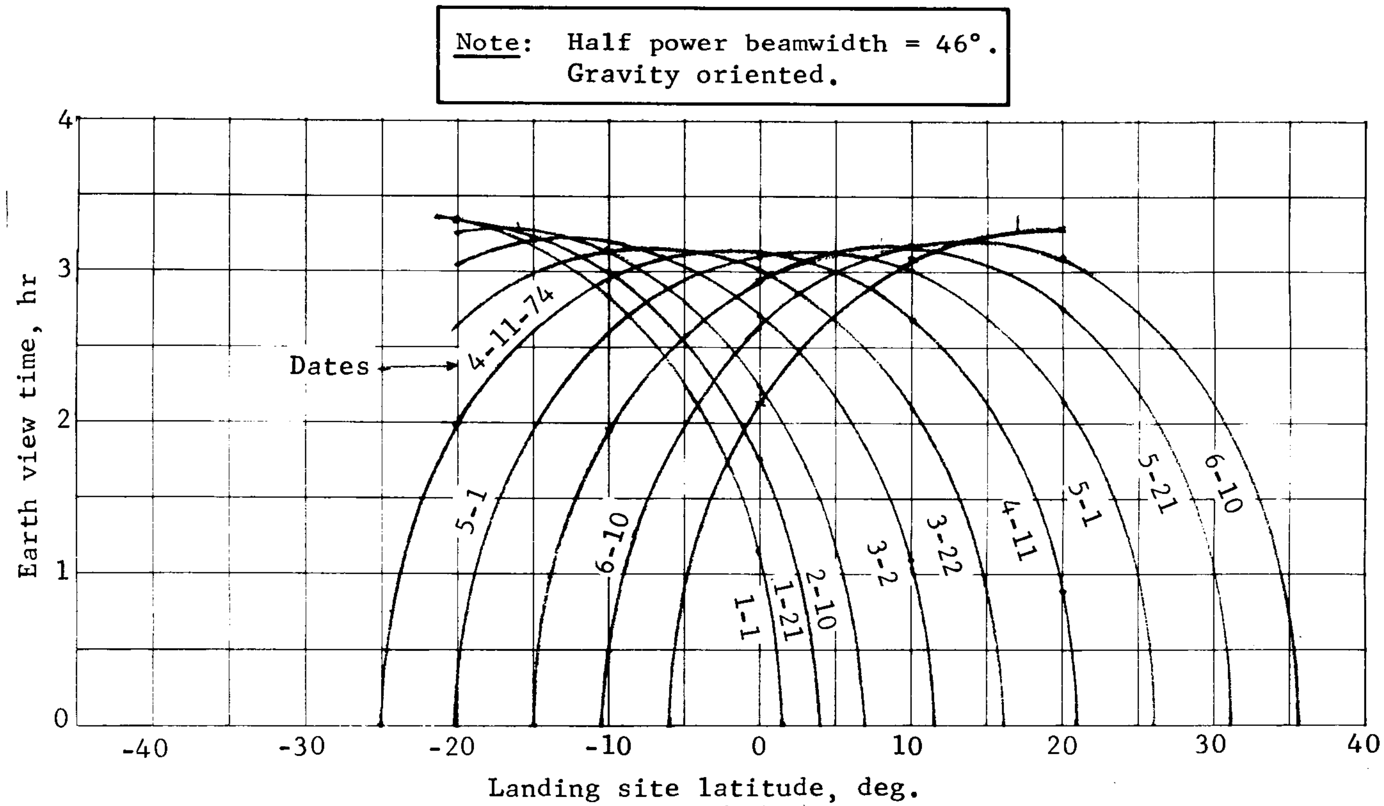


Figure 17.- Postlanding Earth Visibility

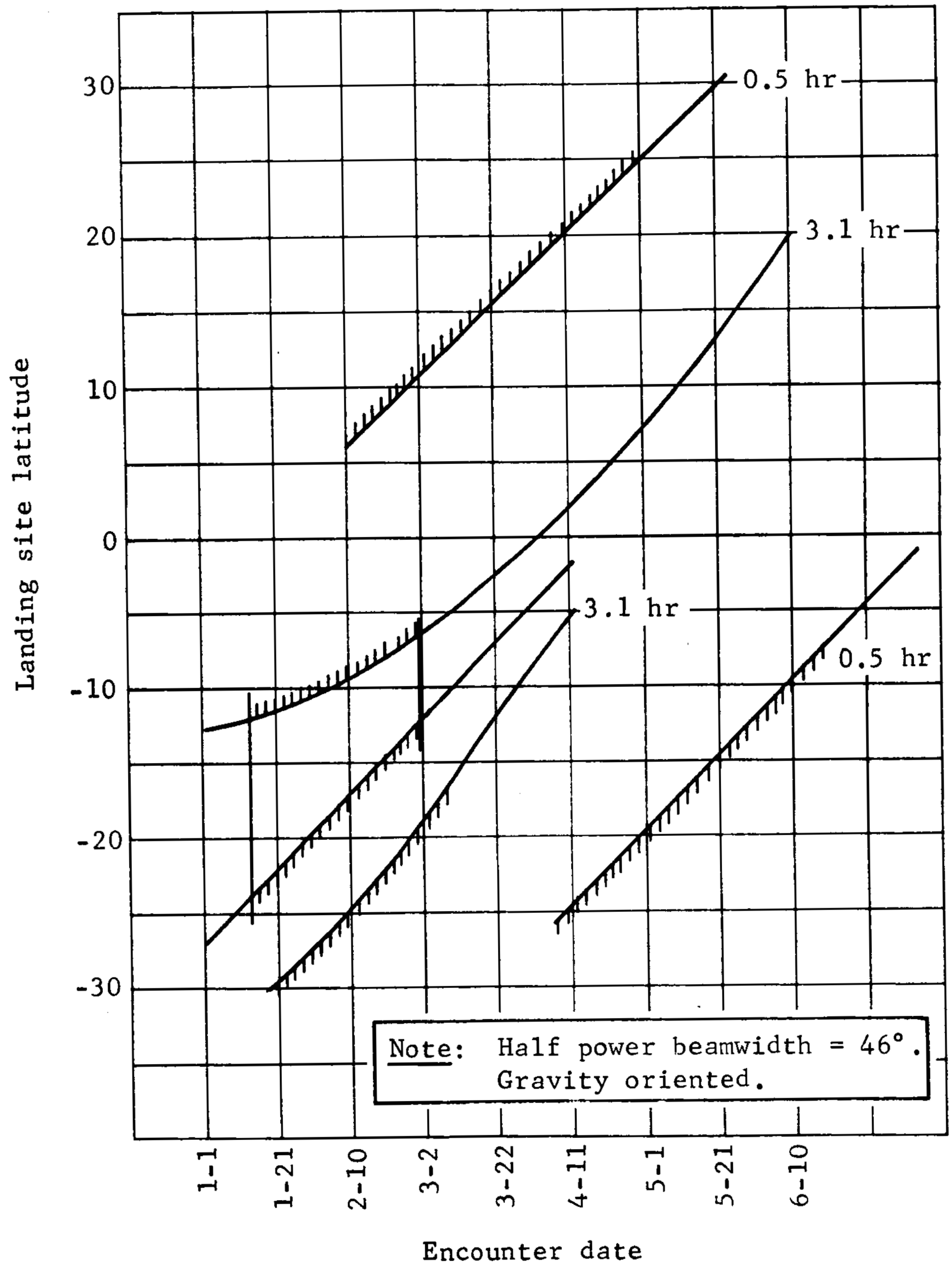


Figure 18.- Landing Site Latitude Flexibility

8. MISSION ANALYSIS CONCLUSIONS

The mission analysis conclusions drawn from this study for the soft lander/support module mission mode concept are as follows:

1) The Titan IIIC provides adequate performance and margins;

2) A 20-day launch period is possible;

3) Longitude targeting flexibility allows full planet coverage. Latitude flexibility is limited by simple direct communication link configuration;

4) A nominal operational sequence has been developed that has sufficient margin to cope with uncertainties, tolerances, and contingencies during the flight;

5) A scientifically significant mission can be realized with this configuration and mission mode;

6) Further trade studies and performance optimization studies can lead to better communication times, more data return, and greater landing site latitude selection flexibility.

CONFIGURATION STUDIES

1. PREFERRED APPROACH

The preferred configuration resulting from this study is an attitude stabilized soft lander using the direct entry mode with a flyby unsterilized support module. Table 7 presents launch to landing performance parameters.

TABLE 7.- PERFORMANCE PARAMETERS

Launch vehicle	Titan IIIC
Launch date	Jul. 18, 1973
C_3, km^2/sec^2	15.7
Arrival date	Jan. 25, 1974
V_{HE}, km/sec	3.3
Injected payload capability, lb	2787
Spacecraft weight, lb	2568
Space vehicle margin, lb	219
$\Delta V_{M/C}$, m/sec	30
Encounter weight, lb	2350
Flyby periapsis altitude, h_P, km	2500
Deflection ΔV, m/sec	75
γ_E, deg (max.)	-24
V_E, fps	19 200
B_E(11-ft diam), slug/ft^2	.375
Entry weight, lb	1859
Parachute deployment altitude, ft	11 500
B_{DEC} (55-ft diam chute), slug/ft^2	.038
Vernier ignition altitude, ft	5300
Useful landed weight, lb	947

System Definition

The spacecraft, which consists of the flight capsule and its support module, performs all flight functions from separation from the Titan IIIC launch vehicle through the end of the 90-day mission on the Mars surface.

The spacecraft is composed of six major assemblies shown schematically in figure 19. Major components in each assembly are identified. To soft land the 947 lb of useful landed weight, a 2568-lb total spacecraft is required. The lander has an aeroshell diameter of 11 ft requiring a 12.5-ft bulbous shroud on the launch vehicle. Figure 20 shows the preferred spacecraft integrated with the shroud and last stage of the launch vehicle. Figures 21 thru 23 present a layout study of the flight capsule and support module elements.

The major system elements and characteristics are described in the following paragraphs.

Science.- The science subsystem comprises instruments for obtaining science, data handling equipment for special science data conditioning, formatting, encoding, storage, and instrument sequencing.

Instruments included are an accelerometer triad, a stagnation pressure sensor, and a total temperature sensor for obtaining data during the ballistic entry phase. From parachute deployment to landing, ambient pressure and temperature sensors, a hygrometer, and a double-focusing mass spectrometer obtain atmospheric data.

Landed experiments are a facsimile camera for panoramic and site survey imaging, a pyrolysis/gas chromatograph-mass spectrometer (GC-MS) experiment for analyzing soil samples, a biological experiment, a soil sampler, and a subsurface probe for measuring soil temperature and humidity. Landed atmospheric experiments are temperature, pressure, windspeed, and composition.

Data handling equipment consists of digital multiplexers, signal conditioners, and encoder, an instrument sequencer, and a static data storage element.

Structures and mechanisms.- The spacecraft structure includes the following major elements -- adapter/sterilization canister, support module structure, aeroshell, aerodecelerator, lander, and landing system.

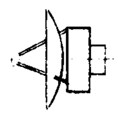
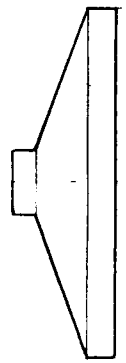
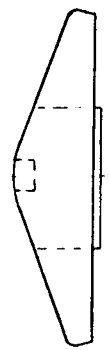
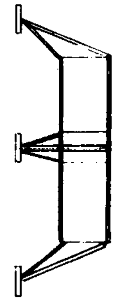
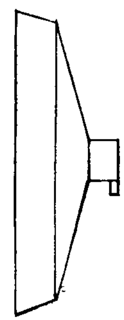
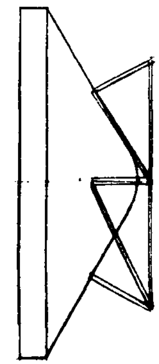
					
Support module	Cruise module	Aeroshell assembly	Lander	Aerodecelerator assembly	Canister adapter
S-band low-gain antenna S-band high-gain antenna uhf low-gain antenna Spin rockets Nutation damper Thermal insulation S-band transmitter uhf receiver Storage batteries	Forward canister skin and structure Cruise solar array Sun sensors Canopus sensor Cruise ACS Thermal insulation Separation systems	Aeroshell Heat shield Entry science Lander adapter AMR antenna	Lander structure Landing legs Hot gas ACS Vernier propulsion Landed science Guidance and Control Thermal insulation Isotope heaters Landed solar array Communication equipment Storage batteries Separation systems Data acquisition systems	Parachute Mortar Mortar reaction energy absorber Mortar support structure Base cover S-band low-gain antenna Separation system	Aft canister skin Biovent Truss Canister seal Canister separation

Figure 19.- Six Major Spacecraft Assemblies

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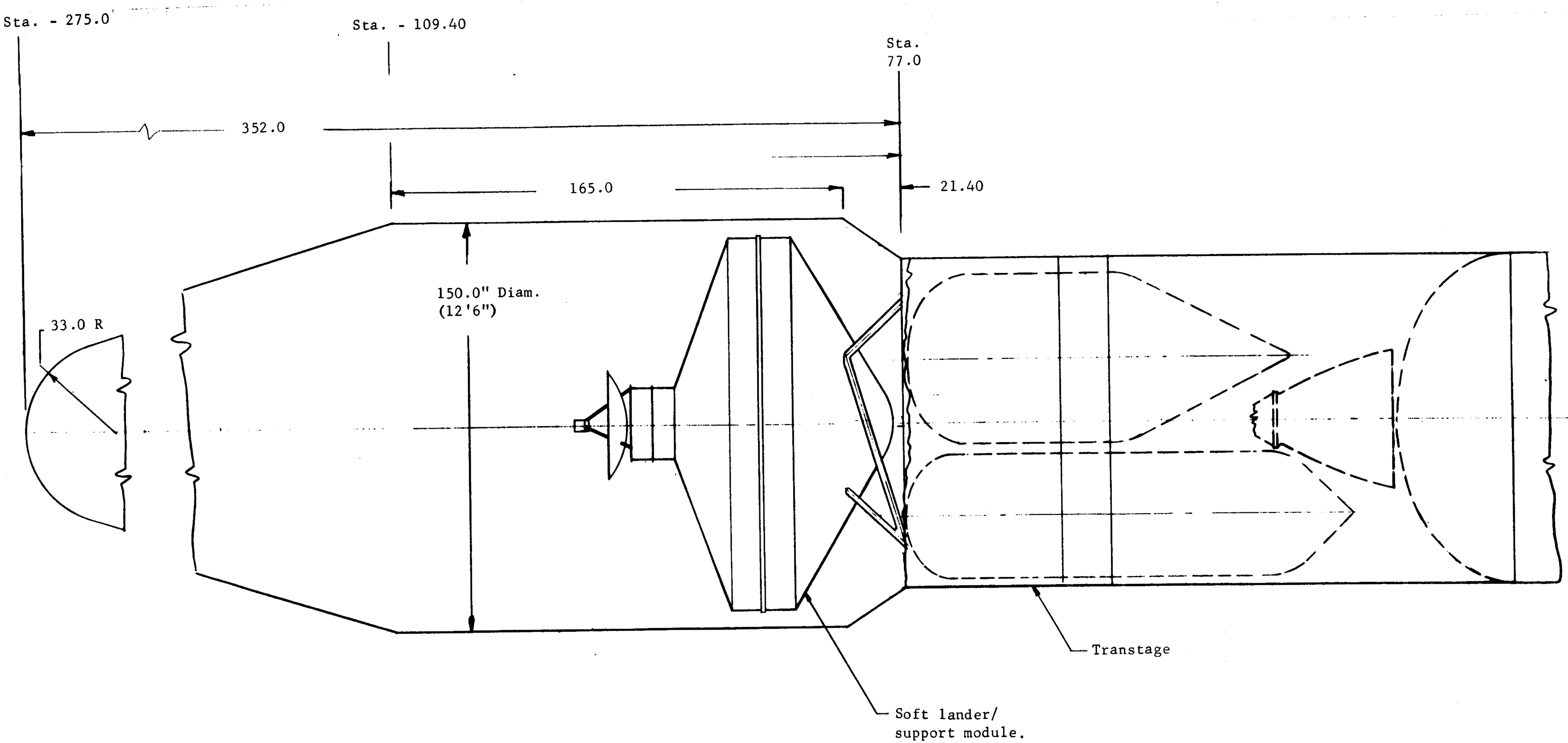


Figure 20.- Integration Sketch, Soft Lander/Support Module (Unsterilized)

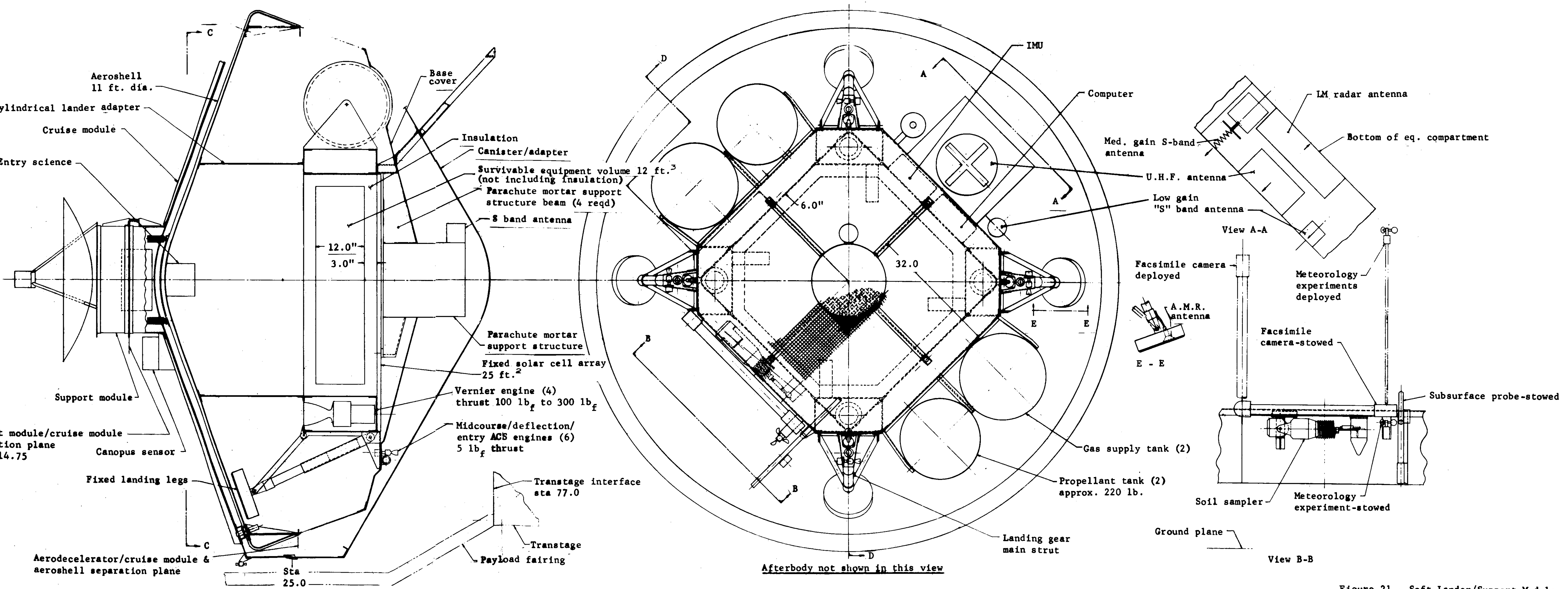


Figure 21.- Soft Lander/Support Module

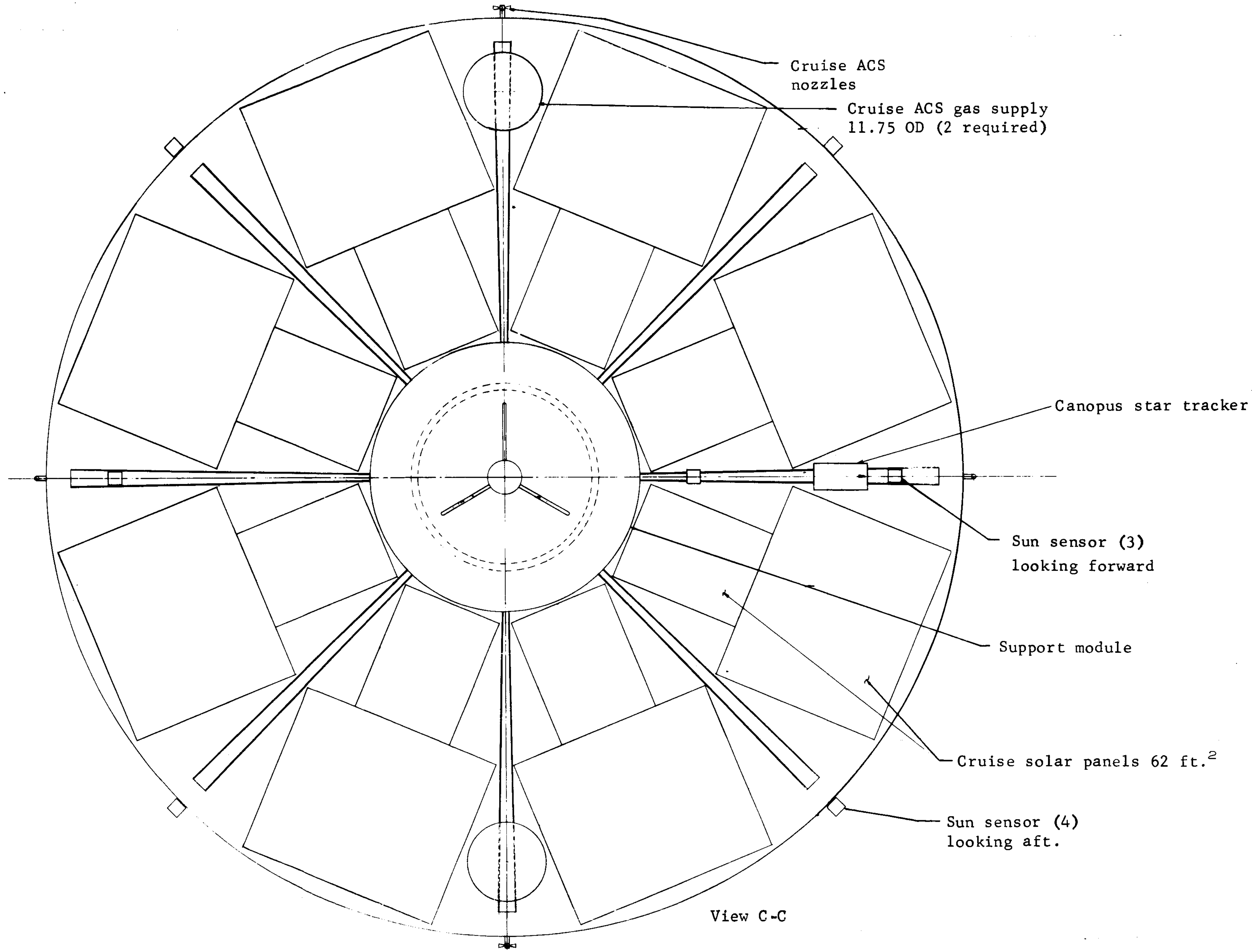


Figure 22.- Soft Lander/Support Module

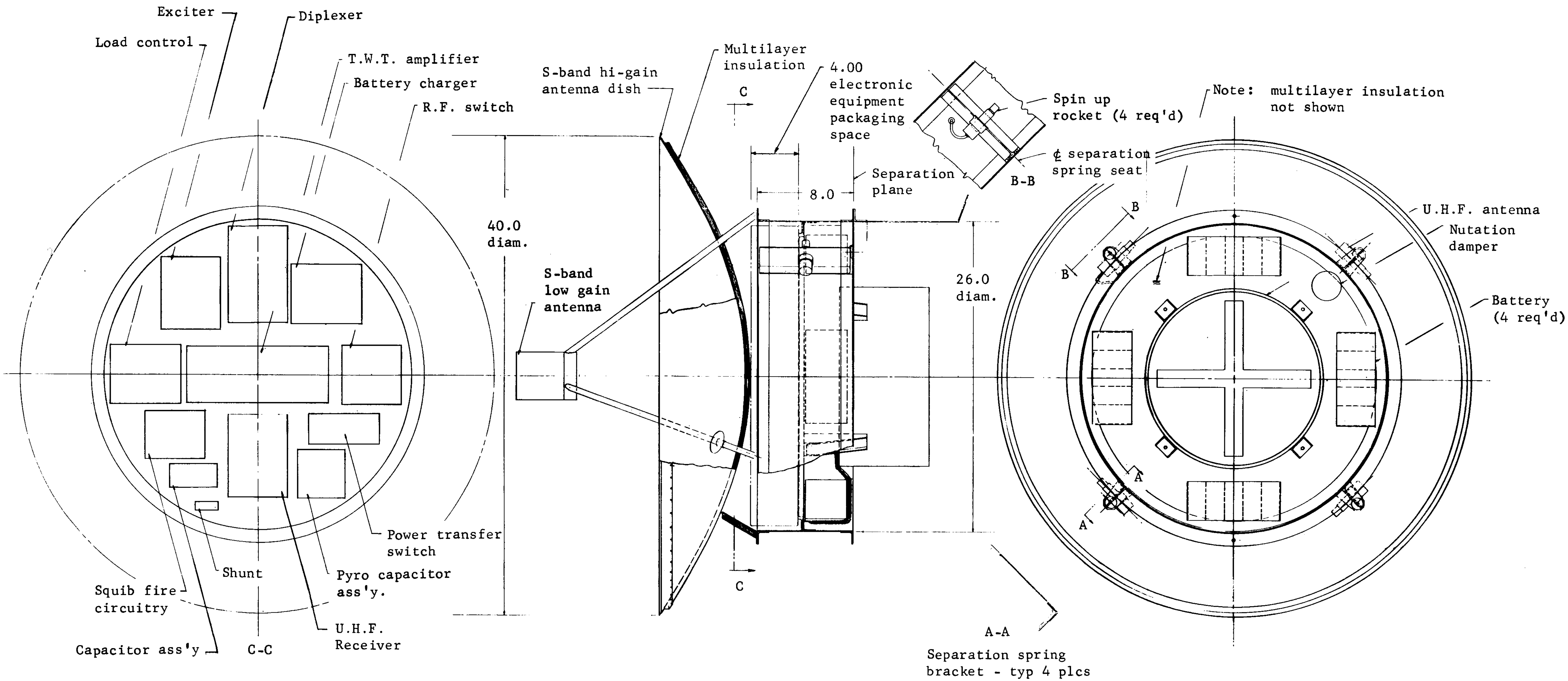


Figure 23.- Support Module

The adapter/sterilization canister has two distinct parts. The aft portion covers the launch vehicle end of the sterilized payload and provides a structural path between the spacecraft and the launch vehicle. It seals to the forward half to exclude biota after sterilization and is provided with biovents to maintain a low differential pressure during launch (1.0 psig or less). The aft section remains with the launch vehicle when the spacecraft is separated. The forward section remains with the spacecraft to encounter. It serves a dual-function during this time -- (1) separates the unsterilized support module and cruise-peculiar equipment from the sterilized flight capsule, and (2) serves as the mounting structure for the cruise solar array, the cruise ACS, and the sun and Canopus sensors.

The support module structure is a disc-shaped equipment-mounting baseplate centrally mounted in a short cylindrical body. The skin also serves as a radiator to maintain temperature control of the equipment.

The aeroshell is a 70° half-angle cone, 11 ft in diameter covered with an SLA-561 ablator heat shield material. A beryllium spherical segment cap is provided for entry science. A cylindrical aluminum adapter provides a structure between the aeroshell and the lander. The primary structure is conventional ring-stiffened aluminum construction.

The aerodecelerator assembly is composed of a parachute, harness, mortar, mortar reaction assembly, mortar-support structure, and a base cover (backface). The parachute, 55 ft in diameter, is deployed at Mach 2 on an altitude mark command from the guidance and control system. The lander is suspended on a four-link bridle attached through a swivel to the risers.

The lander structure is conventional aluminum web and cap design with provisions for equipment mounting, landing gear, parachute, and aeroshell attachment.

The landing system consists of four equally spaced legs with crushable foot pads and Surveyor-type main struts.

<u>Propulsion</u>.- Four 300-lb-thrust throttleable monopropellant engines are used for the propulsive landing maneuver; pitch and yaw attitude control is maintained by differential throttling while roll control is provided by the roll ACS thrusters. Midcourse maneuvers, deflection ΔV, and flight capsule attitude control from separation to parachute deployment is provided by 5-lb-thrust monopropellant engines. Two pitch, two yaw, and four roll engines are used. Blowdown pressurization, with gaseous nitrogen, is used in common tankage for both systems.

Positive propellant orientation is provided by bladders. Attitude control during the trans-Mars cruise phase is maintained by a cold gas (GN_2) system of 12 thrusters, 4 for each axis providing pure couples.

Guidance and control.- The guidance and control system comprises an inertial measurement unit (IMU), a general-purpose digital computer, sun and Canopus sensors, valve drive amplifiers, a terminal descent and landing radar, two radar altimeters, and a sequencer for landed operations.

The IMU contains three strapped-down gyros for attitude reference and three accelerometers for axial acceleration sensing. This unit provides the vehicle references for all maneuvers continuously from flight capsule separation to parachute deployment. During the terminal descent phase, it is used in the backup inertial guidance mode and the final descent to the surface.

The 4000-word computer fulfills all computing, seuqencing, and command decoding requirements for the spacecraft from launch vehicle separation until landing on the Mars surface.

The modified LM terminal descent and landing radar furnishes velocity and range data required for terminal descent and landing.

Altitude data for science data correlation and an altitude mark for parachute deployment are provided by a radar altimeter that operates from 200 000 ft to aeroshell separation. A second altimeter is used in the backup guidance mode for the terminal descent and landing phase. Mariner '69 sun and Canopus sensors provide the reference during the interplanetary cruise phase of the mission.

The support module is spin-stabilized after separation from the flight capsule by four 5-lb solid rocket motors. A nutation damper is included to damp out residual oscillations. A simple sequencer is included in the support module to initiate spinup and control the transmitter.

Telecommunications.- Three communication systems are used in this system; a low-gain S-band, a uhf relay, and a medium-gain S-band direct link. The S-band system, with two low-gain antennas providing omnidirectional coverage, is used during the trans-Mars mission phase. This system, with the exception of one of the low-gain antennas, is located in the support module. The same system using a 40-in. high-gain antenna is used to relay real-time data to Earth from flight capsule separation through landing. Data from the flight capsule to the support module during this period are transmitted via a uhf link. An antenna, receiver, and bit demodulator are located in the support module to accomplish this function. After the initial postlanded period, all lander communications are direct S-band link to Earth. The S-band links described also support the command and tracking functions.

The telemetry subsystem has a data encoder with multiple modes and output data rates of 8-1/3 and 33-1/3 bps for use during cruise, 2400 bps during entry, and 8-1/3, 80, 210, and 353 bps after landing. Data storage, formatting, and signal conditioning are provided in this subsystem.

Power and pyrotechnics.- Power for the spacecraft is provided from one or a combination of four sources. Solar panels mounted on the forward canister furnish the power during the trans-Mars cruise phase. During maneuvers, lander silver-zinc batteries provide the energy required. Power for the support module is furnished by an unsterilized silver-zinc battery. The flight capsule uses the lander batteries from separation through landing. After landing, power is supplied from either the lander batteries or a fixed solar array depending on the availability of solar energy.

All pyrotechnic functions are initiated by capacitor-stored energy. Pyrotechnic devices have two squibs per function and one bridgewire per squib. Solid-state safe/arm and fire switches are used.

Thermal Control.- Thermal control of the support module and flight capsule before landing is achieved by passive means using multilayer insulation, coatings, and thermostatically controlled heaters. The Mars surface thermal control incorporates radioisotope heaters, 3 in. of surface insulation, and phase change material on the S-band transmitter. Proper temperature is maintained by moving one or more of the four 50-W radioioscope heaters into or out of the insulated area by thermostatically controlled actuators.

Functional Sequence

During the terminal launch countdown, portions of the guidance and control, power, and telecommunications subsystems are energized; all other systems are dormant. After injection into the Mars transfer trajectory and separation form the launch vehicle, the cruise ACS and the communication equipment are activated. When the spacecraft has emerged from the Earth's shadow, sun/Canopus references are acquired and the system is powered down for cruise. Data are transmitted continuously at 33-1/3 bps until the first midcourse corrections are completed.

Two midcourse maneuvers are performed, if required. Commands are sent from Earth controlling the direction, magnitude, and time of these maneuvers. On receipt of these commands, the inertial guidance mode is activated, the 5-lb thrusters are enabled, and the maneuver is performed. Following the thrusting period, sun/Canopus references are reacquired, the thrusters are disabled, and the system is powered down for cruise. Shortly after the first midcourse maneuver, the data rate is reduced to 8-1/3 bps until encounter.

As the spacecraft approaches Mars, commands from Earth are received updating the separation and deflection maneuver parameters, if required. About 17 hr before entry, the system is again put on inertial guidance and oriented with the spacecraft roll axis pointed toward Earth. The support module is activated, data are transmitted through the high-gain antenna at 2400 bps, and the lander uhf system is turned on. The support module is then separated, spun up, and continues on a flyby trajectory with a periapsis of 2500 km.

The 5-lb monopropellant thrusters are again enabled to perform the attitude control function for the remainder of the flight mission. The vehicle is then oriented to the deflection maneuver attitude. Approximately halfway through this maneuver, the cruise module is separated, taking the cruise solar array, cruise ACS, and sun and Canopus sensors. Lander battery power is used through landing. After attaining the proper orientation, the four 5-lb pitch/yaw thrusters are fired to deflect the flight capsule into the Mars atmosphere. The capsule and support module are then powered down until just before entry.

One hour before entry, the communication links are reactivated. The capsule is oriented to the correct entry attitude, and entry science is turned on. During entry, the ACS is switched to a rate damping mode. The high-altitude radar altimeter is activated at 200 000 ft. Parachute deployment is commanded on receipt of the altitude mark from the altimeter, and the terminal descent sequence is initiated. Aeroshell separation occurs shortly after parachute deployment.

The terminal descent and landing radar (TDLR), which is activated at aeroshell separation, commands vernier engine ignition when the proper altitude is reached. The low-altitude altimeter, which was also activated at aeroshell separation, is a backup for this function. Engine ignition and proper operation are verified, the parachute system is jettisoned, followed by vernier descent under closed-loop TDLR control to 60 ft. The final constant velocity letdown is performed on inertial guidance with altitude information from the radar. Ten feet above the surface, the descent is stopped, the engines are shut down, and the lander is allowed to free fall to the surface.

Before loss of the relay communications link ($\geq$ 10 min) the facsimile camera and atmospheric instruments are deployed, and the low-power sequencer is activated. An image of the soil sample site is obtained (and possibly a panoramic view and high resolution image) and transmitted to the support module for relay to Earth.

Weather-station data and engineering are collected every 2 hr and stored for transmission to Earth when the link becomes operational. This operation is continued until the end of the mission. Atmospheric composition data obtained by the mass spectrometer are obtained at 8-hr intervals through the first day.

When the Earth is in the gravity-oriented antenna pattern the following day, transmission is initiated at 353 bps. Thirty minutes is allowed to acquire the carrier on Earth. Most of the data transmitted during this period is imaging, which is sent real time; however, all stored data and selected real-time engineering data are also transmitted. A total transmission period of approximately 3 hr is used. During the early part of this period, the command system is locked up. If alternative soil sample or subsurface probe deployment sites are desired (as a result of image evaluation from the relayed data), commands to update these programs are transmitted from Earth. These instruments are then activated, the sample collected, conditioned and distributed, the soil analysis conducted, and the life experiment started. The latter experiment is continued for a minimum of 2 days and up to 2 weeks if power is available from the solar array. The second and third days operation is similar to the first day except for initiation of the soil and life experiments and deletion of the atmospheric composition cycle.

Days 4 thru 90 are planned for weather-station operation, solar array power permitting. The data rate is reduced to 8-1/3 bps and the transmission time to 40 min. Command capability is provided to alter this sequence if excess power is available and alternative data rates of 80 and 210 bps may be selected to increase the data return.

Weight Statement

A detailed sequential weight statement for the preferred configuration is presented in table 8.

TABLE 8.- SEQUENTIAL WEIGHT STATEMENT, PREFERRED CONFIGURATION

Total spacecraft weight			(2568)
Aft canister and adapter			179
Structure skin and rings		104	
Adapter truss		40	
Fittings and brackets		14	
Separation mechanisms		7	
Umbilical connectors, tubing and wiring		9	
Biological vents		5	
Spacecraft weight (separated weight)			(2389)
VCS propellant (30 m/sec)			32
ACS propellant			7
Encounter weight			(2350)
Support module			126
Structure		15	
Floor	9		
Skin and frames	3		
Brackets	3		
Power		43	
Power transfer switch	2.5		
Battery charger	3		
Load control assembly	2		
Diodes, shunts, and isolation assemblies	1.5		
Ag-Zn battery (50 A-h)	34		
Pyrotechnics		6	
Capacitor assembly	2		
Safe/arm switch assembly	3		
Squib firing switch assembly	1		
Control and sequencing		9	
Sequencer	3		
Spin rockets	4		
Nutation damper	2		

TABLE 8.- SEQUENTIAL WEIGHT STATEMENT, PREFERRED CONFIGURATION - Continued

Telecommunications		39	
uhf receiver and bit demodulator	6		
uhf receiving antenna and supports	6		
Modulator-exciter	3		
TWTA and power supply	9		
S-band diplexer	4		
S-band high-gain antenna	6		
S-band low-gain antenna and supports	2		
rf-switch assembly	3		
Thermal control		2	
Insulation	1		
Heater and thermostats	1		
Cabling, supports and mechanisms		12	
Separation mechanisms	2		
Separation connector	2		
Cabling and supports	8		
Forward canister			170
Support module adapter		8	
Skin		47	
Longerons, stiffeners, and rings		68	
Separation mechanisms		13	
Brackets		19	
Insulation		11	
Separation connectors		4	
Canister mounted equipment			126
Inverter		2	
Sun sensors		2	
Canopus sensor and sun shutter		9	
Cruise solar array		62	
Cruise ACS		44	
Tanks	18		
Thrusters	6		
Components and lines	5		
Supports	8		
Residuals	7		
Cabling and supports		7	
Separated capsule weight			(1928)
Deflection ΔV propellant (75 m/sec)			68
ACS propellant			1
Entry weight $(B_E = 0.375)$			(1859)
Aeroshell (11-ft diam)			256
Structure		141	
Ablator		105	
Separation mechanisms		5	
Miscellaneous		5	

TABLE 8.- SEQUENTIAL WEIGHT STATEMENT, PREFERRED CONFIGURATION - Continued

Science in aeroshell			8
Instruments		4	
Cabling and packaging		4	
ACS propellant			1
Decelerator load			(1594)
Parachute system (55-ft diam, qd = 17 psf)			125
Backface			104
Skin and stiffeners		52	
Parachute canister supports		12	
rf window		5	
Ablator		17	
Insulation and coatings		17	
S-band antenna		1	
Lander weight (verniered weight)			(1365)
Lander VCS propellant			159
Lander ACS propellant			1
Landed weight			(1205)
Lander propulsion and residuals			258
Fuel tanks		12	
Nitrogen		11	
Nitrogen tanks		12	
Landing engines		140	
ACS thrusters		9	
Components and lines		24	
Supports		37	
Residuals		13	
Useful landed weight			(947)
Lander structure			209
Truss		155	
Landing legs		54	
Lander subsystems			(655)
Power		273	
Ag-Zn batteries (3 - 80 A-h)	186		
Solar array (25 ft^2)	17		
Battery chargers	9		
Converter/regulator	7		
Power transfer switch	2		
Load control assembly	4		
Shunts and isolation	2		
Diode assemblies	2		
Undervoltage sensors	6		
Cabling, packaging and supports	38		

TABLE 8.- SEQUENTIAL WEIGHT STATEMENT, PREFERRED CONFIGURATION - Continued

Guidance and control			146	
Inertial measurement unit		20		
Computer		28		
Landing radar		39		
AMR-1		15		
AMR-2		14		
Valve drive amplifiers		6		
Low power sequencer		3		
Cabling and supports		21		
Telecommunications			103	
uhf transmitter		3		
uhf antenna and coupler		4		
Modulator-exciter		3		
TWTA and power supply		9		
S-band medium-gain antenna		2		
S-band low-gain antenna		1		
rf switch assembly		2		
Command receiver		5		
Command detector		4		
Signal conditioner		1		
Data encoder		16		
Transducer power supply		1		
Sterilization/battery measurement multiplexer		4		
Static storage		4		
Cabling and supports		44		
Thermal control			77	
Survivable equipment		71		
Isotope heaters with actuators				
Insulation	46			
Phase change material				
Thermal switches	5			
Insulation support structure	20			
Phase change material on computer		6		
Pyrotechnics			56	
Capacitor assemblies		2		
Safe/arm assemblies		17		
Squib firing switch assemblies		5		
Cabling, packaging and supports		32		
Landed science				83
Entry science (mounted on lander)			14	
Accelerometers		4		
Hygrometer		2		
Mass spectrometer		8		

TABLE 8.- SEQUENTIAL WEIGHT STATEMENT, PREFERRED CONFIGURATION - Concluded

Landed science		39
Facsimile camera	5	
Pyrolysis/GC-MS	16	
Biology	8	
Soil sampler, processing, and distribution	2	
Subsurface probe	3	
Hygrometer	1	
Atmospheric temperature	1	
Atmospheric pressure	1	
Anemometer	2	
Data handling		12
Data automation unit	8	
Core storage unit	4	
Cabling and supports		12
Deployment mechanisms and masts		6

2. ALTERNATIVE CONFIGURATIONS

Other configurations investigated during this study included:

1) A sterilized support module;

2) Support module trajectories that allow it to enter the Mars atmosphere;

3) An attitude stabilized support module;

4) A solar powered support module;

5) A flight capsule communications subsystem that provides all cruise communications using the support module only for relaying entry data.

The following paragraphs briefly discuss the pros and cons of each of these alternatives.

A configuration was investigated in which the entire spacecraft is enclosed in the sterilization canister and sterilized. This allowed support module trajectories that enter the Mars atmosphere to be considered. This study began looking at a "trailer" mode of operation, i.e., the support module was separated with a low relative velocity and allowed to trail the capsule into the atmosphere. Obtaining communications geometry through entry and the immediate postlanding period proved impractical. Flyby trajectories are required to satisfy these requirements. The sterilized support module configuration investigation was continued, however, because of some apparent advantages in total system weight, a more integrated system and potential use of identical equipment in the support module and the lander. It was subsequently judged that the disadvantage of having to sterilize the support module and cruise-peculiar hardware (cruise solar array, cruise ACS, sun sensors, and Canopus sensors) outweighed these advantages.

Another alternative studied was an attitude stabilized support module. The sun and Canopus sensors and the cruise attitude control system were available for this function but additional control electronics and possibly a second inertial measurement unit were required. This system would certainly be more complex and costly than the spin stabilized configuration and is not recommended.

Keeping the cruise solar array with the support module for power after separation from the flight capsule was considered. This concept was rejected after analyzing the size ($\sim$60 ft^2) and physical configuration of the cruise solar array required to support the spacecraft during the trans-Mars mission phase. To make this configuration rigid enough to be compatible with spin stabilization requirements was deemed impractical.

The final trade conducted concerned the cruise communications function. Because both the support module and the lander have an S-band transmitter, either might be used to transmit data during this phase. The 1973 mission geometry dictates omnidirectional coverage. Two hemispherical coverage antennas, one located on the support module and one on the lander, best satisfy this requirement. A single command receiver and detector located in the lander can satisfy the command function. The decision to use the support module transmitter for the entire cruise phase was based on avoiding the thermal control complexity required to use the lander transmitter for a portion of the mission.

3. CONCLUSIONS

The preferred configuration described in this document is a practical means of achieving the technical and program objectives as outlined in the contract statement of work.

The prime objective of minimizing cost by using proven hardware and technology is achieved in the following areas:

1) A modified LM radar is used in the terminal descent and landing system;

2) An engine being qualified by Walter Kidde and Company is modified for use in the vernier propulsion system in conjunction with a modified LTV throttle valve;

3) ACS thrusters being qualified by Rocket Research Corporation are used for midcourse and deflection maneuvers as well as the lander ACS function;

4) A modified Mariner cruise ACS is incorporated;

5) The silver-zinc batteries in the lander are sized so that technology from the Electric Storage Battery development contract is directly applicable;

6) The battery in the support module uses the Mariner '69 cells directly;

7) The guidance and control digital computer is a modified Centaur unit;

8) The telecommunications subsystem incorporates a combination of hardware and technology from the Mariner program;

9) Mariner '69 sun and Canopus sensors are used as built.

Many of these are included at a significant weight penalty. In addition to these items, system simplifications (relative to the configurations studied in the previous mission mode studies) have been incorporated to reduce the cost. The fixed landing legs, fixed lander solar array, common midcourse/deflection/entry ACS thrusters, and common tanks for both monopropellant systems are prime examples.

SUBSYSTEM STUDIES

1. SCIENCE

Activity in design of the science subsystem covered three main areas. These were a study of biology, the candidate instruments, and the effects of other parameters and devices on the biology experiment; acquiring, processing, and distributing a representative soil sample; and the subsurface soil probe with the devices for detecting moisture and measuring temperature fluctuations. Studies in depth were made in soil moisture detection -- the soil probe, soil sample acquisition, and direct biology. Detailed reports covering these studies are included in volume II of this report.

Subsystem Description

The mission objectives and the science required to meet them shown in table 9 were given by Langley Research Center. This table also shows the parameters measured by each instrument in meeting the mission objectives. These instruments and parameters measured are discussed in the following section.

Entry science, ballistic phase.- Instruments carried on the aeroshell include a dynamic pressure sensor and a total temperature sensor. These are mounted on a special plate carried at the apex of the aeroshell. These two instruments, together with the accelerometer triad (at the center of gravity), and the radar altimeter, yield most of the data during the ballistic entry phase to satisfy the mission objective for atmospheric reconstruction.

The aeroshell-mounted instruments are discarded along with the aeroshell at parachute deployment. Figure 24 shows the location of these instruments.

Entry science, terminal descent phase.- During terminal descent there is an opportunity to measure atmospheric quantities by direct measurement. A group of four instruments are used to make the measurements of temperature, pressure, humidity and composition during descent from 11 500 to 5300 ft. These are important measurements because they provide an anchor point for vertical atmosphere reconstruction and these yield the first composition data. The location of these instruments is shown in figure 24. The data taken during entry and terminal descent are transmitted to Earth in real time via the support module.

TABLE 9.- SCIENCE PAYLOAD INSTRUMENTS

		Objectives	Parameters	Instrument
Entry	Ballistic phase	Atmospheric composition and vertical structure	Total density (drag) Total temperature Stagnation pressure	Accelerometer triad Total temperature transducer Stagnation pressure transducer
	Terminal descent phase		Ambient temperature Ambient pressure Ambient humidity Ambient composition	Platinum resistance thermometer Capacitive diaphram transducer Aluminum oxide hygrometer Double focusing mass spectrometer
Surface	Meteorology	Pressure Temperature Humidity Wind	Pressure variations with time Temperature variations with time Humidity variations with time Wind velocity variations with time	Entry instrument used Entry instrument used Aluminum oxide hygrometer Rotating cup anemometer
	Imagery		Surface imaging	Facsimile camera
	Soil Sample for analysis Observation of Biological activity Soil organic composition		Soil sample acquisition Direct biology Soil phrolysis gas analysis	Clamshell scoop on DeHavilland boom To be specified by cusotmer GC-MS/pyrolysis instrument
Sub-surface		Soil temperature Soil moisture	Temperature Moisture	Soil probe containing moisture and temperature transducers

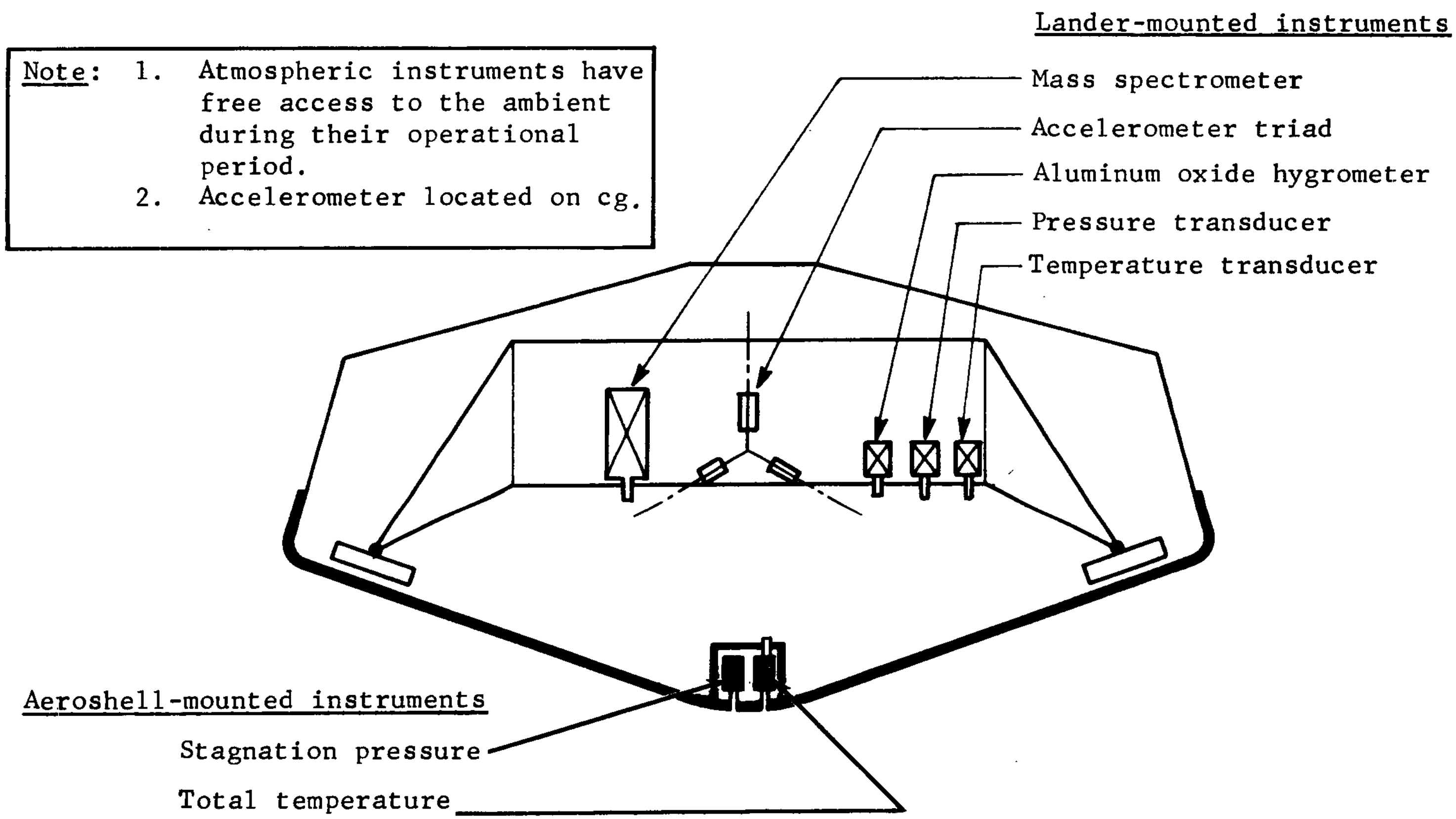


Figure 24.- Entry Science Instruments, Aeroshell and Lander Mounted

Surface science.- The surface science instruments are identified in table 9 and include meteorology instruments (pressure, temperature, wind velocity, and moisture), facsimile camera, gas chromatograph-mass spectrometer/pyrolysis, direct biology, soil sampler, and a subsurface probe. The landed science configuration, showing the meteorology instruments, facsimile camera, soil sampler, and sursurface probe in the deployed position, is shown in figure 25.

After landing, the facsimile camera is extended for surface imaging, and meteorology instruments are deployed on an extending mast. These include a rotating cup anemometer and an aluminum oxide hygrometer. The terminal descent instruments, measuring temperature and pressure, now perform their second function as meteorology instruments. The terminal descent aluminum oxide hygrometer is now used as a calibration comparison instrument and, because it has a lower detection threshold and a subsequent narrow detection range, it serves as a backup within this range.

Immediately after the meteorology instruments are deployed, weather data are sensed and transmitted. The facsimile camera starts taking the first site survey imagery data, a sector 90° in azimuth and 25° in elevation, having a resolution of 0.1°/line. (Other azimuth and elevation combinations are possible.) These imagery data along with the meteorology data are transmitted to Earth in real time via the flyby support module. This transmission period lasts for a minimum of 10 min. Panoramic and high-resolution imagery is programed and returned if the relay link is available. All future data transmissions are direct to Earth.

The sequence of science instrument operation and data transmissions are shown in more detail in the Operational Description subsection.

Science data handling.- Figure 26 is a function block diagram of the science subsystem showing the science data automation unit (DAU). The science instruments are categorized by entry instruments -- those that acquire data during the ballistic phase of the entry period; by lander mounted terminal descent instruments -- those that acquire data during parachute descent; and by the lander-mounted surface science instruments.

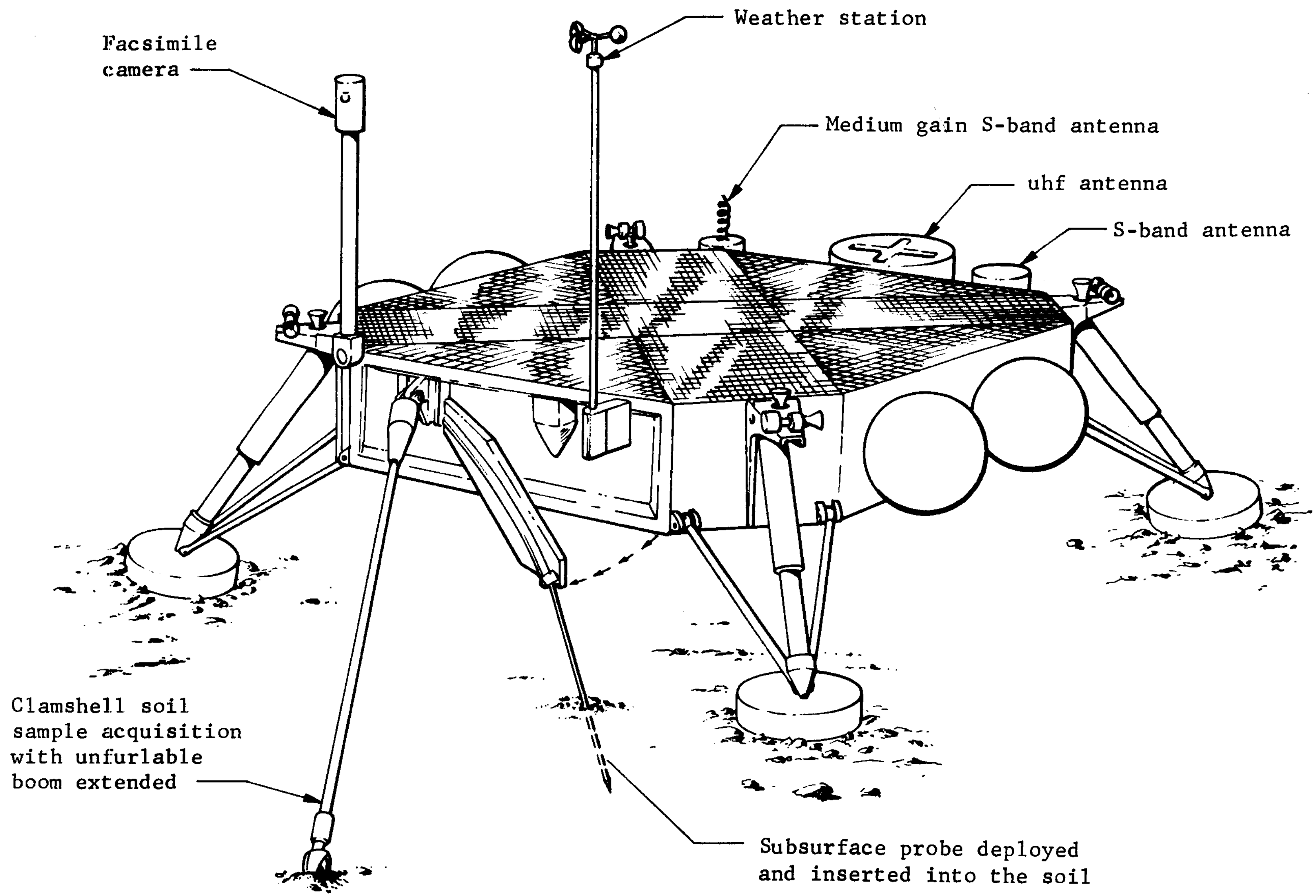


Figure 25.- Landed Configuration with Science Instruments Deployed

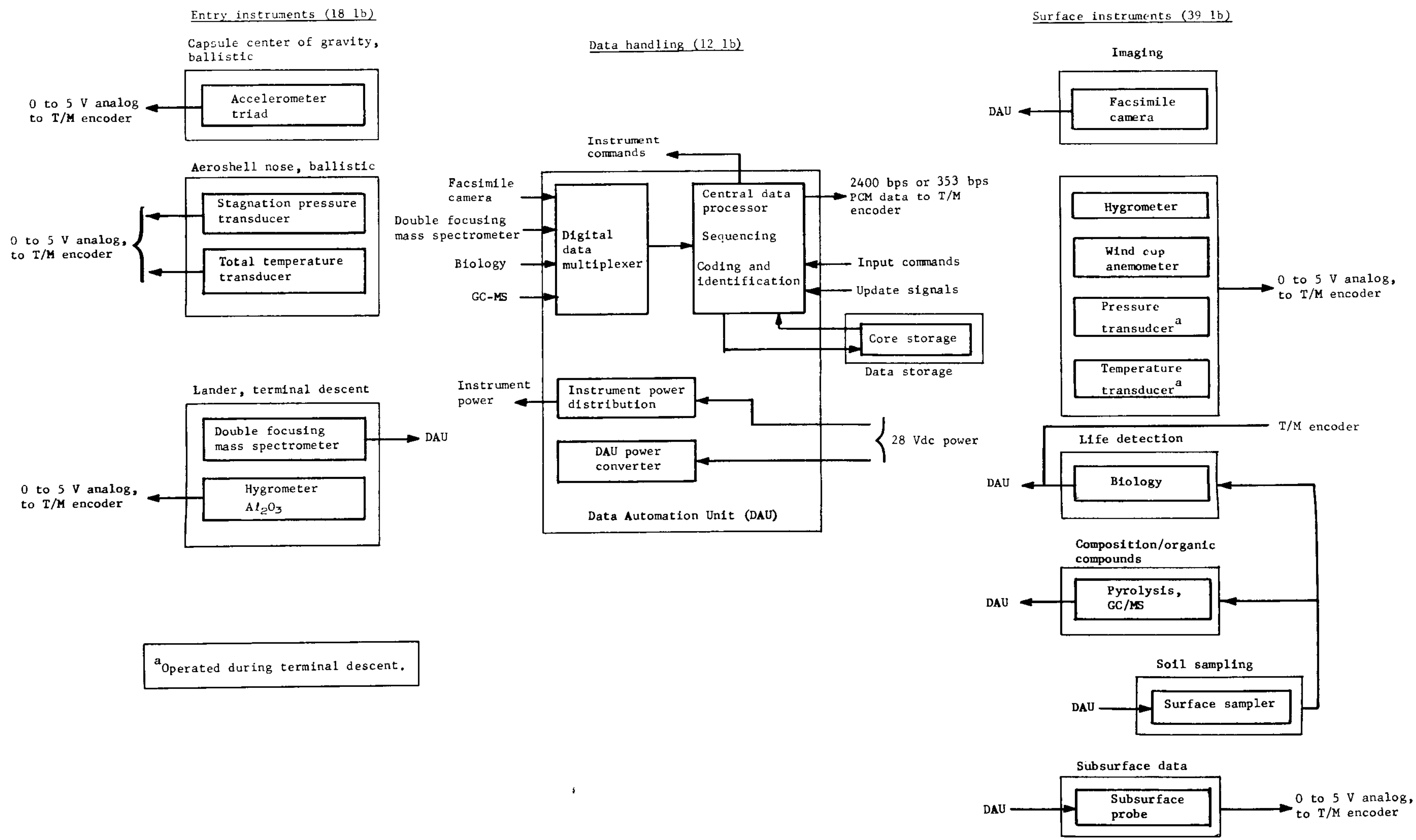


Figure 26.- Science Subsystem

Data from those instruments that have 0 to 5 V analog output signals are not multiplexed by the DAU, but are multiplexed and stored directly by the telemetry data encoder. The instruments included in this category are the accelerometer triad, stagnation pressure transducer, total temperature transducer, and the $A\ell_2O_3$ hygrometer entry instruments. The surface instruments in this category are the meteorology instruments that include the pressure, temperature, hygrometer, and the anemometer, and the subsurface probe instruments that include moisture and temperature data. Biology data will also be sampled by the telemetry encoder after the first three-day period, when the DAU is turned off.

The remaining science instruments, including the two mass spectrometers, gas chromatograph, pyrolysis, biology, soil sampler, subsurface probe, and facsimile camera, which require sequencing commands and updated operational commands from Earth, interface with the DAU. The update commands to the DAU will be received from the command decoder. The DAU will provide power distribution to those instruments that it operationally sequences. Science data that are sequenced from science instruments by the DAU are stored in a static storage unit for later transmission to Earth. This is true of all instruments except for the facsimile camera, from which data are sequenced and transmitted in real time during the data transmission periods.

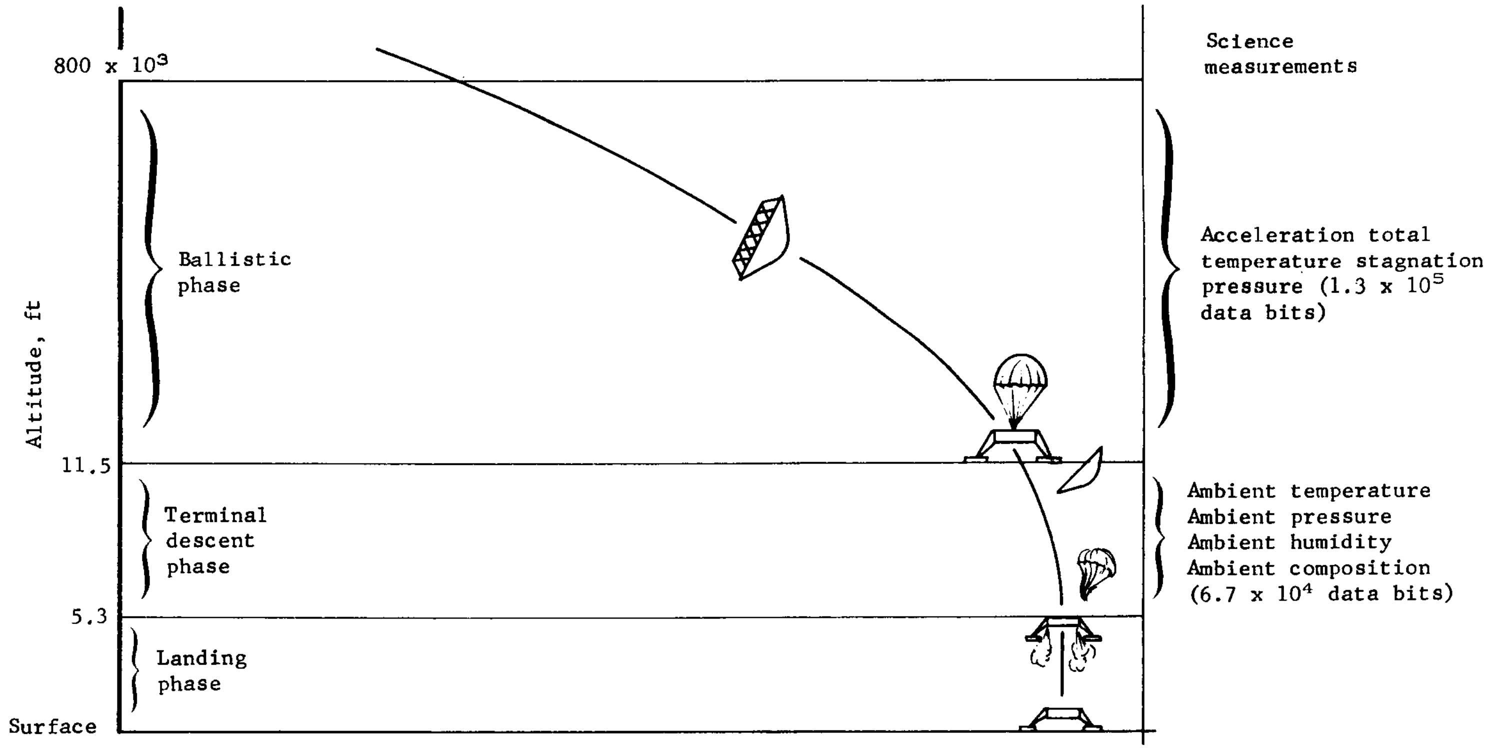
Table 10 is a detailed weight statement for the complete science subsystem.

TABLE 10.- SCIENCE WEIGHT AND POWER

Instrument	Weight, lb	Power, W
Entry science, aeroshell mounted		
Stagnation pressure	3.0	3.0
Total temperature	1.0	1.0
Cabling and packaging	4.0	
Total	8.0	
Entry science, lander mounted		
Accelerometer triad	4.0	4.0
Hygrometer	2.0	1.0
Mass spectrometer	8.0	8.0
Landed science		
Facsimile camera	5.0	10.0
Pyrolysis/GC-MS	16.0	20.0
Biology	8.0	10.0
Soil sampler	2.0	10.0
Subsurface probe	3.0	.5
Hygrometer	1.0	1.0
Atmospheric temperature	1.0	.5
Atmospheric pressure	1.0	1.4
Anemometer	2.0	2.2
Data handling		
Data automation unit	8.0	8.0
Core storage unit	4.0	3.0
Cabling and supports	12.0	
Deployment mechanisms and masts	6.0	
Total	83.0	

Operational Description

Figure 27 identifies the instrument grouping and sequence of the entry instrument operation, together with the gross data bit collection by entry phases. Entry data are correlated with altitude data from the radar altimeter for purposes of atmospheric reconstruction.



Note: Data returned via the support module before landing.

Figure 27.- Entry Science Mission Profile

Figures 28 thru 30 show the surface science instrument sequencing over the first 3-day period. Figure 31 shows the surface science mission profile for day 4 and beyond.

Landed science, day 1 relay transmission.- Immediately after landing, the boom-mounted facsimile camera is deployed 3 ft above the top surface of the lander, and imagery data (90° azimuth x 25° elevation) are taken of the soil sample acquisition area and the subsurface soil probe insertion area. Included in this image is one lander footpad showing an impression of the footprint. Information from this photograph is used to make preliminary soil mechanics determination and serves as the major input on which to base update commands for soil sample acquisition and for soil probe insertion. Panoramic and high-resolution imagery are programed and will be returned if the relay link time is available.

The meteorology instruments, consisting of a rotating cup anemometer and the aluminum oxide hygrometer, are deployed on an extending boom to a height of 4 ft above the top surface of the lander. This deployment is performed at the same time as the camera deployment. Meteorology data, including pressure, temperature, wind velocity, and moisture, are immediately taken and transmitted along with the imagery data over the uhf relay link during the initial postlanded period.

Meteorology data are taken at 2-hr intervals after the postlanded transmission period and placed in storage, along with other data, for transmission during the daily direct link transmission period.

About 1 hr after landing, an atmospheric sample is analyzed by the entry mass spectrometer. This is repeated 8 and 16 hr later for a total of three atmospheric samples at the surface spaced symmetrically about one diurnal cycle. The reasons for this spacing are as follows: the interim period gives sufficient time for the mass spectrometer ion pump to thoroughly purge the system from a previous sample; this gives an opportunity to observe diurnal variations, if any; it also affords an opportunity to observe the depletion rate of species associated with the landed vehicle.

Landed science, day 1 direct transmission.- About 16 hr after landing, direct communication lockup from Earth is achieved (see fig. 28). The stored data and real-time facsimile camera imagery data are transmitted. After 1 hr of transmission, the command link lockup is achieved. The soil sample acquisition and the subsurface probe will be preprogramed for operation; however, update deployment commands based on site survey imagery data may be transmitted at this time, if required.

Legend:

1	Postlanding imaging and meteorology data (data transmission, 2400 bps, 10 min min.)
2	Mass spectrometer, atmospheric sample
3	Meteorology data, sampled every 2 hr
4	Transmission of stored science data
5	Transmission of imagery data
6	Update commands
7	Soil sample acquisition
8	Sample processing and distribution
9	Sample pyrolysis and GC-MS operation
10	Direct biology experiment started (continues for 3 days min.)
11	Subsurface probe deployed
12	Meteorology and subsurface probe data
A	Communication link lockup
B	Command link lockup

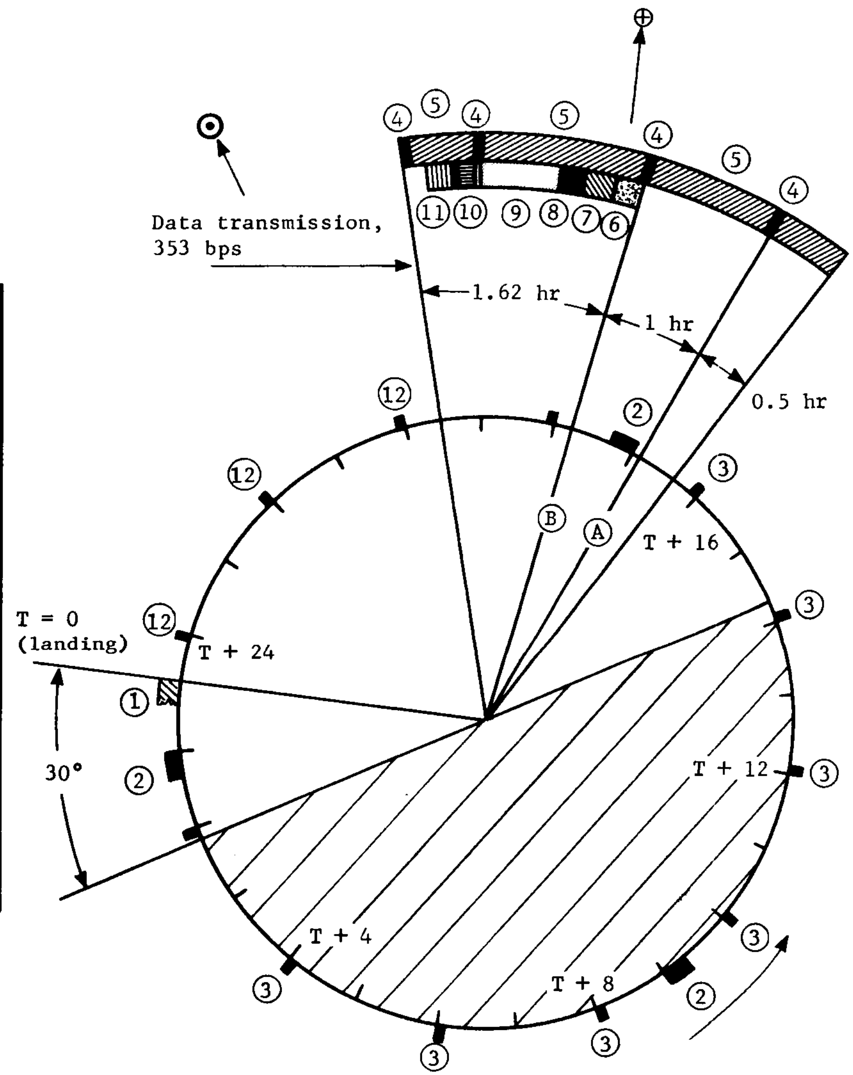


Figure 28.- Surface Science Mission Profile, Day 1

Legend:

① Meteorology, biology, and subsurface probe data

② Pyrolize soil sample 2, perform GC-MS analysis

③ Pyrolize soil sample 3, perform GC-MS analysis

④ Stored science data transmission

⑤ Imagery data transmission

⑥ Update commands

Ⓐ Communication link lockup

Ⓑ Command link lockup

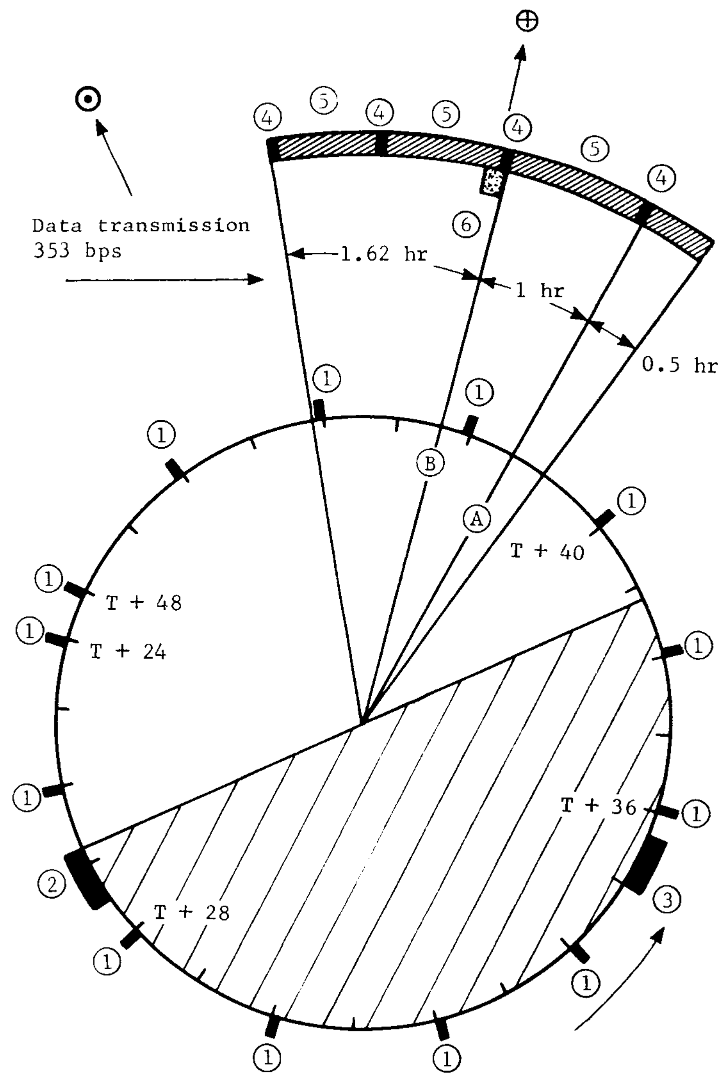


Figure 29.- Surface Science Mission Profile, Day 2

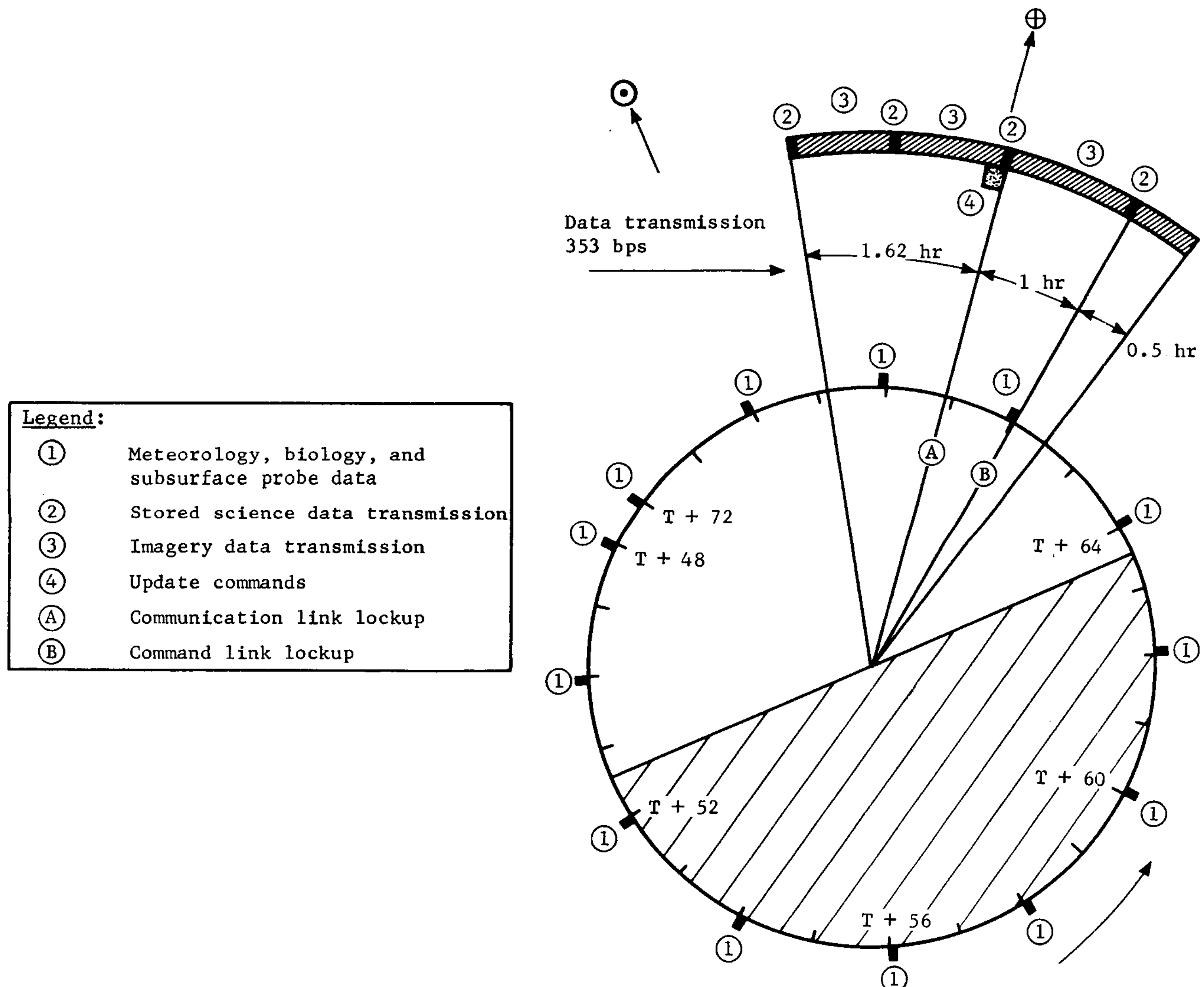


Figure 30.- Surface Science Mission Profile, Day 3

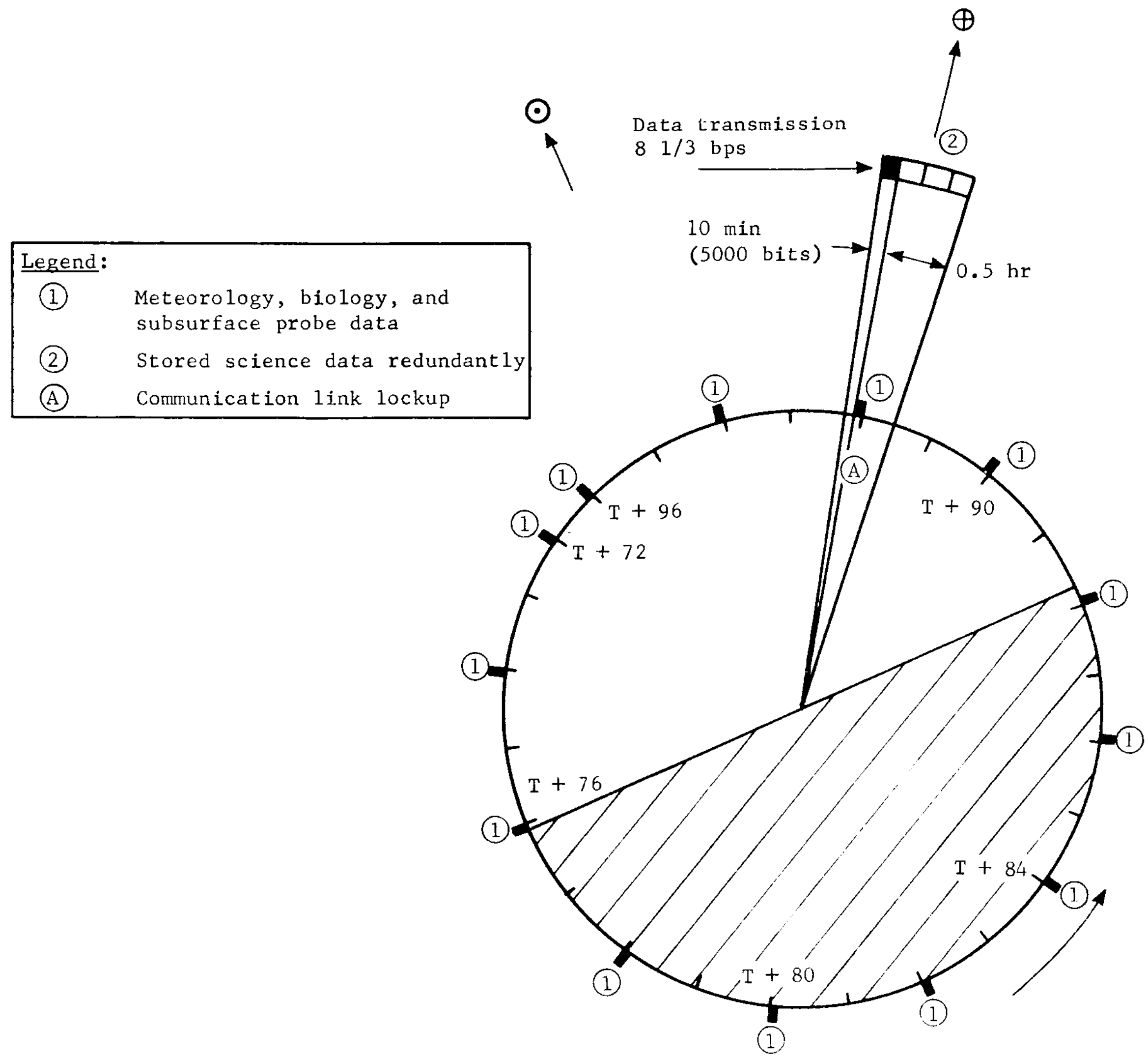


Figure 31.- Surface Science Mission Profile, Day 4 and Beyond

A soil sample is acquired and deposited in the soil processor. This processing consists of three progressively smaller screens, which are vibrating to assist in breaking up loosely bonded large pieces (if such bonding exists), and permits only those particles to enter the distribution system that have passed through the smallest screen. A portion of this vibration energy is transmitted to the sample distribution system giving added assurance of sample flow.

The GC-MS pyrolysis experiment is initiated with the pyrolysis, chromatograph chemical separation, and mass spectrometer mass separation of the no. 1 soil sample.

The biology experiment is initiated and the subsurface probe is deployed to the surface. The first data from the GC-MS subsurface probe and biology experiment are transmitted during this data period.

Landed science, day 2 direct transmission.- About 42 hr after landing, direct communication lockup from Earth is achieved (see fig. 29). The stored data collected during the 22-hr period consists of meteorology and subsurface moisture and temperature data taken every 2 hr; and direct biology data and the gas chromatograph and mass spectrometer data from pyrolyzed soil sample no. 2 and 3 taken at 8-hr intervals. An opportunity for update commands is available; these are transmitted as determined during the actual mission. Analysis of earlier photographs may show an area of interest and an update will make it possible to rephotograph or photograph certain sections in high resolution.

Landed science, day 3 direct transmission.- Day 3, shown in figure 30, is similar to day 2 with the exclusion of the GC-MS data. This instrument has performed its function and is now turned off. Stored meteorology, direct biology, and subsurface probe data are transmitted followed by imagery data in real time.

Landed science, day 4 direct communication.- Day 4, shown in figure 31, is similar to day 3 with the exclusion of imaging data. The data now consist of meteorology, direct biology, and subsurface probe data.

Landed science, day 5 and continuing.- At the end of day 4, the biology is turned off. From day 5 through the rest of the mission (a goal of 90 days), meteorology and subsurface probe data are taken every 2 hr. The facsimile camera can be reactivated if sufficient regenerative power is available. Experiments can also be performed with the soil acquisition device.

Figure 32 is a presentation of an imaging mission and presents a summary of the data bits and surface area imaging patterns that may be covered in a 3-day mission period. Figures 33 and 34 are supporting figures for the imaging mission. As noted, the area of complete surface image coverage extends from a range of 5 m from the camera to the horizon (depending on the surface flatness and lander angles with respect to the local surface). In addition to the panoramic imaging, data capacity is available within the first 3 days for close-in (1 to 2.5 m) site survey imaging and high-resolution, 0.01°/line, site examination images.

The first image to be transmitted after landing (T + 1 min) will be preprogramed and will be a site survey image (90° azimuth by 25° elevation) to evaluate the landing site for subsequent selection of direction of deployment of the subsurface probe and area from which surface soil may be gathered by the soil sampler. A sequence of imaging patterns for the mission will be preprogramed; however, these can be updated if required.

Additional imagery data will be transmitted after the first 3 days, depending on lander power conditions. For data transmission capabilities, see section 5, Telecommunications.

Figure 35 shows a conceptual design of the soil sample acquisition device consisting of a clamshell scoop mounted on the end of an unfurlable DeHaviland type boom. The scoop can be opened and closed by a miniature linear actuator motor. The boom can be provided with two or three degrees of freedom, depending on the weight budget and the priority for sampling flexibility. The conceptual design suggests three degrees of freedom and extension capability up to 8 ft. Greater extension can be provided; however a weight penalty would be imposed.

Figure 36 is a presentation of a subsurface soil probe concept.

Probe insertion: A study has been conducted, evaluating the various parameters associated with a probe to carry temperature sensors and a soil moisture sensor. The study considered probe design, insertion force required, and the deployment and insertion mechanism. Figure 36 shows a conceptual design of the probe deployment and insertion device along with the insertion force. The force required for penetration at various depths was arrived at by conducting a series of laboratory tests in a Scott model soil, and in an ultrasonically compacted soil. These tests show that 1 lb of force will insert a 0.105-in.-diameter probe to a 12-in. depth in the Scott soil.

Data summary	
Imagery	Data bits
Low resolution: 0.1°/line	
Site survey 1	
90° azimuth by 25° elevation (1 to 2.5 m range)	1.4×10^6
Site survey 2	
90° azimuth by 25° elevation (1 to 2.5 m range)	1.4
Panoramic scan 1	
360° azimuth by 10° elevation (10 m to horizon)	2.16
Panoramic scan 2	
360° azimuth by 10° elevation (5 to 10 m range)	2.16
High resolution: 0.01°/line	
High resolution scene 1	
5° azimuth by 5° elevation (1.8 m range)	1.5
High resolution scene 2	
5° azimuth by 5° elevation (1.8 m range)	1.5
Total bits	10.12

Distance from camera, m	Low-resolution 0.1°/line 10 TV lines/deg	High-resolution 0.01°/line 100 TV lines/deg
1.76 at 18° from axis	0.88 cm/optical line pair, best resolution	0.088 cm/optical line pair, best resolution
10	5 cm	0.5 cm
100	0.5 m	5 cm
1000	5 m	0.5 m
	Resolution = $\frac{\text{distance}}{200}$	Resolution = $\frac{\text{distance}}{2000}$

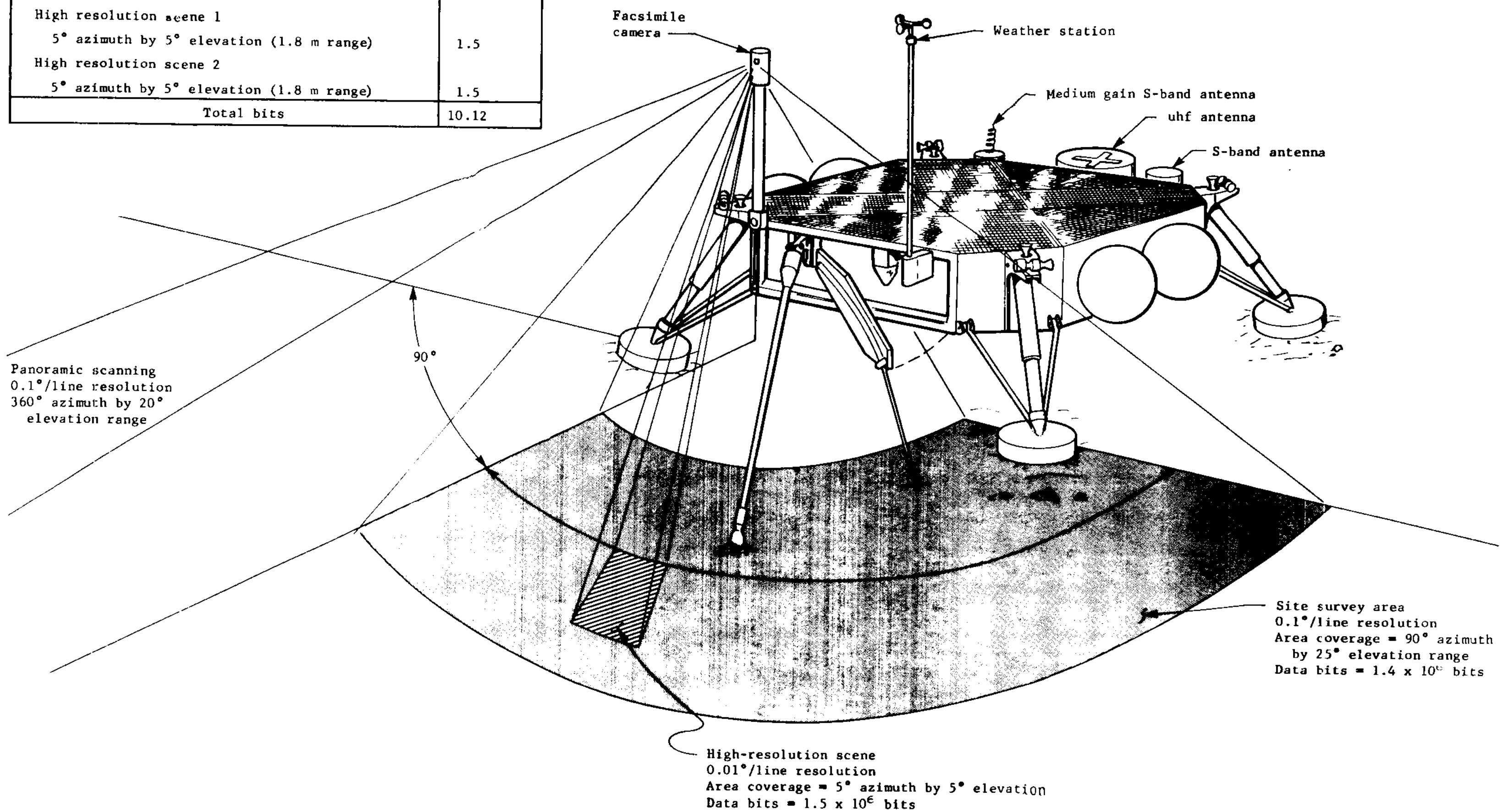


Figure 32.- Surface Imaging, Resolution and Data Bits

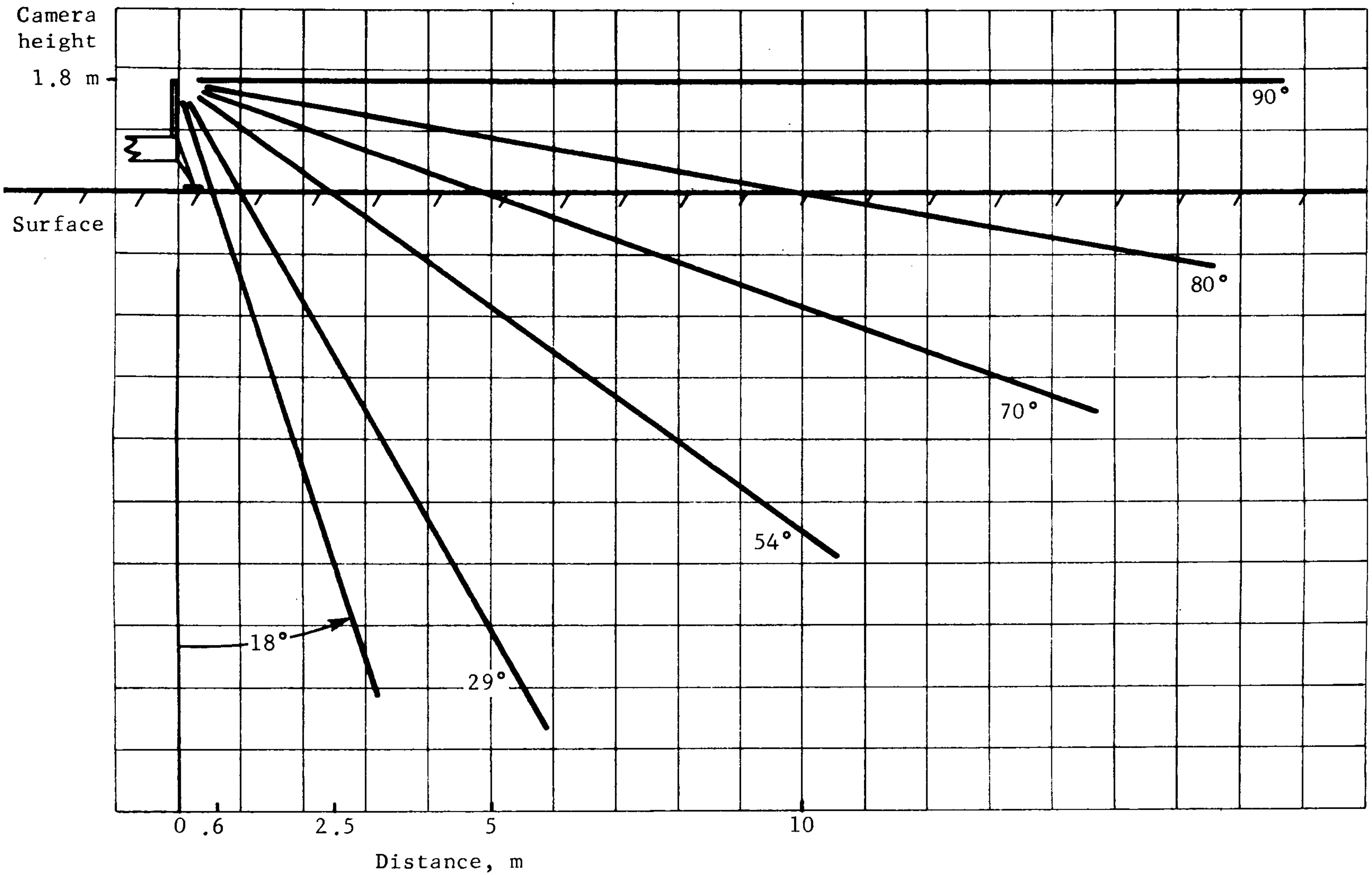


Figure 33.- Camera Elevation Angles vs Range

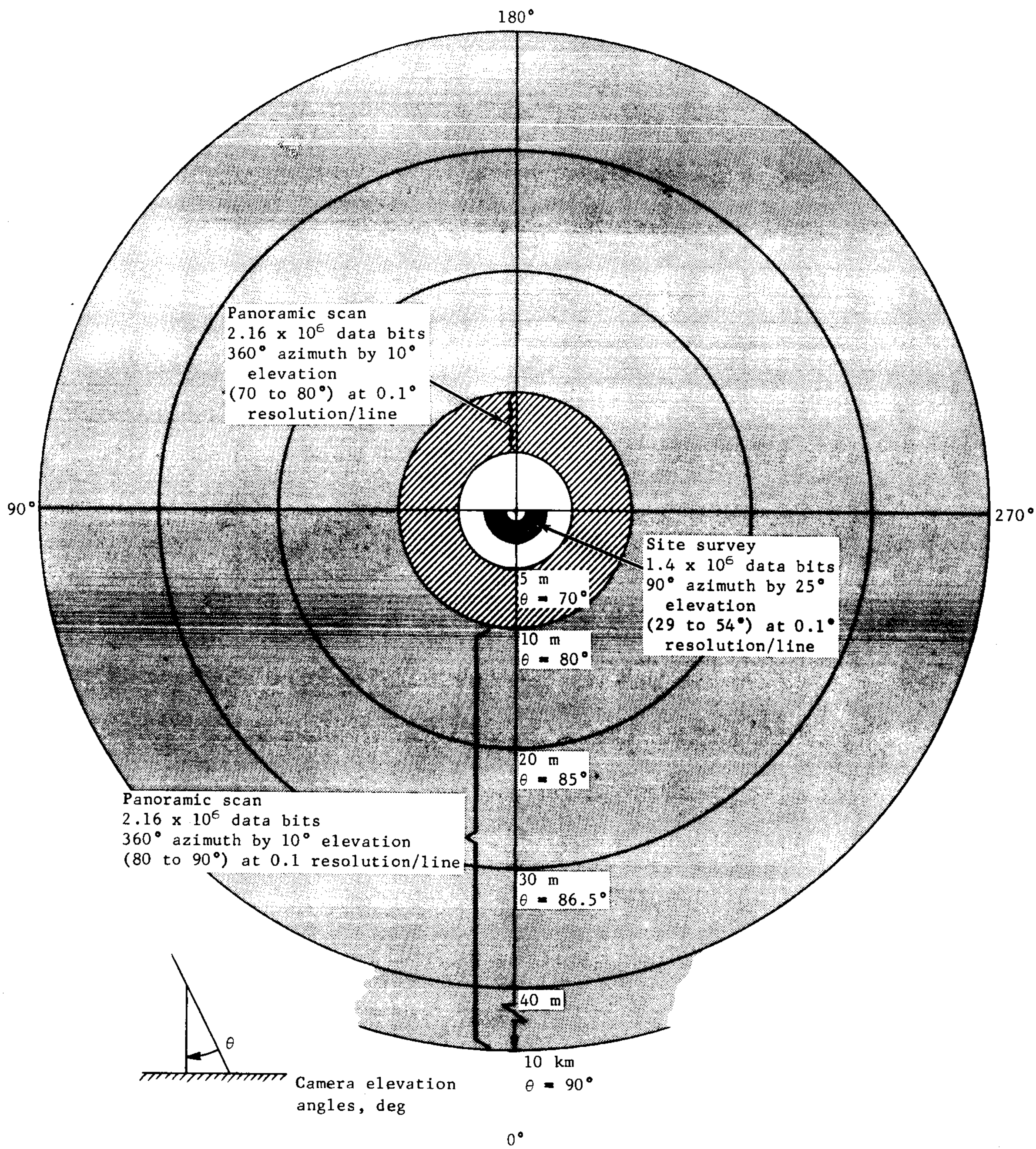
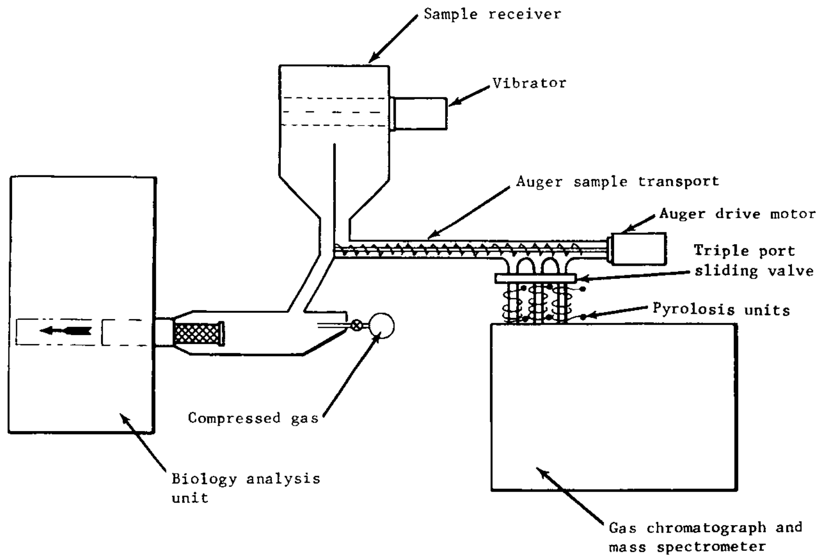
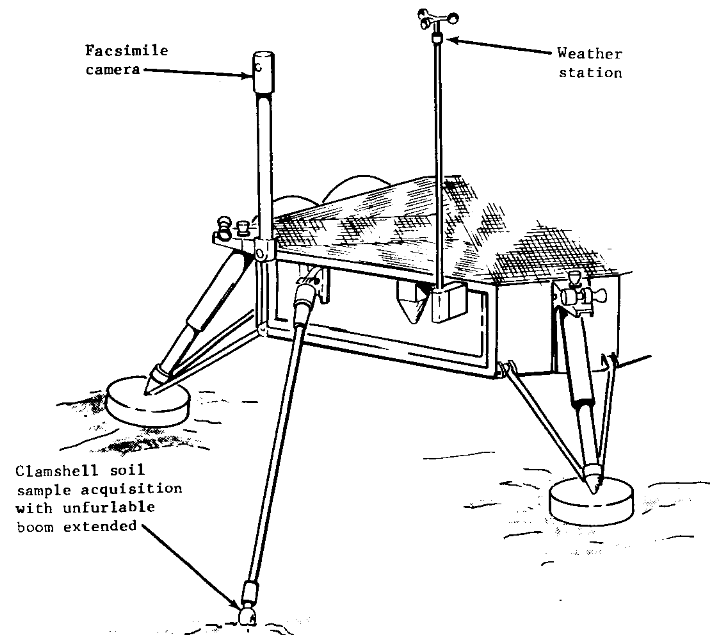


Figure 34.- Surface Imaging Coverage and Data Bits



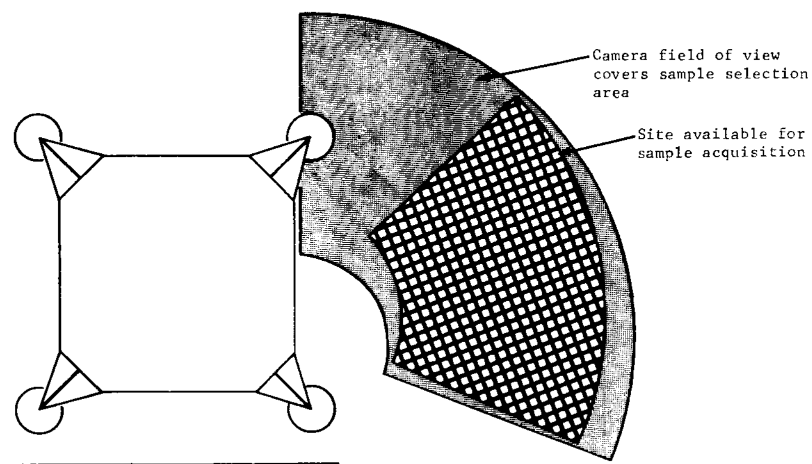
(a) Sample Processing, Distribution and Introduction



(b) Sample Acquisition

1) Take site survey imagery data;
2) Conduct background count;
3) Analyze site survey imagery data;
4) Choose sampling area;
5) Command sample acquisition;
6) Acquire sample;
7) Process sample (pass through three vibrating progressively smaller grading screens. This breaks up the loosely bonded larger pieces and places an upper size limit on the particle size entering the sample distribution system);
8) Distribute and introduce sample,
   a) A portion is fed, by gravity, to the direct biology experiment where it falls on a receiving plate. After a certain accumulation, a jet of compressed gas blows the finer particles into the screened enclosure of the biology receiver by a process called "puff distribution." The receiver is then withdrawn into the biology experiment instrument and the gate closed,
   b) Another portion drops into a transport auger that distributes and compacts the sample into three pyrolyzing units. The triple port sliding valve closes, sealing the sample in the pyrolyzing oven.

(c) Sampling Operational Sequence of Events



Note: Site survey photograph is made and studied before selecting specific sample acquisition area.

(d) Sample Selection

Figure 35.- Soil Sample Acquisition, Processing, and Distribution

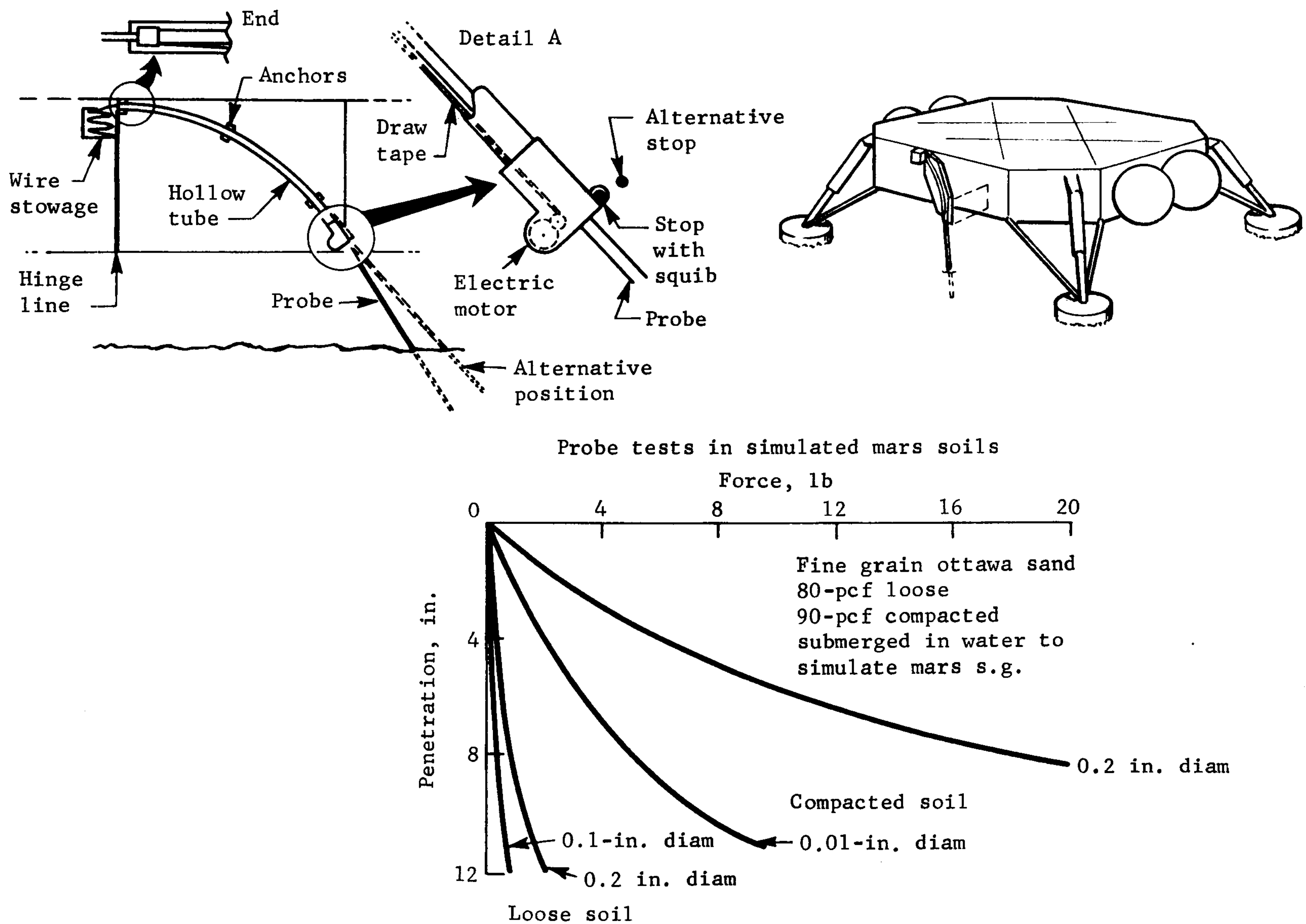
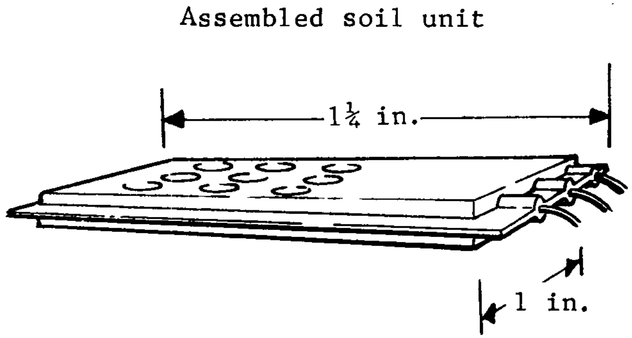


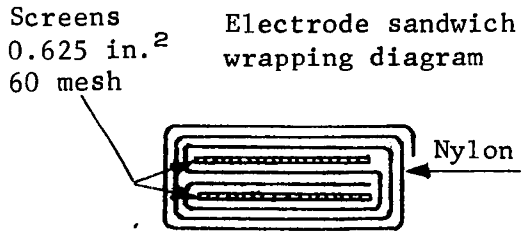
Figure 36.- Soil Probe Concept

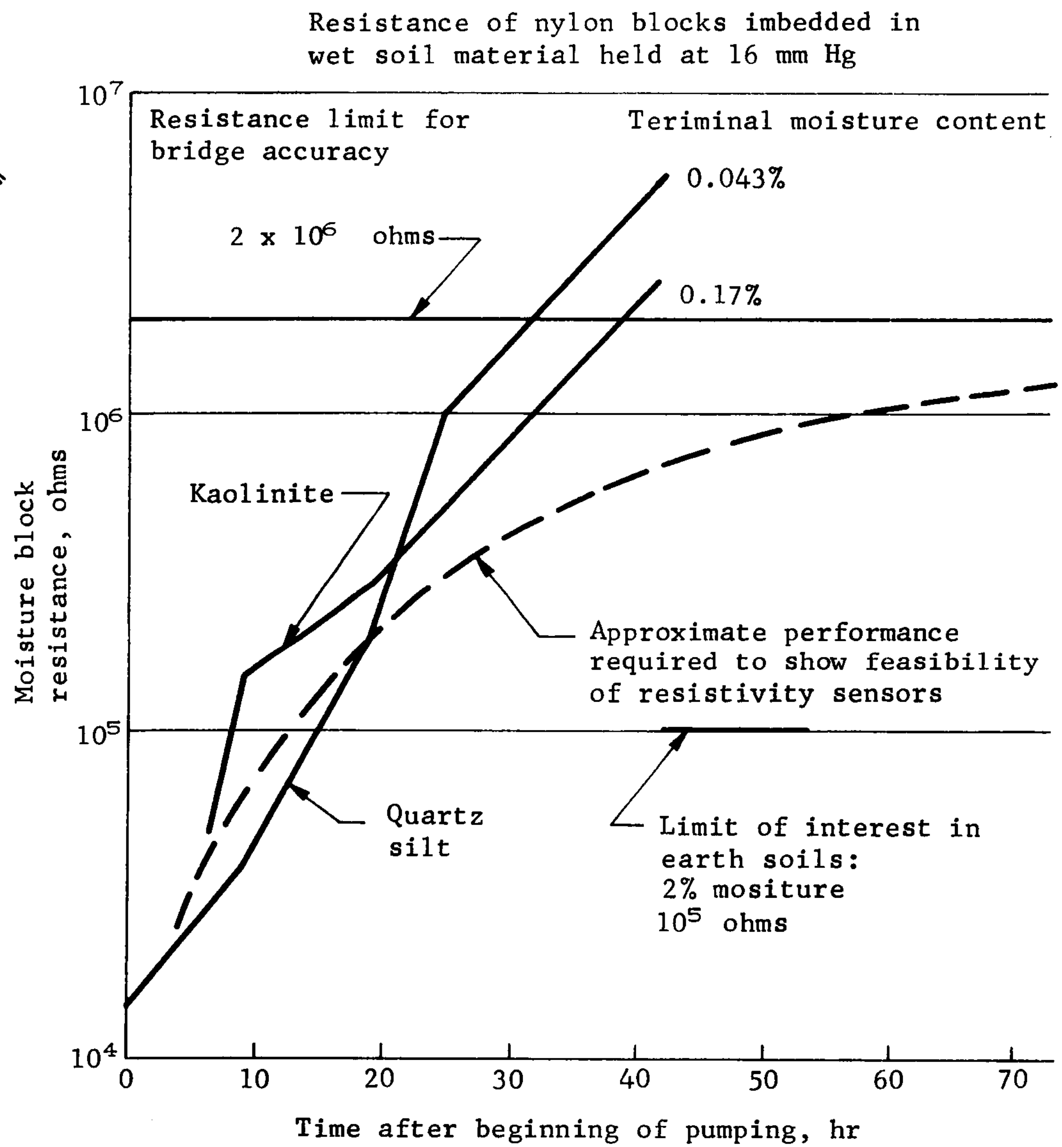
Measuring soil moisture: A study was made of terrestrial soil moisture cells to determine their detection threshold. Most of this effort was concentrated in an evaluation of the nylon block cell produced by Industrial Instruments Company. The cell and some of the test results are seen in figure 37. A more detailed description of these tests is included in volume II. As is seen in figure 37, these moisture cells are unacceptable candidates for detecting the Mars soil moisture.

Soil temperature gradient: Measuring the subsurface soil temperature at two or more depths could yield some very interesting data. In addition to the direct temperature data, these data can yield indirect measures of the soil diffusivity, thermal conductivity, soil moisture, and permafrost. Volume II shows how a study of the diurnal fluctuations can be used to indicate a measure of the soil moisture and the state of the moisture. By placing a small heating element at the tip of the probe and artificially raising the soil temperature about 10 or 20° in the vicinity of the tip, the diffusivity gradient can be observed. This again will yield information on soil moisture and permafrost.

Subsurface gas analysis: The subsurface soil probe presents an opportunity to make a mass spectrometer analysis of the subsurface soil gas. By ventilating the probe to the soil, preferably at the tip, and providing a gas-carrying tube to the mass spectrometer inlet, it is entirely feasible to make a composition measurement of the soil gas. A measure of the biological activity in the soil can be made by comparing this analysis with an atmospheric analysis. Difference in these analyses could be attributed to biological activity in the soil. Measuring the conductance of the soil gas can give a measure of the soil density and grain size calculated from the permeation rate of the gas.







Resistance of sensors alone wet for 2 days at 20 mm Hg			
Block no.	½ hr	1 hr	2.5 hr
1	190 ohms	190	>5 500 000
2	210	5500	>5 500 000

Figure 37.- Test of Soil Moisture Sensors

2. STRUCTURES AND MECHANISMS

The structure and mechanisms subsystem consists of the canister/adapter, aerodecelerator, lander, aeroshell, cruise module, and support module.

Canister/Adapter

The canister/adapter (fig. 21), which encloses approximately half of the sterilized hardware, is an aluminum shell structure combined with an adapter truss that attaches the spacecraft to the booster. At its maximum diameter, the canister mates with and is sealed to the closure frame of the cruise module. The seal is a large-diameter elastomeric band that is stretched over the canister and cruise module closure frames to effect a seal. Internal pressure is limited to 1 psi or less during launch by venting through biovents. A short cone frustum inside the canister skin is fitted with longerons and provides a load path from the lander structure through the canister skin. The adapter truss transmits launch loads through the canister skin from the longerons to the booster transtage payload support fittings.

Aerodecelerator

The aerodecelerator assembly (fig. 21) consists of the parachute mortar support structure and the base cover.

The parachute mortar support structure is a cylinder that is attached to the lander by four radial beams. At the tip of each beam is a fitting that mates the lander with the canister/adapter. Attachments are made by bolts and separation nuts. Access to these fittings is through access holes in the canister/adapter skin.

The base cover consists of two cone frustums of different angles joined in tandem. At the large-diameter end, the base cover seats against the aeroshell base ring. At the small diameter end, the base cover is attached to the parachute mortar support structure. Fiberglas reinforced composite structure is currently being considered for fabricating the base cover.

Lander

The lander (fig. 21) is composed of aluminum/magnesium body assembly, aluminum landing legs, and miscellaneous brackets.

The lander body is square in planform with the corners truncated to produce 16-in. flats that form a base for landing leg attachment. The structure consists of two 6-in. deep eight-sided frames at the upper and lower levels of the 18-in. deep body structure. Vertical corner fittings span the 18-in. height joining the upper and lower level frames, providing landing leg attachment fittings, and supports for the four vernier engines and support for the survivable equipment pallet. Transverse shear loads are carried by sheet metal webs between the upper and lower level frames.

Survivable equipment is packaged on a pallet (46 in. square with clipped corners), which is attached 3 in. above the lower level frame to the vertical corner fittings. The equipment is enclosed inside 3 in. of thermal insulation. Approximately 12 ft^3 of survivable equipment compartment volume is provided.

The landing legs consist of a main strut, a bipod, and a foot pad assembly. The main strut contains a liquid spring shock attenuator similar to the type used on Surveyor. The bipod is a fixed structure having no energy-absorbing capability. The foot pad assembly is equipped with crushable energy-absorbing material. The lateral shear strength of the foot pad is designed to limit the bipod loads in the event of high friction forces between the Mars surface and the foot pad. The landing legs are attached to the lander in a fixed position and, therefore, require no deployment.

Items requiring mechanical design are the medium gain S-band antenna, science experiments, and the isotope heaters. The S-band antenna requires erection to the local gravity vector.

The science instruments requiring mechanization are the facsimile camera, the weather sensor mast, the soil sampler, the sample distribution system, and the subsurface probe (see fig. 21, view B-B).

During transit and landing, the facsimile camera is stowed against the lander body upper level frame. The camera is attached to the frame by a simple pivot fitting that is spring driven 90° to the erected position when released. The camera is located so that it can survey the area of operation of the soil sampler and the subsurface probe.

The weather mast is a "jack-in-the-box" type of tape mast. This device requires release of a restraining cover to allow deployment.

The soil sampler is an extendible and retractable tape mast equipped with articulated spoons at the tip and with a two-axis drive at the base. The soil sampler places a sample in the distribution system, which delivers small samples to the direct biology experiment and the pyrolysis unit.

The subsurface probe is a small diameter tube (approximately 0.10-in. diameter), which is inserted to a depth of about 1 ft into the soil. The 0.10-in. insertion tube is supported inside a larger diameter tube that is attached to the lander via a deployment arm.

One of the four 50-W isotope heaters is mounted at each corner of the survivable equipment compartment, near the landing leg. During cruise, these heaters must be external to the survivable compartment to allow the heat to be rejected to space. After landing, the heaters are positioned either in or out of the compartment depending on the internal temperature.

Aeroshell

The aeroshell (fig. 21) is an aluminum skin/aluminum frame shell 11 ft in diameter.

The lander adapter structure is a part of the aeroshell assembly and is composed of forward and aft frames and a 56-in.-diameter stiffened cylinder. Attachment to the lander is by four bolts and separation nuts.

Attachment of the cruise module to the aeroshell is by eight bolts that pass through the ablator.

Cruise Module

This assembly (see fig. 22) consists of an aluminum skin, longerons, ring frames, support module adapter, separation system and brackets for attachment of solar panels, sun sensors, Canopus sensor, cruise ACS, and thermal insulation.

The skin, which forms half the sterilization canister frustum, is a cone with a spherical segment cap and a cylindrical 136-in.-diameter section 12-in. long.

To react the support module inertial loads during launch, stiffeners are added to the skin of the cruise module to prevent buckling. A cylindrical aluminum adapter is provided at the apex to attach the support module. The cruise module is attached to the aeroshell at eight points on a 124-in.-diameter circle.

Springs to eject the cruise module from the aeroshell are located in the support module adapter. The springs and seats remain with the cruise module after separation.

Support Module

The support module (fig. 23) consists of an aluminum cylinder and disc shaped floor. A 26-in.-diameter, 8-in.-long cylinder forms the body and also serves as the radiator for thermal control. The cylinder is equipped with ring frames at each end to attach the S-band high-gain antenna and to mate with the cruise module.

All equipment is mounted on the disc shaped floor fabricated of aluminum alloy, approximately 0.125 in. effective thickness. The floor transmits component loads to the cylindrical body and also serves to conduct heat from the equipment to the cylindrical body, which acts as a radiator.

3. PROPULSION

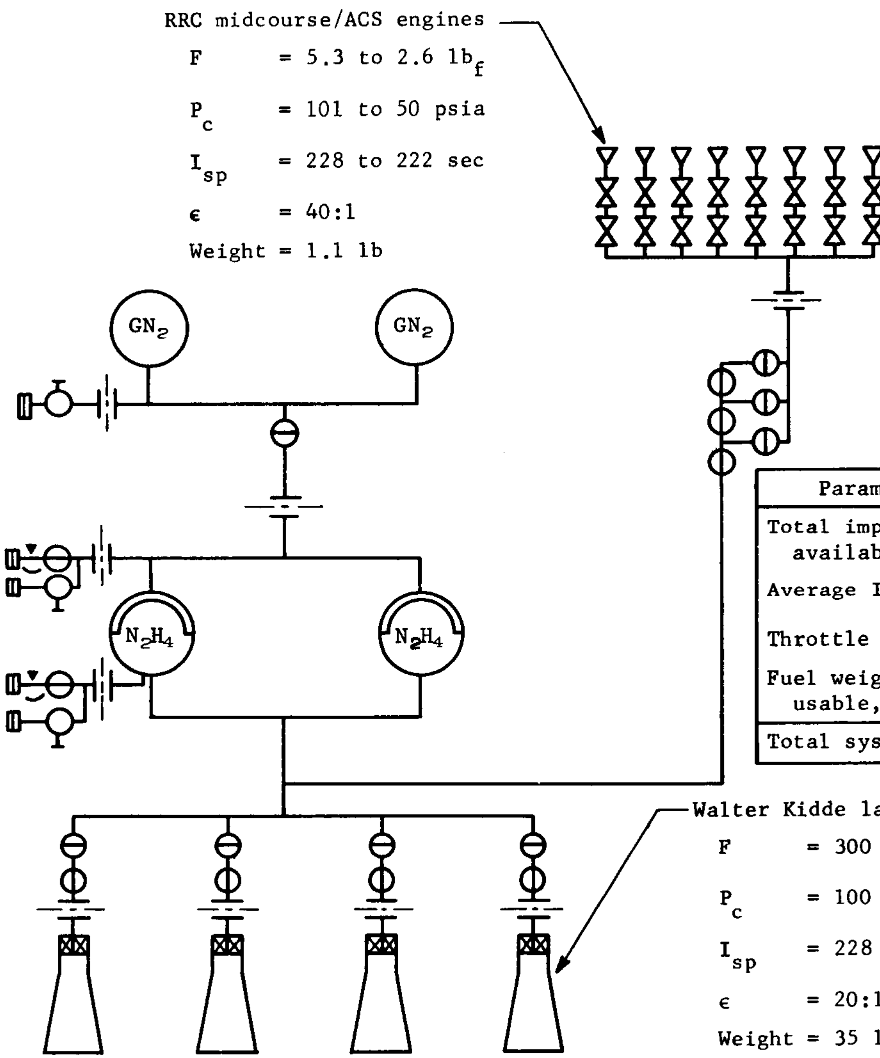
There are three separate propulsion subsystems on the preferred Mars Lander configuration. As shown in figure 38, the landing system also includes the propellant, plumbing, and engines associated with the midcourse, deflection, and entry attitude control maneuvers. The principal design choices made in the determination of this system are:

1) A monopropellant system was selected in lieu of a bipropellant system for design simplicity;
2) The landing and maneuver engines are developed and will be qualified by Mar. 1969;
3) Blowdown pressurization eliminates the reliability problems associated with regulators;
4) Nitrogen gas pressurant avoids the helium leakage problems;
5) Sterile propellant loading eliminates the operational problems associated with terminal sterilization;
6) Developed, qualified components (i.e., ordnance valves, filters, hand valves) were selected wherever possible;
7) Ethylene propylene (EP) bladders are more compatible with hydrazine than are butyl bladders;
8) Common tankage was selected to simplify packaging.

The cruise mode attitude control system is shown in figure 39. The principal features of this system are:

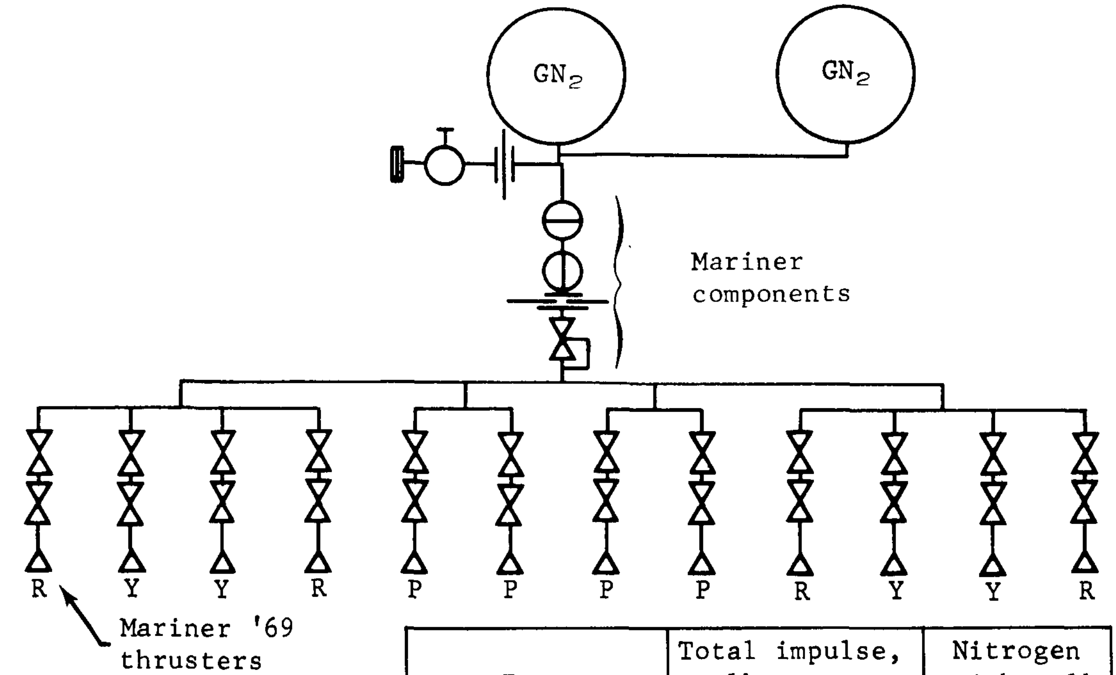
1) The propellant is gaseous nitrogen;
2) The critical components (ordnance valves, regulator, and thruster valves) are unmodified Mariner '69 components;
3) Series thruster valves are included to protect against the valve open failure mode.

The support module solid spin rocket is shown in figure 40. As in the case of the other engines and critical components, these motors are developed and qualified.



Parameter	Midcourse	Deflection	ACS	Landing
Total impulse available, sec	7277	14 445	488	36 180
Average I_{sp}, sec	226	225	200	225
Throttle Ratio	--	--	--	3:1
Fuel weight usable, lb	32.2	64.4	2.4	160.8
Total system weight, lb	521.4			

Figure 38.- Landing/Midcourse/Entry ACS Propulsion System



Thrust

Pitch, yaw = 0.006 lb
Roll = 0.0085 lb

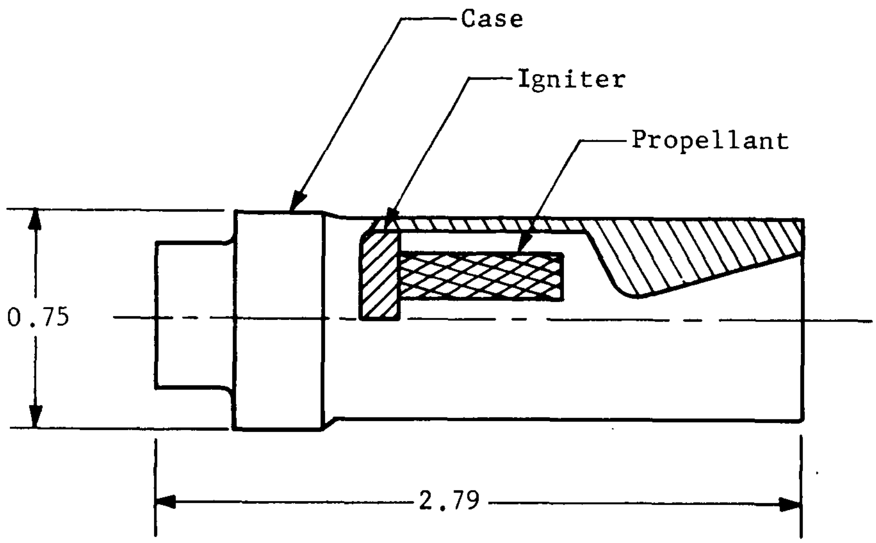
I_{sp}

Pulse mode = 35 sec
Steady state = 60 sec

Inlet pressure = 15 psia

Item	Total impulse, lb-sec	Nitrogen weight, lb
Maneuvers	109	1.82
Attitude hold	176	5.03
Total required	285	6.85
Margin	285	6.85
Residue	--	.3
Total	570	14.0

Figure 39.- Cruise Attitude Control System



Manufacturer	Atlantic Research Corporation
Designation	0.3-KS-5.0
Application	Tiros
Status	Qualified 1960
Average thrust, lb	5.0
Burntime, sec	.3
Total impulse, lb-sec	1.5
Weight, lb	.085

Figure 40.- Support Module Spin Rocket

Subsystem Studies

The preferred design is based on a series of analyses and surveys to determine the availability and applicability of critical long-lead components.

The landing engine selected for the terminal descent phase is manufactured by Walter Kidde and Company and is currently under development for an Air Force program. When compared to a new development, this engine represents a program cost savings of approximately $3.5 million. The only required modifications to this engine are a reduction in expansion ratio to be compatible with the Mars atmosphere and the addition of a throttle valve to provide the thrust variation required for the lander.

The engine throttle valve was also selected on the basis of availability and compatibility. The preferred valve design is an LTV Electrosystems, Inc., Minuteman III liquid injection thrust vector control valve in which the pintle and seat are modified to be compatible with the landing engine pressure schedule and flow rate.

The engines that perform the midcourse, deflection, and entry attitude control maneuvers are similar to the landing engines in that they are also currently under development for the same program as the landing engine and are capable of satisfying the mission requirements. There are no modifications of the thrust chamber necessary to satisfy the lander requirements. However, the thrust chamber valve will have to be requalified to comply with the requirement of terminal heat sterilization.

Mariner '69 hardware was studied for its applicability to the preferred propulsion subsystem design. On the basis of these studies, the regulator, thruster valves, and ordnance valves were selected. As in the case of Mariner, the cold gas nozzles will be tailored to the spacecraft. With the preferred concept of an unsterilized support module, no modifications of the Mariner hardware are required for the Mars lander.

Blowdown pressurization was selected for the lander as a result of previous studies (vol. VI of ref. 1 and ref. 7) and due to the fact that the landing and midcourse engines will be qualified for a blowdown system.

Conclusions

The significant conclusion for the propulsion subsystems are:

1) The preferred designs for the propulsion subsystems minimize program development risk and cost;

2) Critical components are available with minor modifications.

4. GUIDANCE AND CONTROL

The guidance and control subsystem consists of those sensors and associated electronics required to control the attitude of the spacecraft during the coast phase, to perform midcourse and deflection maneuvers, and to provide attitude control during the entry phase as illustrated in figure 41. It also provides the required reference for controlling the capsule during the terminal descent and landing phase.

Stellar sensors provide an attitude reference during the cruise phase. The error signals from these sensors are fed to the cruise attitude control system (ACS). During midcourse and deflection maneuvers, the inertial measurement unit (IMU) provides the error signals required for attitude control and is initialized from the stellar sensors.

After separation from the capsule, the support module is spun up. Coning will be minimized through the use of a nutation damper.

Before separation, the inertial measurement unit is initialized and provides the attitude reference on in through entry. The stellar sensors are jettisoned as a part of the cruise module. At entry, the capsule entry ACS is switched to a rate damping mode in pitch and yaw. Roll attitude hold is maintained for antenna pointing. At a preset altitude measured by the high-altitude altimeter (AMR-1), the parachute sequence is initiated and the aeroshell is jettisoned.

The low-altitude altimeter (AMR-2) and the landing radar (LR) are then turned on. At a preset altitude, the vernier engines are ignited and the precomputed velocity vs range descent contour is followed. During the descent, the LM radar is used as the primary sensor provided all beams are locked. Otherwise, the inertial navigator is used as the primary sensor.

The preferred approach was strongly shaped by two factors -- the requirement to investigate the use of existing equipment such as Mariner 69 sensors and the LM radar, and the sterilization requirement.

The guidance and control equipment configuration selected for the soft lander/support module will provide the required functions for the 1973 Mars lander provided that development of the IMU and the LR is initiated during the Phase C study.

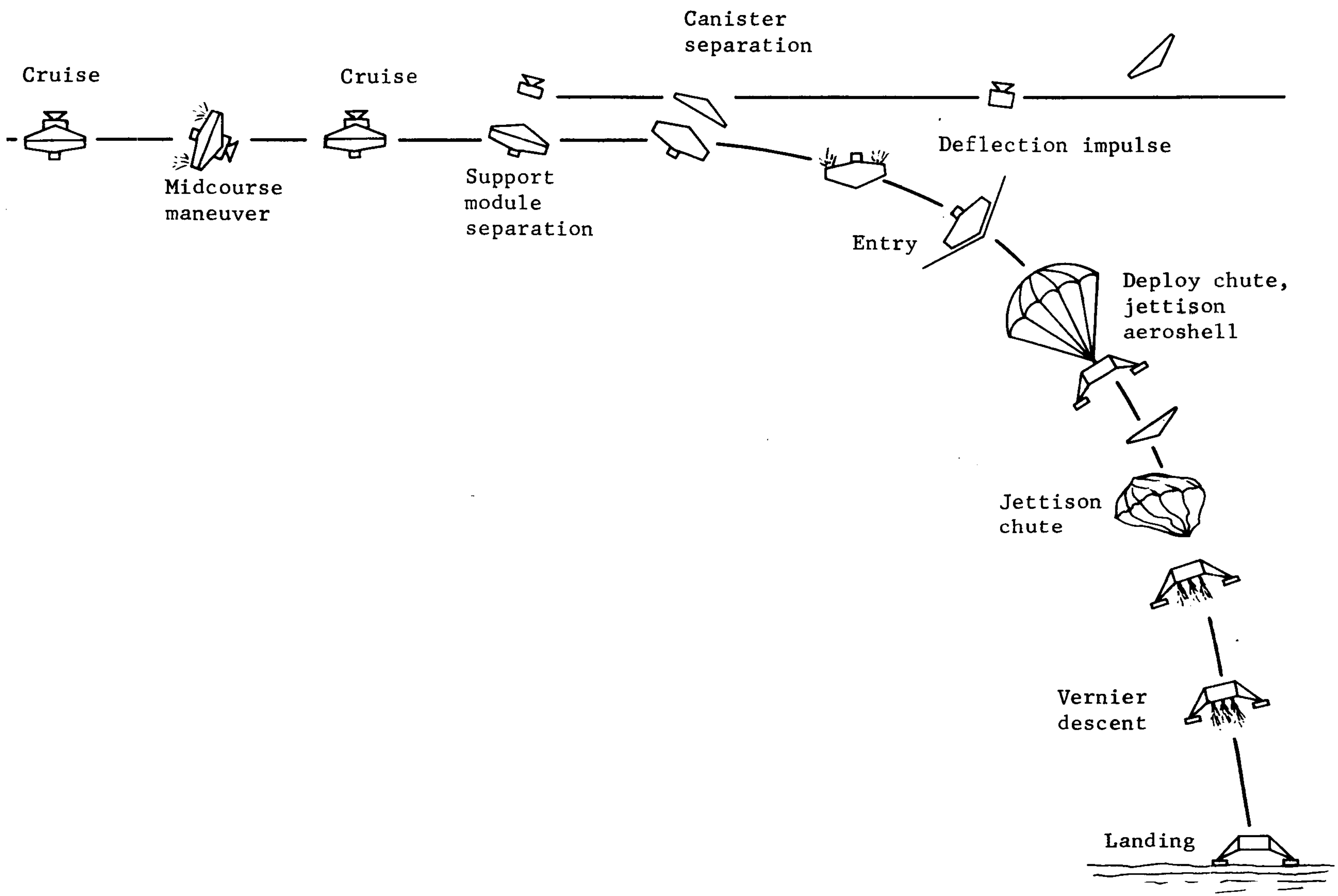


Figure 41.- Guidance and Control System Sequence

The preferred configuration is as shown in figure 42.

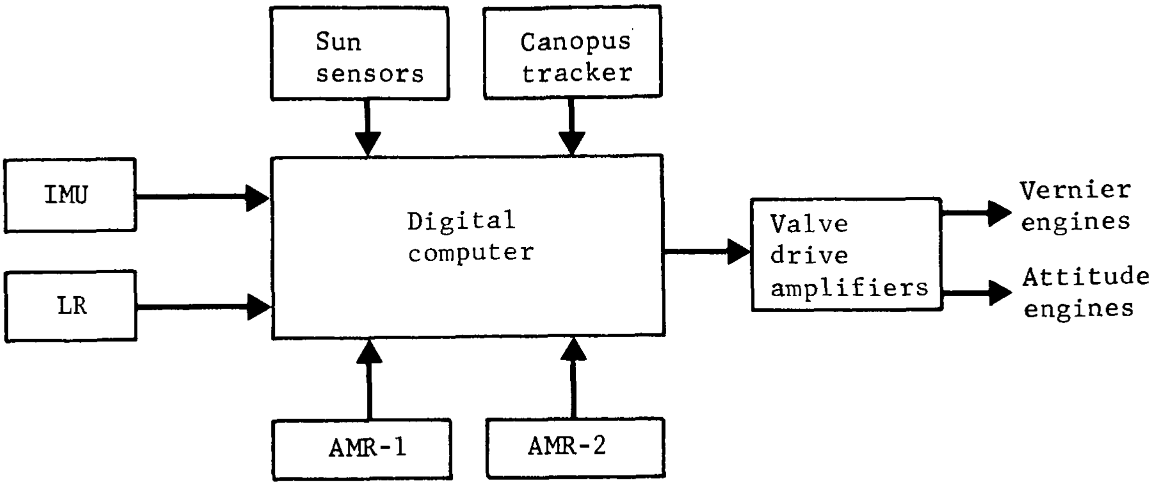


Figure 42.- Guidance and Control Subsystem Block Diagram

The IMU provides attitude and velocity data for attitude control during the midcourse corrections, deflection maneuver, and entry through landing phases as illustrated in figure 41. The IMU also provides a backup to the TDLR and is the primary sensor during the vertical descent phase, i.e., 100 ft to landing. The IMU contains three gyros and three pendulous accelerometers in a strapdown configuration with sensor input axes coincident with vehicle pitch, yaw, and roll axes.

The LR selected during this study is a modified LM radar. The radar measures velocity and slant range of the capsule with respect to the planet surface by a three-beam doppler velocity sensor and a single beam altimeter. The LM radar consists of an antenna assembly and an electronics assembly. The required LM radar modifications and mission ramifications are listed below and are based on the simulations and results outlined in appendix A:

1) Radar modifications,

   a) Relocate the range beam to the antenna pattern center,

   b) Reduce the acquisition search time to 3 sec,

c) Change the mode switch altitude to 1250 ft,

d) Delete the LM coordinate transformation,

e) Add three counter registers or convert all channels to analog format,

f) Change 51 of 124 materials and 24 of 4970 parts;

2) Mission modifications,

a) Constrain parachute size and deployment altitude,

b) Use radar altimeter to initiate terminal descent phase,

c) Use inertial system and radar altimeter for LM radar backup,

d) Use inertial system for vertical letdown.

The high-altitude radar altimeter (ARM-1) is required for parachute deployment as well as for atmospheric reconstruction.

The low-altitude radar altimeter (ARM-2) is required to guarantee accurate vernier ignition at the correct altitude and also for initialization and updating of the inertial navigation computations.

The sun sensors consist of six sensor assemblies that provide error signals to the cruise attitude control system to attain sun acquisition and to maintain sun lock after acquisition during the cruise phase. These sensors are the same as those proposed for Mariner 69 and provide a 4π sterad field of view.

The Canopus tracker is the position sensor in the roll attitude control loop used during the cruise phase. It is identical to the Mariner 69 tracker.

The digital computer is required to convert sensor outputs into commands to the active attitude control and propulsion subsystems and to generate time-and event-dependent discrete signals. It is a general-purpose 4000 (expandable to 8000) 18-bit word machine with a 4-μsec memory cycle time.

The valve drive amplifiers (VDA) provide the interface between the digital computer and the engine valves. Throttle level and attitude commands are mixed in the VDA for vernier engine control.

Error Budget

The guidance and control accuracy requirements indicated in table 11 can be provided with the reference configuration. An error budget was computed to produce sensor performance requirements as shown in table 12. These requirements are based on available equipment modified for the soft lander/support module mission as previously discussed.

TABLE 11.- GUIDANCE AND CONTROL PERFORMANCE REQUIREMENTS (3σ)

Cruise attitude errors including limit cycle . . .	0.5°
Midcourse maneuver	
Pointing	0.2°
Impulse	0.5%
Orient for support module release	0.5°
Spinup and stabilize support module	2.0°
Entry deflection maneuver	
Pointing	0.2°
Impulse	0.5%
Control entry angle of attack	5°
Achieve soft landing	
V_V	<25 fps
V_H	<10 fps

During the descent stage after parachute release a preprogramed range/velocity descent contour as illustrated in figure 43 is followed. The engines are throttled up to the setting required to follow the descent contour to a range of 60 ft above the surface and a velocity of 10 fps. From that point, the lander descends at 10 fps to an altitude (or range) of 10 ft at which time the vernier engines are shut down and the lander descends to the surface. During the descent, the LM radar is used as the primary guidance sensor provided all beams are locked. If a beam should unlock, the inertial navigator, periodically updated during the descent phase by the radar altimeter and the LR, is used as the primary sensor.

TABLE 12.- GUIDANCE AND CONTROL SUBSYSTEM CHARACTERISTICS

Component	Performance	Power, W	Weight, lb	Status
Sun sensors	±0.25° resolution	4	2	Production
Canopus tracker	±0.05° resolution	5.5	8	Production
Sun shutter		[a]6.5	1	
IMU				
Gyros	±0.25°/hr drift	30	20	Sterilization in progress
Accelerometer	±10 μg bias (compensated)			Sterilization in progress
Computer	4000 18-bit words	100 max.[b]	28	
	4 μsec memory cycle time			Teledyne Centaur computer in development (typical)
LR[c]		130	39	
3 velocity beam	2.8% or 2.8 fps			Developed with mods required
1 range beam	1.4% + 5 ft (worst case)			
AMR-1[c]	±1%, 200 000 to 15 000 ft	7.2	15	Developmental
AMR-2[c]	±1%, 15 000 to 10 ft	30	14	Developed, mods required

[a]Only when activated.

[b]25 W during cruise.

[c]Weight estimates include antenna.

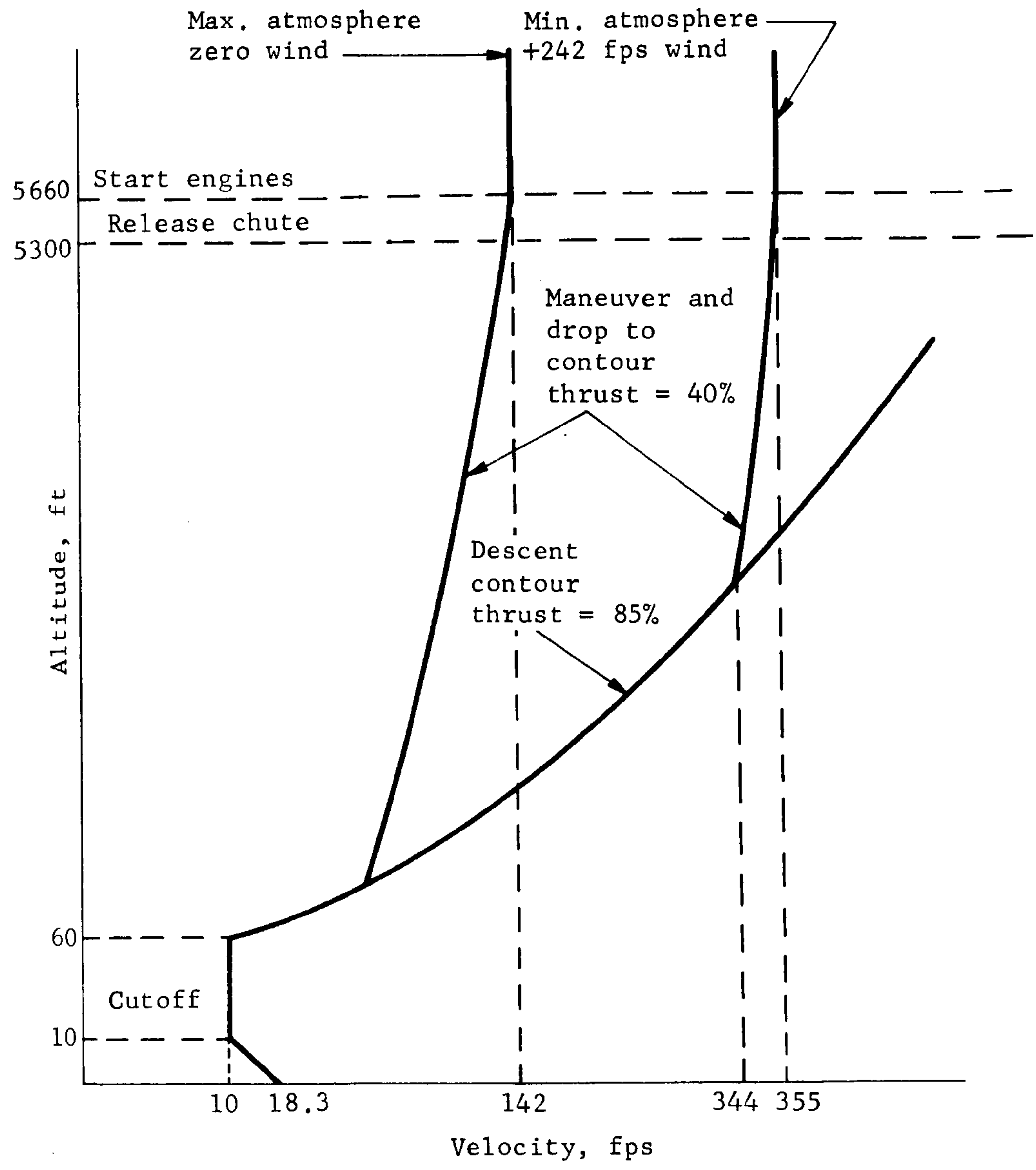


Figure 43.- Terminal Descent Axial Control Plan

Conclusions and Recommendations

The guidance and control subsystem conclusions are:

1) Mariner 69 sun sensors and Canopus tracker can be used without modifications for this application;

2) The LM radar can be used as the primary guidance sensor during the terminal phase, provided the aforementioned modifications are incorporated;

3) A low-altitude radar altimeter is required to guarantee accurate vernier ignition and to update the inertial navigator;

4) A backup inertial navigation mechanization is required to ensure mission success in case of a zero doppler or incidence angle unlock in the radar.

5. TELECOMMUNICATIONS

The soft lander/support module configuration for the Mars 1973 mission must provide the functions of command, tracking, and telemetry, as applicable, throughout the mission. These functions are to be implemented with S-band radio subsystems in conjunction with the DSIF/DSN. For this study, the DSN was assumed to consist of a network of three 85-ft antennas, and a second network of two 210-ft antennas. The 85-ft stations and one 210-ft station at Goldstone are operational sites. The other 210-ft station will be located near Canberra and will support the 1973 mission.

Radio Subsystem

The proposed radio subsystem design was evolved from an analysis of telecommunications requirements, functions, and parameters for the various mission phases. Support module rf transmitter requirements for cruise and entry relay are identical at the 25-W output power level. Command capability is required for the support module during cruise only. The soft lander requires a postland command capability. By the crossing of interfaces, an integrated design for telecommunications can be achieved resulting in an economy of hardware.

The soft lander/support module telecommunications block diagram is shown in figure 44. The elements contained in the support module form a uhf receiving subsystem and an S-band radio subsystem. The S-band radio subsystem consists of a modulator-exciter, a 25-W TWTA with an integral power supply, two low-gain, nonsteerable antennas, a fixed 40-in. circular aperture parabolic reflector, and a diplexer-rf switch assembly. The low-gain antennas are cavity-backed crossed-slots with helical feeds providing right-hand circularly polarized energy to single ports. These antennas are characterized by an on-axis gain of 5 dB with a 160° beamwidth at the 0-dB points. Two low-gain antennas are required on the support module to provide spherical coverage because Earth is not within one hemisphere for the selected launch window. Engineering data are transmitted by the 25-W TWTA over one of the two low-gain antennas at a data rate of 33-1/3 bps (uncoded) up to the completion of the first midcourse maneuver, and at 8-1/3 bps (uncoded) up to planetary encounter. The command functions for the support module are provided by the command receiver and command detector that are located in the soft lander. The requirements for near-Earth ranging and doppler tracking are provided by the hardwires crossing the interface.

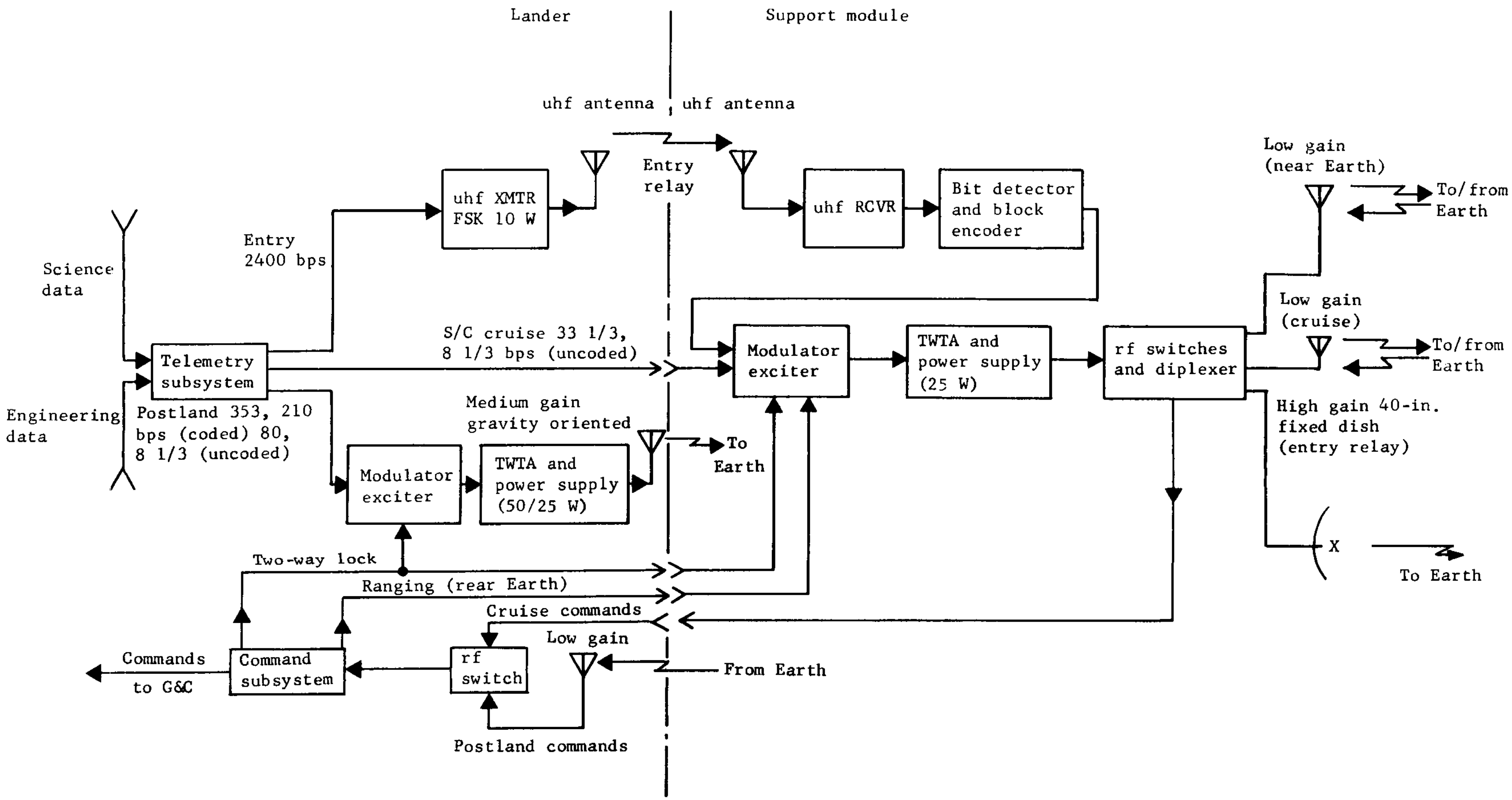


Figure 44.- Telecommunications Block Diagram

The uhf receiving subsystem consists of a body-fixed, broadbeam antenna, a receiver, and a bit detector and (32,6) block encoder. The receiving antenna is boresighted along the spin axis and provides better than 0-db gain over 160°. Capsule entry data, which are transmitted at a rate of 2400 bps, are received by this support module subsystem and retransmitted to the DSIF in real time. During the entry relay phase, the 25-W TWTA is connected to the 40-in. antenna. The characteristics of this antenna are an on-axis gain of 25.6 dB and a half-power beamwidth of 9°.

The elements contained in the lander form a uhf transmitting subsystem, an S-band radio subsystem, and a telemetry subsystem. The uhf transmitting subsystem consists of a 10/1-W FSK transmitter (400 MHz) and a broadbeam, body-fixed, transmitting antenna. The transmitting antenna is a cavity-backed crossed-slot, which provides better than 0-dB of gain over 160°. The low-power mode of the FSK transmitter is used for checkout and data transmission shortly after separation of the lander capsule from the support module.

The lander S-band radio subsystem consists of a modulator-exciter, a 50/25-W dual-power mode TWTA and integral power supply, a command receiver and command detector, a body-fixed low-gain receiving antenna, and a medium-gain transmitting antenna. The low-gain receiving antenna is identical to the low-gain antennas on the support module. The transmitting antenna is a 12.5-dB helix with a half-power beamwidth of 46.5°, which is oriented to local vertical by a counterweight.

Table 13 summarizes the weight, volume, and power requirements for this baseline design.

The radio subsystem parameters for the support module allow the 8-1/3 bps downlink telemetry channel to be supported entirely with the 85-ft antenna network until the end of Nov. 1973. Beyond this date, the 210-ft antennas are required in a listen-only mode for downlink telemetry reception. The cone angle of Earth and the distance to Earth from 9 days into the mission to planetary encounter is shown in figure 45 for the selected launch window. Telecommunications performance predictions for interplanetary cruise are shown in figure 46. Adequate margin exists in the downlink to support the near-Earth telemetry requirement of 33-1/3 bps and provide turnaround ranging for rapid change updates. The command function can be provided by the 85-ft network out to a range of 264 x 10^6 km. Performance capability of the command link for the 85-ft DSS antennas is as follows:

100 kW transmitter - 264 x 10^6 km greyout range;

25 kW transmitter - 132 x 10^6 km greyout range;

10 kW transmitter - 83.5 x 10^6 km greyout range.

TABLE 13.- WEIGHT, VOLUME, AND POWER SUMMARY

	Weight, lb	Power, W	Volume, in.^{3}
Support module			
uhf receiver	1.9	2.0	40
Bit detector and block encoder	3.6	2.5	50
uhf receiving antenna	4.0	----	7x24 diam
Modulator-exciter	3.0	2.0	90
TWTA and power supply (25 W)	8.8	85	200
Diplexer and rf switch assembly	7.0	----	100
S-band low-gain antenna	.6	----	4x4 diam
S-band high-gain antenna	6.0	----	40 in. diam
Lander/capsule			
uhf transmitter (10-W)	3.0	35	100
uhf transmitting antenna	4.0	----	7.5x15 diam
Modulator-exciter	3.0	2.0	90
TWTA and power supply (50/25 W)	9.0	150/85	240
Command receiver	5.0	2.5	150
Command detector	4.0	1.5	40
rf switch assembly	1.0	----	10
S-band low-gain antenna	1.2	----	4x4 diam (each)
S-band medium-gain antenna	2.0	----	7x5 diam

Note: Does not include packaging and internal cabling.

Telecommunications performance predictions for the uhf relay link are shown in figure 47 for the nominal trajectory parameters. The capsule/lander radio subsystem is required to relay in real time, to the support module all data collected during coast, entry, and landing. The return of all entry science data collected must be accomplished independent of landing success. This radio subsystem must be capable of transmitting 200 kbits of entry science data redundantly before touchdown.

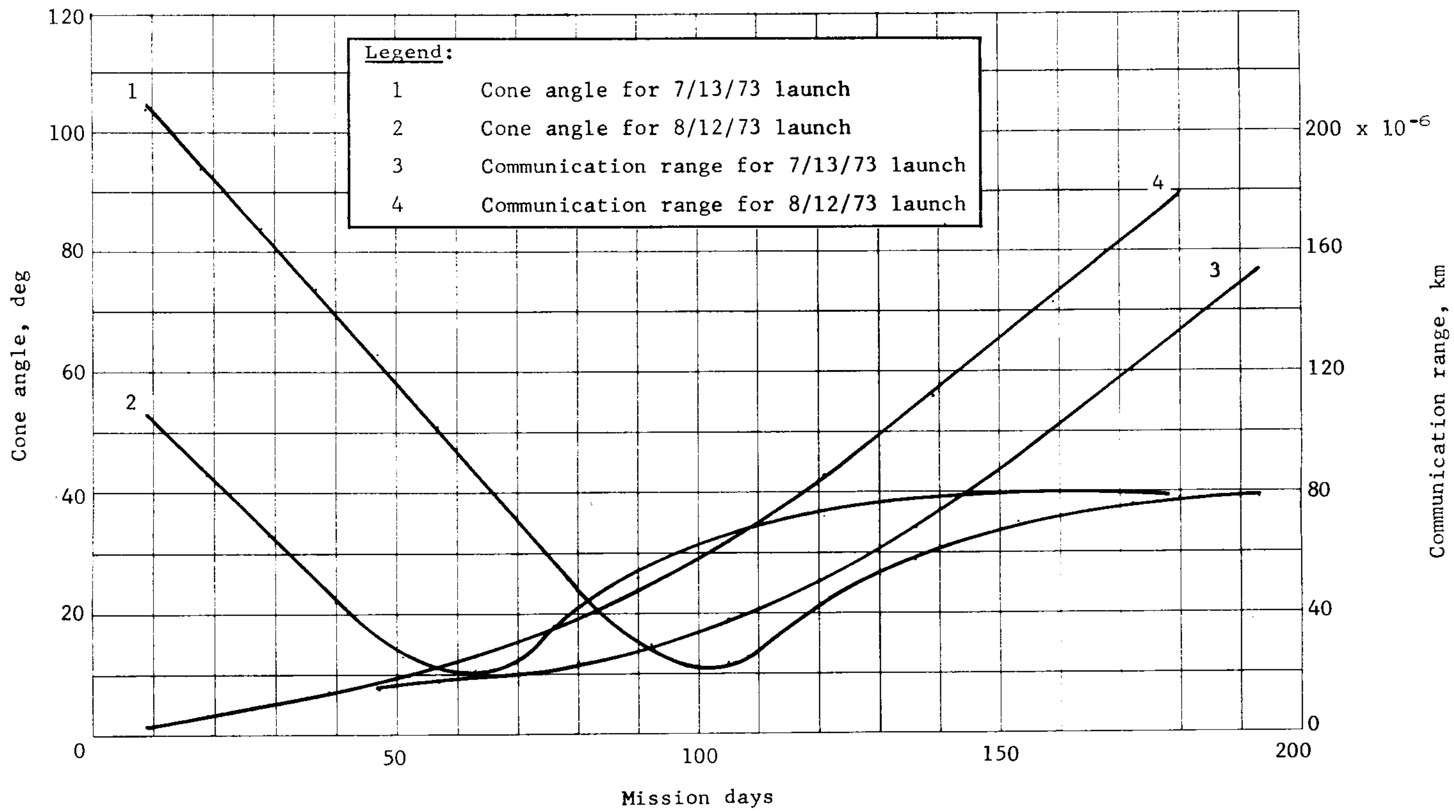


Figure 45.- Cone Angle and Communication Range

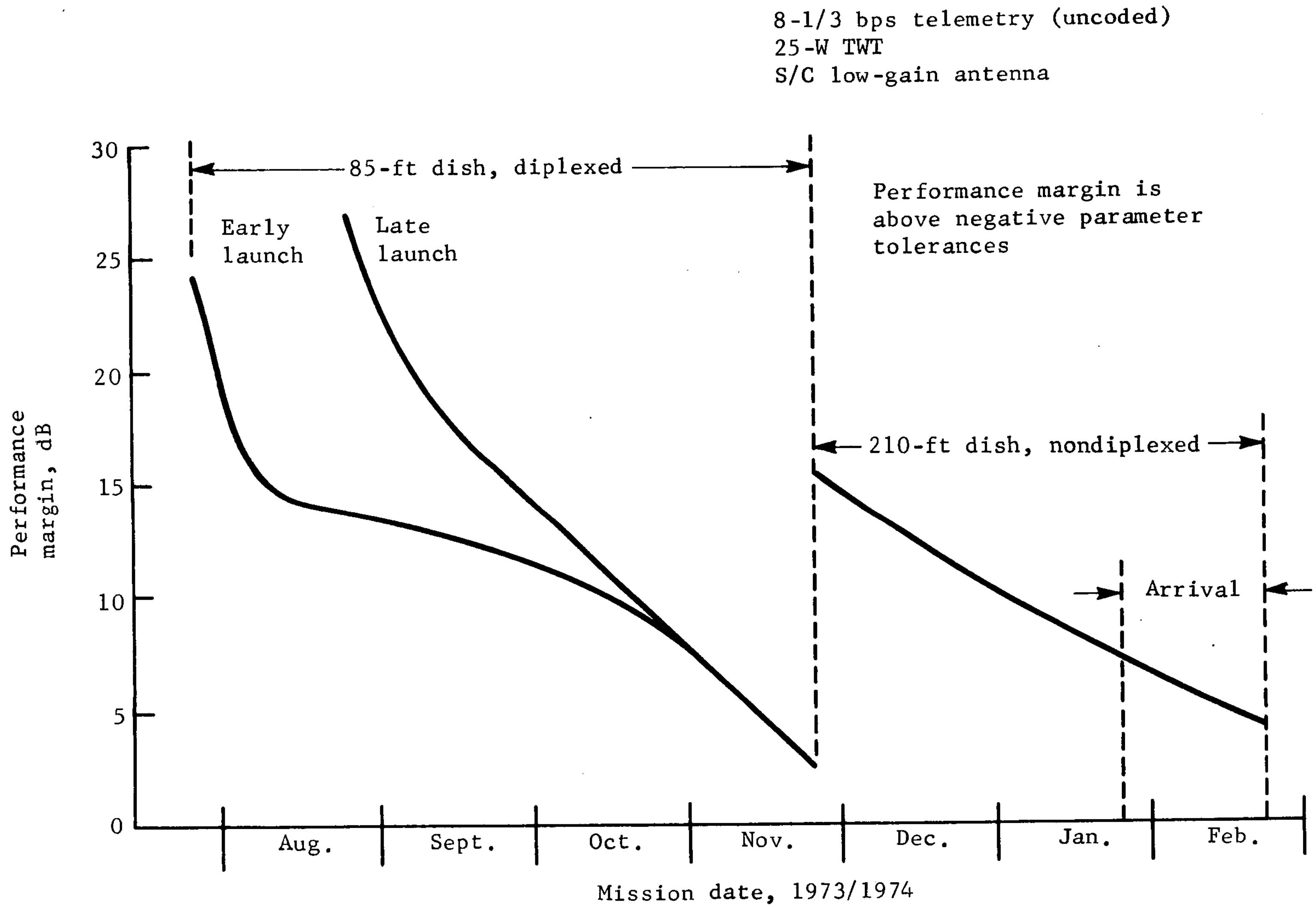


Figure 46.- Telecommunications Performance Predictions, Interplanetary Cruise

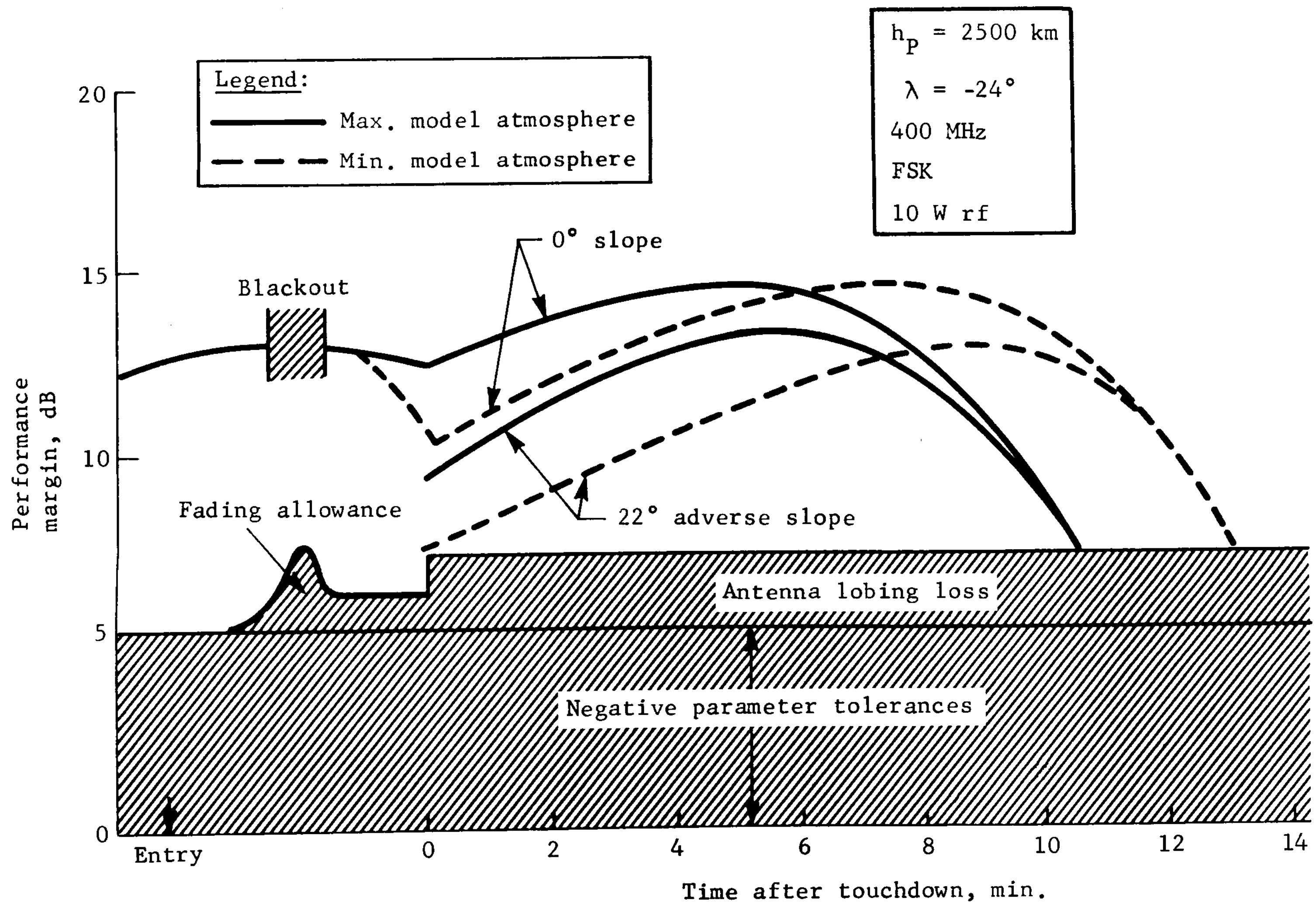


Figure 47.- Telecommunications Performance Predictions, uhf Relay Link

The figure shows that the 2400-bps data rate can be supported over a range of atmospheres and a 22° adverse surface slope. Available link time after touchdown is from 11 to 13 min, which allows for the early transmission of imaging data.

In the high-power mode, the radio subsystem in the soft lander provides an initial telemetry downlink capability of 353 bps. With an effective data transmission time of 2.8 hr/day (3.3 hr of viewtime less 0.5 hr for rf carrier acquisition), this data transmission rate provides 0.36 x 10^7 bits of science data per day. For the minimum guaranteed lifetime of 3 days on the surface, the total volume of data transferred to Earth is 1.08 x 10^7 bits. This total data volume meets the primary science data transfer requirements for the soft lander. For the remainder of the mission (landing plus 4 to 90 days after landing), the subsystem operates in the low-power mode to provide a data rate of 8-1/3 bps (uncoded) for the daily transmission of 5000 bits of meteorological data. The low-power mode data rate is capable of being detected by the 210-ft antenna stations operating in a diplexed mode at the end-of-mission life. Additional high rate telemetry channels at 210 information bits per second and 80 bps (uncoded) are provided for the extended mission by the commanding of the high-power mode of operation whenever there is sufficient energy available to operate the TWTA at the 50-W level. These additional data rates are provided to obtain science data volumes in excess of the required 10^7 bits by the complete use of available energy.

The Mars-Earth communication distance for the earliest arrival date of 1/25/74 is 160 x 10^6 km and increasing with time at a rate of about 1.5 x 10^6 km/day. The position of Earth with respect to the Martian equator is 18° south at the above date and ascending northward with time at a rate of about 0.25°/day. The desired Earth visibility with the soft lander transmitting antenna is obtained by constraining the landing-site latitude to be within ±5° of the declination of Earth existing at the time of arrival. For the 90-day mission, the 4-dB beamwidth of this antenna provides sufficient coverage for 1 hr of visibility at the end of the mission.

Telecommunications performance predictions for the postland data modes are shown in figure 48. The 80 bps (uncoded) capability with the 50-W output level of the TWTA could be changed to a 130 bps coded channel if additional data volume is required near the end of the mission.

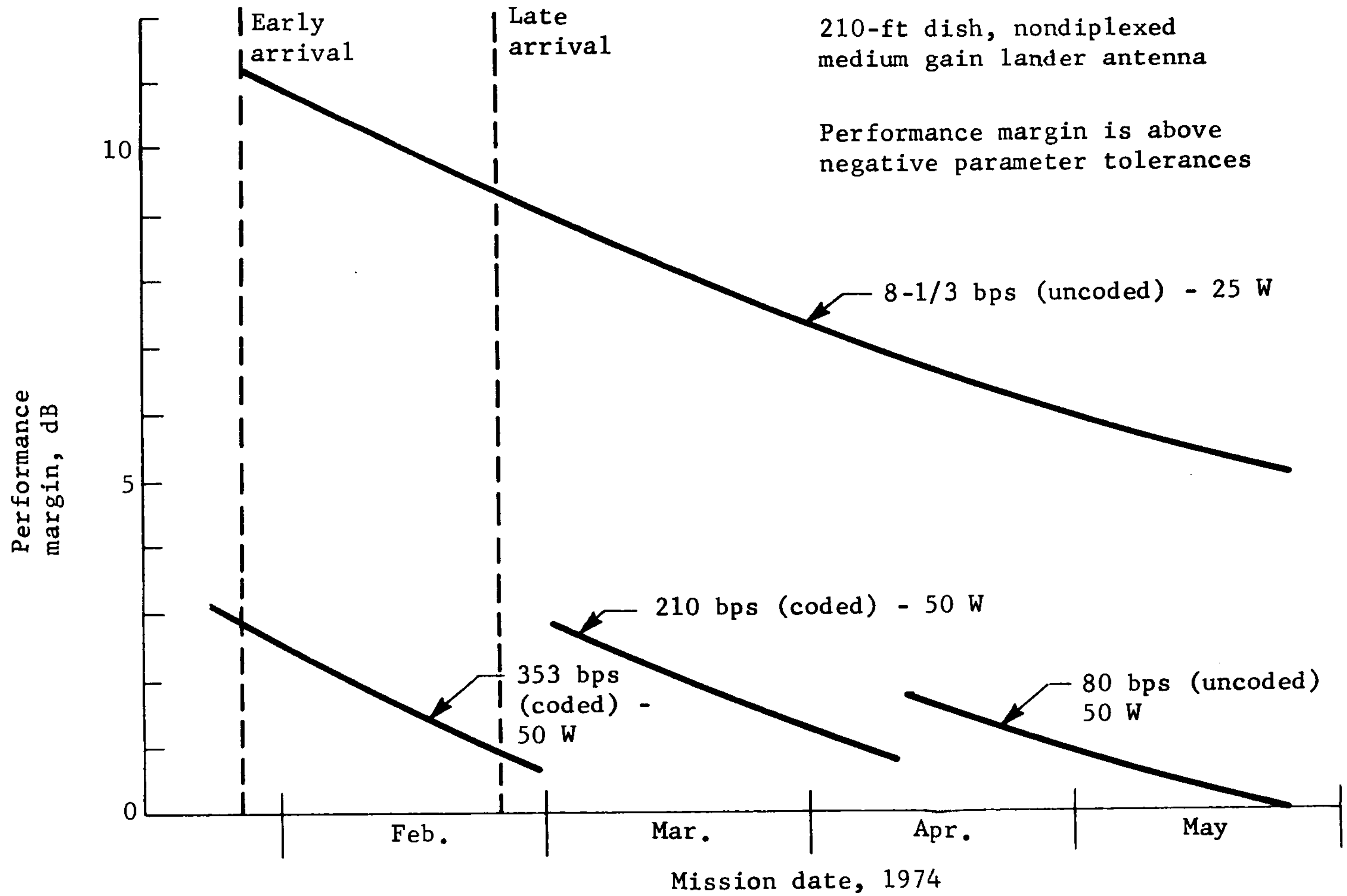


Figure 48.- Telecommunications Performance Predictions, Postland

Postland data return capability for the minimum mission and under favorable conditions are given in tables 14 and 15. The following operating modes are used to obtain science data volumes in excess of the 10^7 bits required. The TWTA will be operated in the 50-W high-power mode to provide 353 bps (coded) for the first 3 days after landing. The effective data transmission time will be 2.62 hr/day. After the primary mission, the TWTA will be normally operated in the 25-W low-power mode to transmit meteorological data at 8-1/3 bps (uncoded) for 10-min/day. Telemetry data will be used during the extended mission to determine the availability of energy in the power subsystem for operation of the TWTA in the high-power mode. The high-power mode will be commanded on whenever this mode of operation is feasible. The mode of operation for table 15, under the conditions identified, is as follows:

1) Landing + 4 days to landing + 39 days,

   2 days of operation in any 3-day interval for 2.5 hr/day at 353 bps (coded);

2) Landing + 40 days to landing + 69 days,

   2 days of operation in any 3-day interval for 2.5 hr/day at 210 bps (coded);

3) Landing + 70 days to landing + 90 days,

   Daily operation for 1.5 hr at 80 bps (uncoded).

The data rate of 8-1/3 bps (uncoded) is used from landing + 4 days to the end of the mission, whenever the high-power mode cannot be supported. Command capability for the soft lander extended mission will be provided by the 210-ft DSIF stations when the 85-ft stations greyout at a communications range of 264×10^6 km.

The following four points are presented to summarize the communications subsystem capabilities and design requirements:

1) The relay link via the support module provides adequate performance from separation to at least 10 min after landing to transmit 200 kbits of entry science data redundantly, and to return 1.4×10^6 bits of landed imaging data before the onset of the first night;

TABLE 14.- POSTLAND DATA RETURN, MINIMUM MISSION

Phase	Transmission time	Information rate, bps	Data volume, bits
After touchdown	10 min	2400	1.44×10^6
Primary landed	2.62 hr/day for 3 days	353	10.00
Weather station	10 min./day for 87 days	8-1/3	.43
Total			11.87×10^6

TABLE 15.- POSTLAND DATA RETURN, FAVORABLE CONDITIONS

Phase	Transmission time	Information rate, bps	Data volume, bits
After touchdown	13.5 min	2400	1.94×10^6
Primary landed	2.62 hr/day for 3 days	353	10.00
Extended life	2.5 hr/day for 24 of next 36 days	353	76.40
	2.5 hr/day for 20 of next 30 days	210	37.80
	1.5 hr/day for last 21 days	80	9.08
	10 min/day for 22 days	8-1/3	.11
Total			135.33×10^6
Conditions 1/25/74 arrival — No clouds 18° S lat — 90-day landed mission 0° local slope			

2) For landing sites at small southern latitudes, the postland data return requirement of 10^7 bits can be met by a direct link to Earth, entirely on batteries, and without an articulated antenna;

3) Under favorable slope and weather conditions, the goal of 10^8 bits can be exceeded during the extended mission, using power from a 25-ft^2 solar panel;

4) The proposed telecommunications designs are conservative and are based on existing technologies proven on Mariner and Lunar Orbiter. Accordingly, there have not been identified any long-lead hardware developments for telecommunications.

Telemetry Subsystem

The telemetry subsystem provides the data management functions and processing of all support module and lander engineering data from launch to the end of the 90-day landed mission. In addition, the subsystem processes analog measurements from entry science instruments and from surface science meteorological instruments, and accepts serial digital data from the science data automation system. All data processed by the telemetry subsystem are sent to the rf communications subsystem as either a split phase encoded serial PCM data stream for relay link transmission during entry and landing, or as a PSK encoded serial PCM data stream for direct-link transmission during planetary cruise and after landing.

The configuration of the telemetry subsystem is indicated by the functional block diagram of figure 49. Predicted weight, power, and volume of the subsystem are shown in table 16. Packaging and internal cabling are not included in the weight estimates, but are included in the sequential weight statement.

A single telemetry subsystem, consisting of a single data encoder, 50 kb static storage, and signal conditioner, has been selected to perform all telemetry processing and formatting functions required during all mission phases. This is an advantage accruing from an integrated lander/support module design. The telemetry subsystem is located in the lander, and the support module engineering status data are, therefore, transmitted across the staging plane over hardwire.

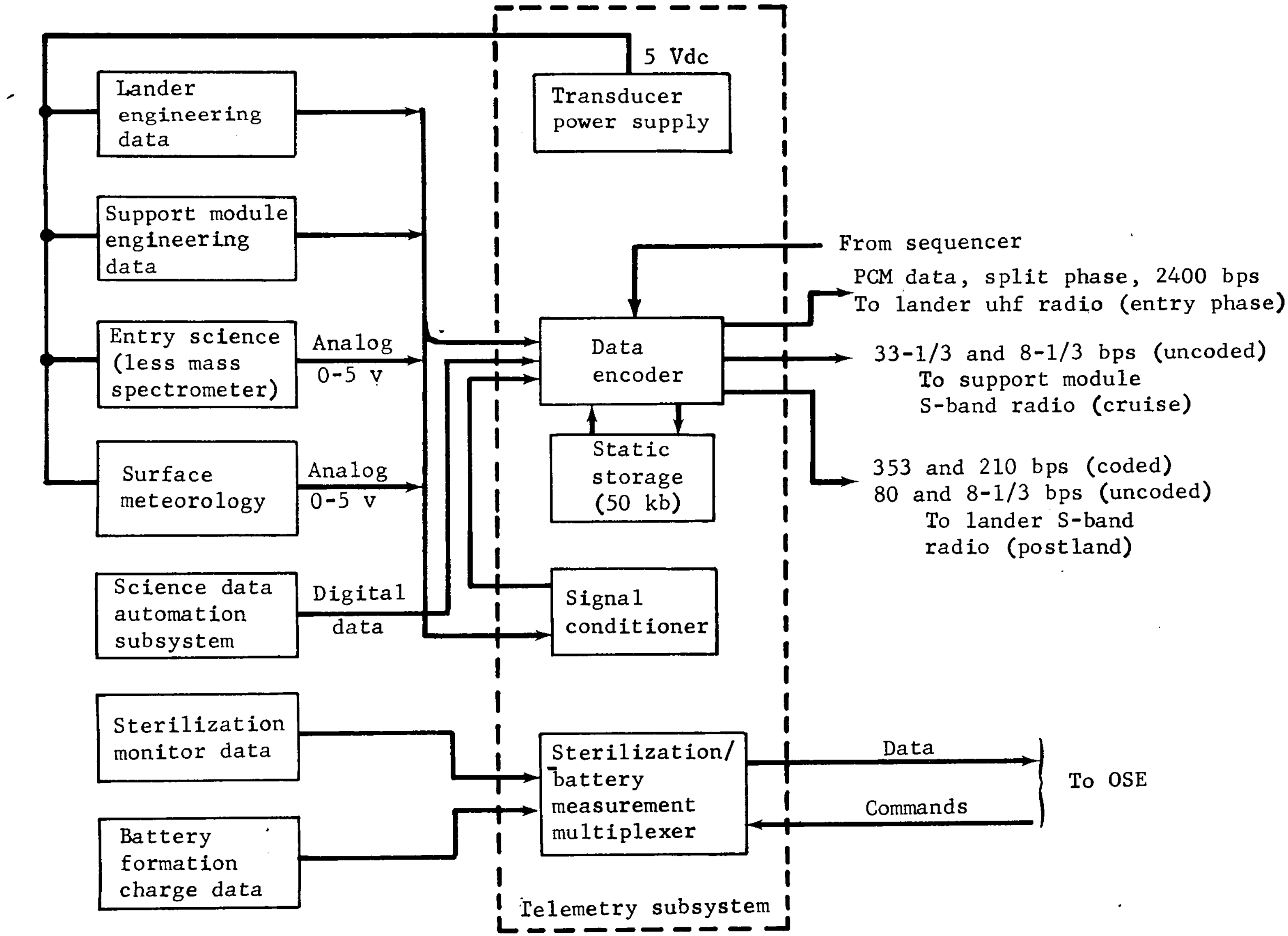


Figure 49.- Telemetry Subsystem Configuration

TABLE 16.- TELEMETRY SUBSYSTEM PREDICTED WEIGHT, POWER, AND VOLUME (a)

Component	Weight, lb	Nominal power, W	Volume, $in.^3$
Transducer power supply	0.7	2.0	16
Data encoder	16.0	15.0	600
Static storage	4.0	1.0	234
Signal conditioner	1.5	----	90
Sterilization/battery measurement multiplexer	4.0	----	108
Total	26.2	18.0	1048

(a) Packaging, supports, and cabling are excluded.

The encoder approaches follow conventional PCM telemetry designs. Hardwired programing is used to develop variable sampling rates and multiple data formats under control by discrete logic inputs from the G&C computer or postland sequencer. The data modes that are compatible with the mission profile and communications capability are shown in table 17.

The static storage provides for the delay of data collected during data mode C (and subsequent readout for interlacing of this stored data with real-time data) to permit the recovery of such data in the presence of a potential rf communications blackout during atmospheric entry. This 50-kb storage also stores postland engineering and meteorological data when real-time transmission to Earth is not possible.

The transducer power supply furnishes regulated power at 5 Vdc to the engineering and analog science transducers during all mission phases.

The sterilization/battery measurement multiplexer is an onboard checkout multiplexer for monitoring the propulsion system and batteries during terminal heat sterilization, and to check battery status during subsequent formation charging.

A brief review of existing space-used telemetry equipments was conducted; it was determined that there is no developed system that meets the formatting requirements for this mission. However, the technology required is readily available, and no major problems are anticipated in the development of hardware.

TABLE 17.- DATA MODES

Mode	Period	Description	Bit rate
A	Interplanetary cruise	Engineering data	33-1/3 or 8-1/3 bps (uncoded)
B	Separation to entry	Engineering data	2400 bps (redundant)
C	Entry to chute deployed	Engineering and science data	2400 bps (redundant)
D	Terminal descent and landing, through deployment of science	Engineering and science data	2400 bps
E	Initial postland (from deployment complete to landing + 20 min)	Science data from DAS, including imaging	2400 bps
F	Postland data collection (T/M)	Engineering and science data	One data sample per hour
G	Postland data collection (DAS)	Science	Variable
H	Postland transmission (phase I)	Stored and real-time data; engineering and science, including imaging	353 or 210 bps (coded), 80 bps (uncoded)
I	Postland transmission (phase II)	Stored and real-time data; engineering and meteorology	8-1/3 bps (uncoded)

6. POWER AND PYROTECHNIC SUBSYSTEM

The preferred design is based on the use of solar arrays using silicon cells as the energy source for the cruise and postland extended missions with Ag-Zn batteries providing the energy required during boost, capsule entry, first three days after landing, and at those times when solar array power is not available or must be supplemented. A block diagram identifying the configuration of the power subsystem is shown in figure 50.

Table 18 shows the power requirements for all elements of the vehicle for each major mission phase, and gives the type and size of each power source of the selected design.

TABLE 18.- POWER REQUIREMENTS AND SOURCE SELECTION

Configuration	Load requirements	Power source
Spacecraft interplanetary transfer	174 W, cruise 287 W, peak loads	Unsterilized solar array 205 W min. - 62 ft^2 Use of lander batteries
Support module flyby	17 hr mission 1307 W-h total	Unsterilized Ag-Zn battery 50 A-h cells - Mariner type
Capsule/lander primary mission	17 hr capsule coast 3 days after landing 6700 W-h total	3 sterilized secondary Ag-Zn batteries 80 A-h each - 39 W-h/lb
Lander extended life	Weather station 400 W-h/day	Sterilized solar array 25 ft^2 - body mounted Continued use of lander batteries

During interplanetary cruise, power is furnished by an unsterilized solar array of silicon cells, which is mounted on a surface having a 20° slope to the sun. The array is sized at 62 ft^2 to satisfy a cruise power demand of 174 W at planetary encounter distances under worst conditions. The battery power required during the boost phase and to support peak power requirements during spacecraft maneuvers is provided by the lander Ag-Zn batteries.

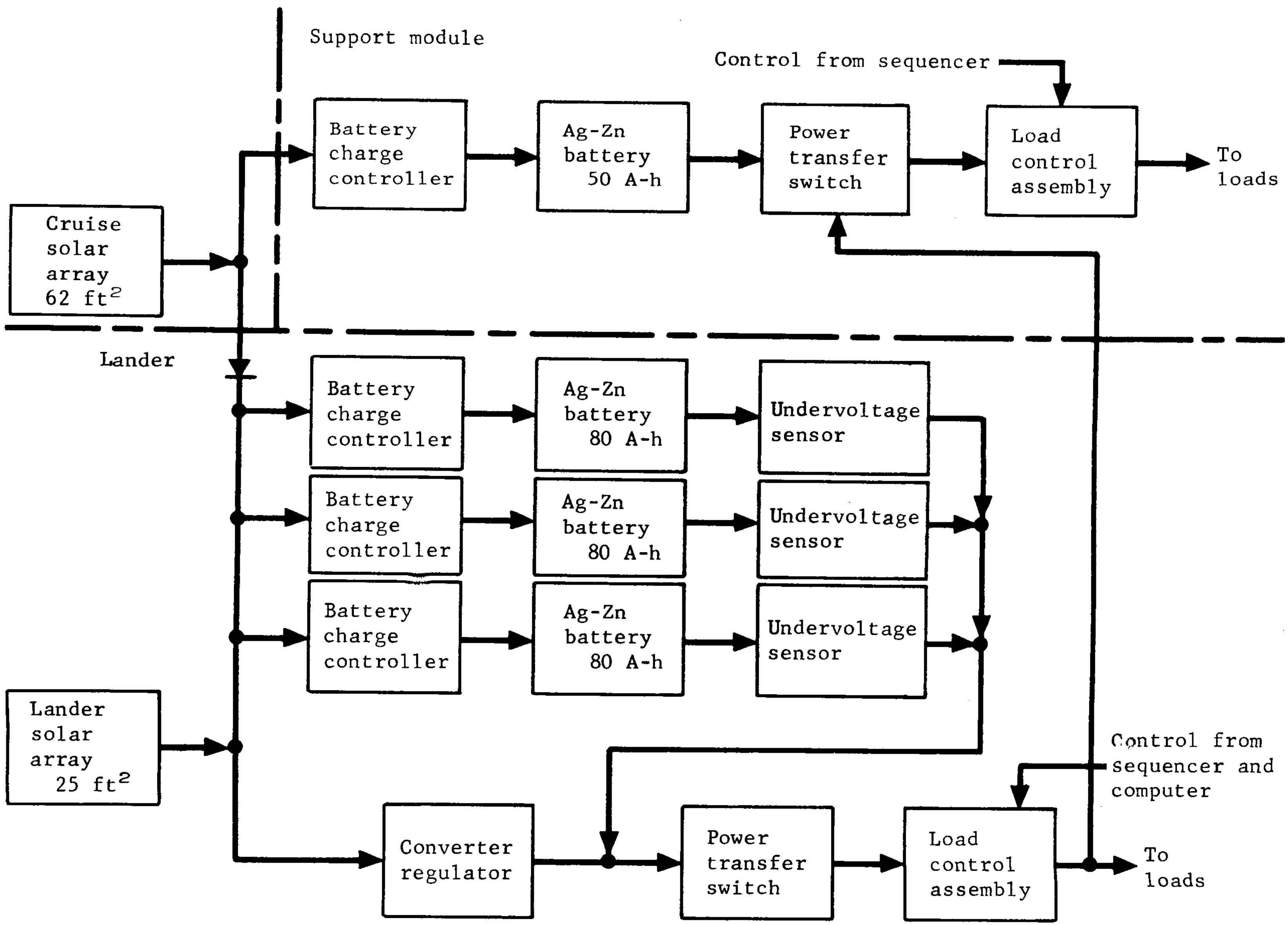


Figure 50.- Power Subsystem Block Diagram

At planetary encounter, the support module is separated, and the cruise solar array is staged. The support module then uses battery power for 17 hr to accomplish its relay link mission. The required energy of 1307 W-h is furnished by an unsterilized Ag-Zn battery. The proposed design uses 50 A-h Mariner-type cells packaged into single cell cases and arranged into the support module toroidal supporting ring as shown in the Structures section.

The power required by the capsule during coast and entry is furnished by batteries. Additional battery power is needed for the first three days of landed operations to assure mission success independent of lander solar array performance. Three 80 A-h Ag-Zn batteries supply the required total energy of 6700 W-h. At a predicted energy density of 39 W-h/lb, they weigh a total of 186 lb. Continued use of these 80 A-h batteries in support of the extended-life phase of the mission is made possible by imposing on them a requirement for about 100 low depth charge/discharge cycles.

The selection of Ag-Zn batteries for surface operations, instead of Ni-Cd as recommended in previous Mars mission mode studies, is based on three factors:

1) Development of a sterilizable Ag-Zn battery with a cycle life goal of 400 cycles to 50% depth of discharge is in progress. The lander extended life requirement is 100 cycles to 10% depth of discharge;

2) Development of a sterilizable Ni-Cd battery would result in an increased cost of approximately $700 000 for development and qualification;

3) Minimum weight - Use of a Ni-Cd battery would result in an increase of 50 lb.

For long-term surface operations, our preferred design is based on the use of a 25 ft^2 sterilizable solar array of silicon cells, body mounted on the top surface of the lander. Compared to the previous Mars mission mode study configuration, we have avoided the use of deployable, adjustable side panels. The resulting design is simpler, and less susceptible to mechanical failure. A reduction in the size of the array is possible for two reasons:

1) The requirement for 3 months landed life results in a more favorable solar array performance at the end of the mission, compared to the previous requirement of 6 months life;

2) We have reduced the power profile during the long-life mission, consistent with weather station data return requirements.

The performance of the lander solar array is shown in figure 51 for the range of proposed landing site latitudes under adverse slope conditions of 22° and a slope of 0°. The energy required from the array to meet the weather station load profile is about 400 W-h/day. The array is sized at 25 ft^2 to meet this minimum requirement at the end of the mission under the adverse conditions with about 15% margin.

More margin exists early during the landed mission, and for the more favorable ground slopes and slope azimuths. The margin resulting from the more likely slopes and azimuths can be used to accomplish three objectives:

1) To accommodate clouds;

2) To offset an unknown degree of degradation of the solar array due to the effects of settling dust and sand storms;

3) To provide improved extended-life performance in excess of weather station operations.

As an example of the latter, assuming sunny skies during the entire landed mission, and a favorable surface slope of 0°, the panel performance is greatly improved. The TWTA may then be used in the high power mode for several hours on many days as indicated in detail in figure 51. Where the requirements curve exceeds the panel performance, the difference in daily energy is made up by the batteries, which are approximately 50% charged.

The pyrotechnic subsystem design is similar to the one presented in the previously Mars mission mode study. The design uses capacitors for energy storage and solid-state switches for arming the events and firing the bridgewires.

Table 19 is a weight summary for the power and pyrotechnic subsystem.

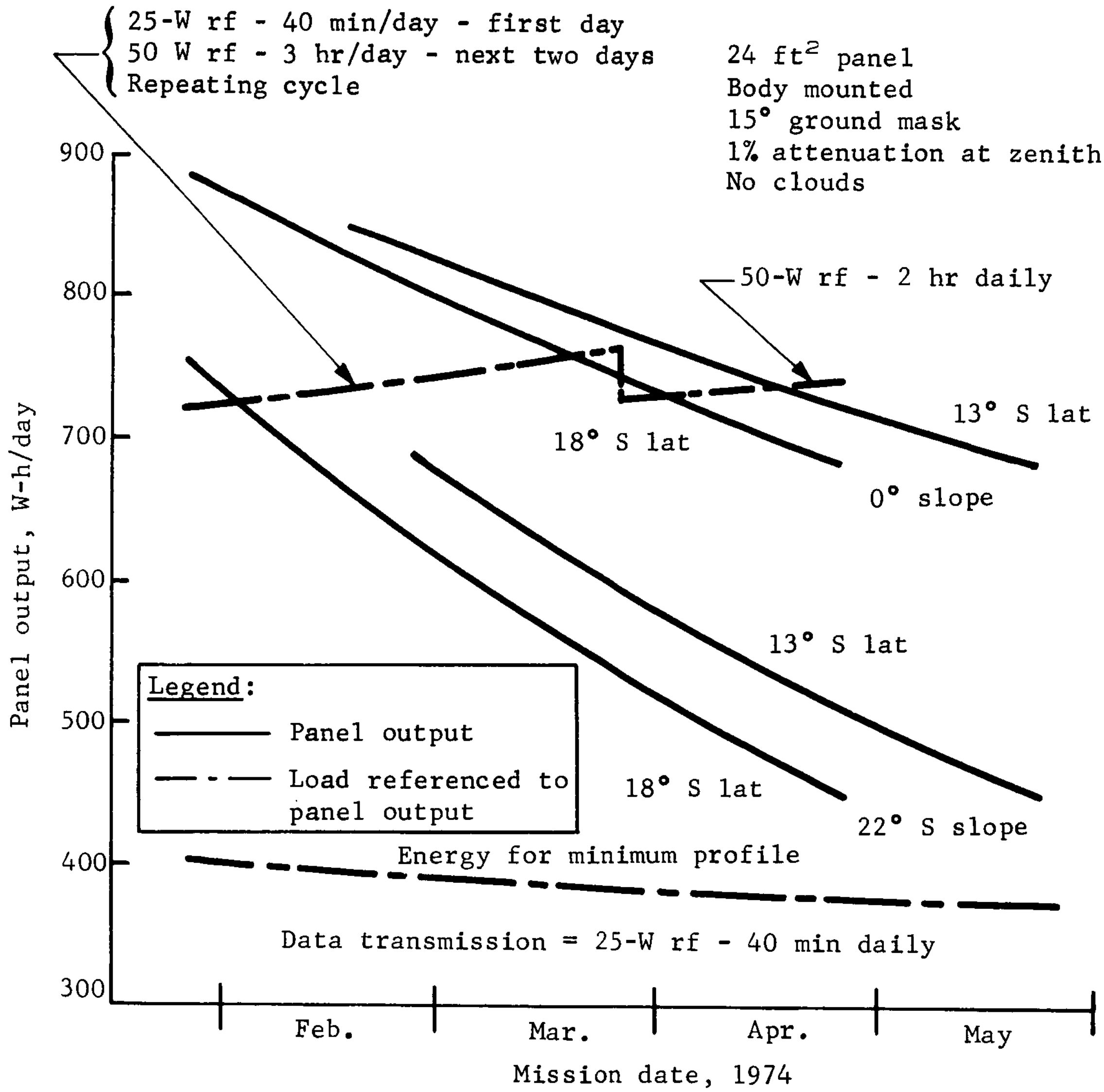


Figure 51.- Lander Solar Panel Performance

TABLE 19.- POWER AND PYROTECHNIC SUBSYSTEM WEIGHTS (a)

Item	Quantity	Weight each, lb	Total weight, lb	
Support module				
Ag-Zn battery (50 A-h)	1	34.0	34.0	
Power control and conversion	----	9.0	9.0	
Pyrotechnic energy storage and switching	----	5.8	5.8	
Support module, total				48.8
Canister-mounted equipment				
Cruise solar array (62 ft^2)	1	62.0	62.0	
Canister-mounted equipment, total				62.0
Lander				
Ag-Zn batteries (80 A-h)	3	62.0	186.0	
Solar array (25 ft^2)	1	17.0	17.0	
Power control and conversion	----	32.0	32.0	
Pyrotechnic energy storage and switching	----	31.6	31.6	
Lander, total				266.6

[a]Does not include external cabling and supports. These items are tabulated in the sequential weight statement.

Conclusions

The power subsystem conclusions are:

1) The use of one type of sterilizable secondary 80 A-h Ag-Zn battery satisfies cruise, entry, and postland stored energy requirements. The development of this sterilizable battery is a significant long-lead task to be started in Phase C;

2) A simple silicon solar array is the best program approach to achieve surface operational life beyond 3 days for a 1973 soft lander;

3) Within the more favorable design constraints of a soft lander (more weight, more volume, and lower impact shock), the power subsystem promises to support a long-term landed mission with good margins.

7. THERMAL CONTROL

From a thermal control standpoint, there are two basic differences between the flight capsule with a support module and the configurations studied during the initial mission mode study. They are:

1) Some of the lander equipment is powered up during the cruise mode to provide basic spacecraft functions;

2) A support module is added for relay communications, which must be thermally controlled through all phases of the mission.

The Mars surface thermal control is basically unchanged. During the initial mission mode study a parametric analysis was performed of many approaches to the lander thermal control system. Based on the study, we recommend the use of radioisotope heaters, primarily because this system is capable of providing long life for a wide range of Mars surface environments at an acceptable weight.

The recommended thermal control system for the unsterilized support module and flight capsule in the cruise phase is shown in figure 52. Multilayer insulation is used on the aeroshell side of the capsule to minimize the effect of the changing solar flux on internal temperatures. Insulation is also used on the opposite side, because the sun is on that side during the postseparation phase of the mission. Normally closed thermal switches are used to conduct the heat out of the heavily insulated lander to prevent overheating. These switches are opened at or shortly after landing. The side (conical) surface is used as the control surface. The emissivity is selected so that the required internal temperatures are maintained when rejecting the entire internal energy including the 200 W from the radioisotope heaters.

Multilayer insulation is used on the sun side of the support module to minimize the effect of the changing solar flux on internal temperatures. Insulation is also applied to the aft side of the support module to minimize the heat leak to space during the postseparation phase of the mission. The support module equipment heat is conducted radially by the support tray to the cylindrical radiator.

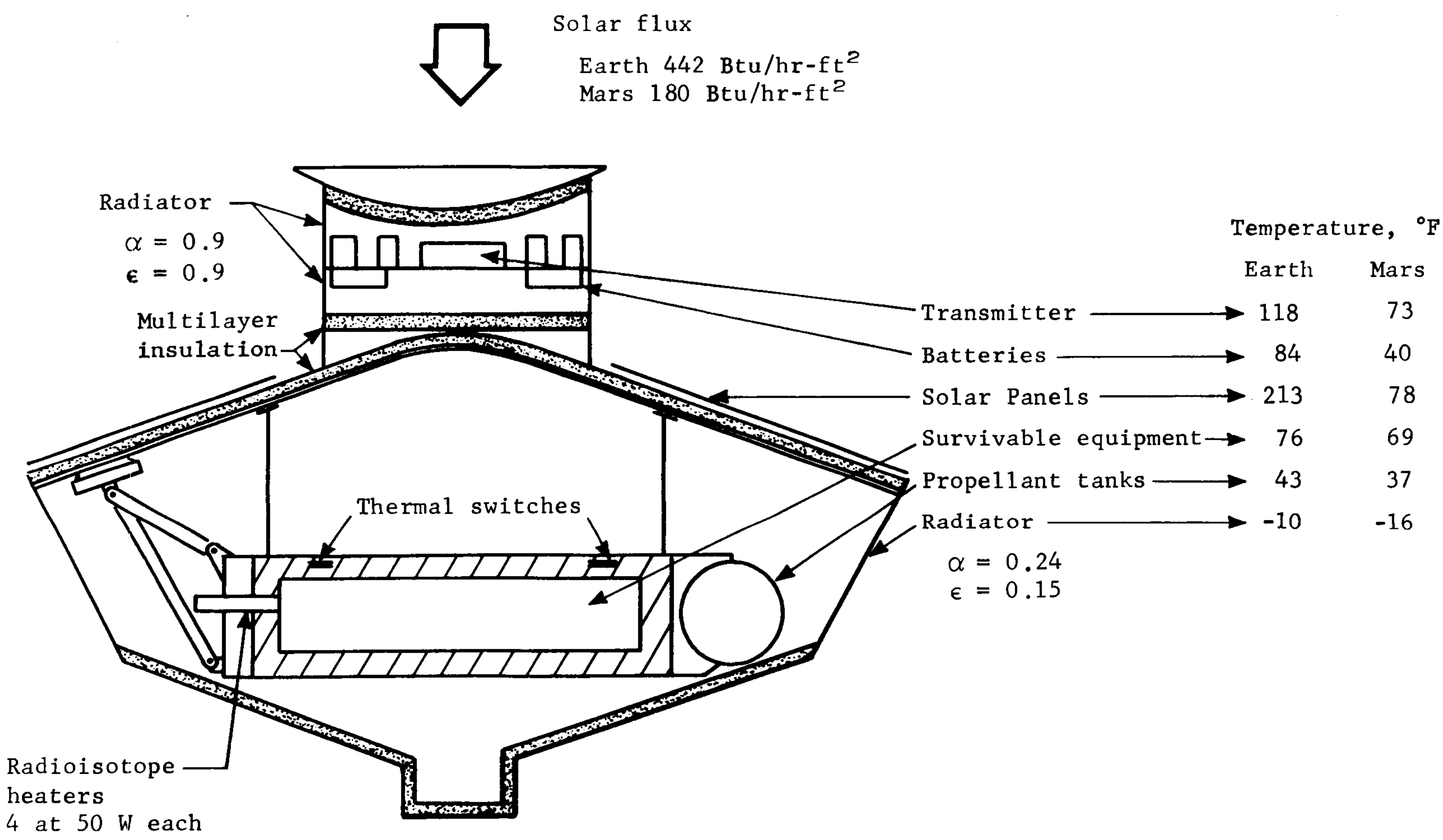


Figure 52.- Cruise Mode Thermal Control System

Computer models of both the flight capsule and the support module were developed. Some calculated temperatures are shown in figure 52 at Earth and Mars. The capsule temperatures do not change significantly in going from Earth to Mars because of the effect of the insulation. The solar panel temperatures are over 200°F at Earth, which decreases their efficiency. However, the solar array is sized by the much lower solar flux at Mars. The temperatures in the support module change in the cruise phase because of the solar flux change as well as changes in the solar reflections and infrared radiation from the capsule surface. All temperatures are within acceptable limits.

The flight capsule thermal control after separation is shown in figure 53. The sun is on the side opposite the aeroshell as the capsule orientation is reversed from its cruise position. The insulation on the aeroshell is jettisoned with the solar panels before the deflection maneuver. The guidance computer and IMU are operating during the postseparation phase of the mission and add about 100 W to the system. The emissivity (ϵ) of the aeroshell and solar absorptivity (α) of the conical surfaces are adjusted to achieve a thermal balance. The computer models were used to calculate equipment temperatures during this phase of the mission. Typical results are shown in figure 53.

Figure 54 shows the support module in the postseparation phase. It is spin stabilized with the high-gain antenna aimed at Earth, and as a result there is a 40° angle between the spin axis and sun line. The transmitter is on for the first hour, off for 15 hr and on for the last hour. Thermostatically controlled electrical heaters are used on the batteries to maintain their temperature when the transmitter is off. The battery energy is available with no weight penalty, because the developed batteries that have been selected have enough excess capacity to provide the thermal control requirements. Computer analysis was performed, and the steady-state results (transmitter off) are shown in figure 54.

Table 20 summarizes the components used and their development status. The most critical items in the thermal control system are the isotope heaters and the Mars surface insulation.

The thermal control system weights are given in table 21.

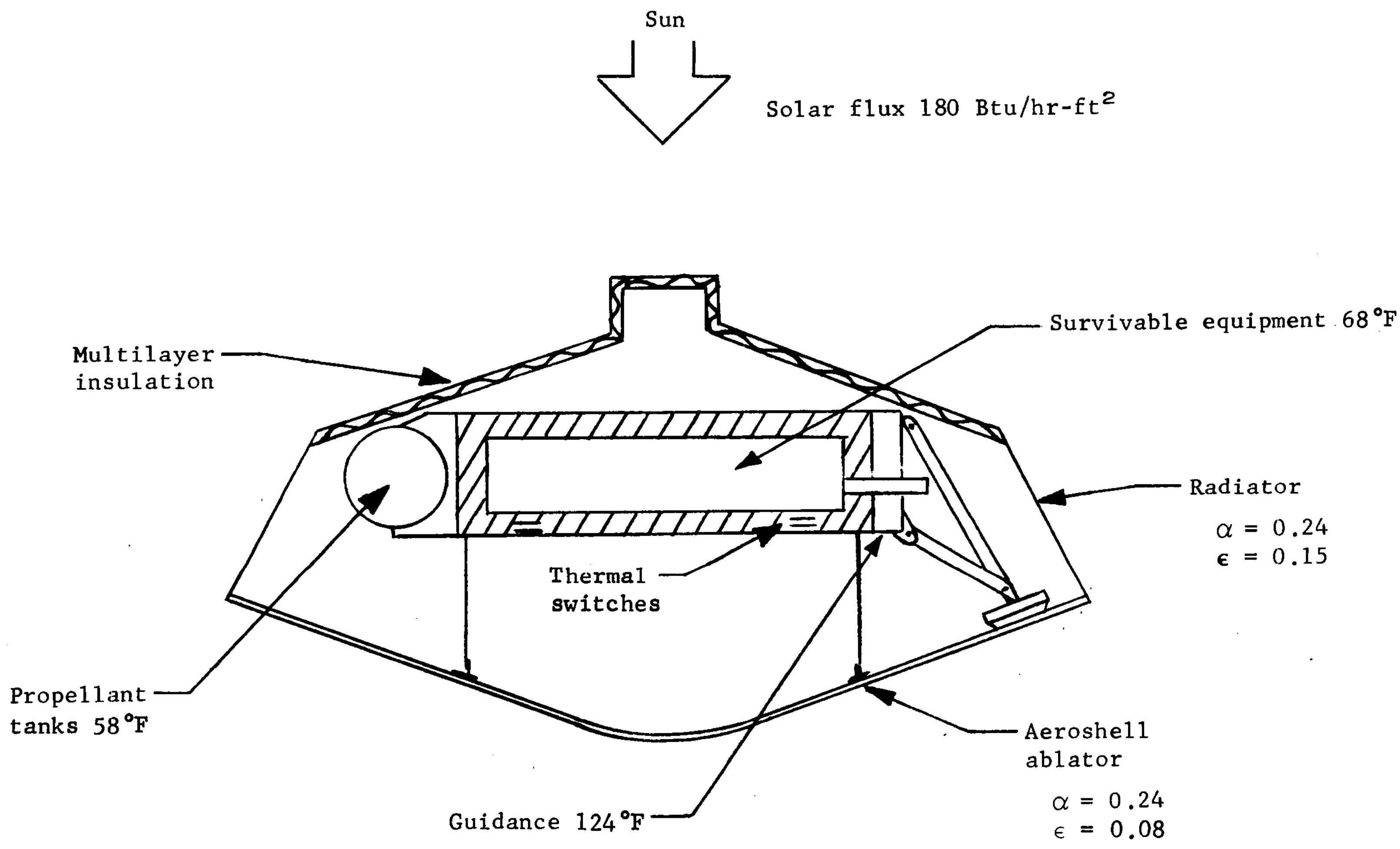


Figure 53.- Capsule Postseparation Thermal Control System

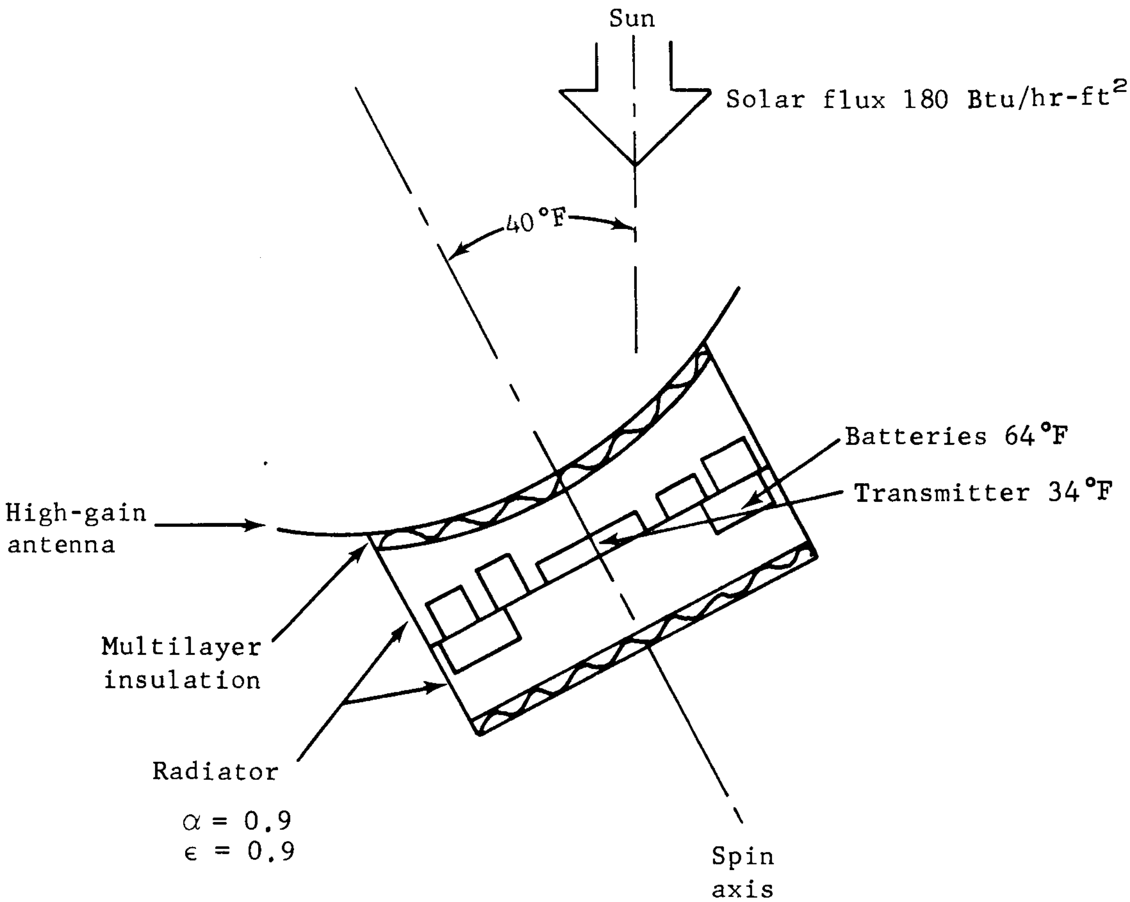


Figure 54.- Support Module Postseparation Thermal Control System

TABLE 20.- COMPONENT DEVELOPMENT STATUS

Component	Development status
Multilayer insulation - Crinkled gold-plated Mylar.	JPL development program complete. Included sterilization, vibration and full-scale thermal testing.
Mars surface insulation candidate - Lightweight fiberglas with gold-plated Kapton shields.	JPL development program has been started which will provide the required design data before phase C.
Thermal switch - Aluminum stud ejected after landing by an ordnance operated separation nut.	Ordnance operated separation nuts commonly used in space vehicles. May be possible to use one of the nuts used in the structure and avoid an additional development
Radioiosotope heater - Fuel enclosed in an ablative covered structure.	NASA Houston Phase I (study) to be complete by May 1969. Qualified units scheduled for July 1970. This program must be directed to meet the requirements for a Mars lander.
Isotope heater control - Thermostatically controlled linear actuator to move heaters in and out of lander.	Candidates are thermostatically controlled electrical actuators and a phase change material contained in a bellows. Similar type devices have been used extensively.
Mars surface coatings compatible with dust accumulation and erosion; candidate, flame-sprayed coatings.	LRC research program presently in progress.
Thermal control coatings - Black, aluminum, white, combinations.	Thermal control coatings commonly used on spacecraft. Preliminary JPL sterilization work indicates no problem with sterilization compatibility.

TABLE 21.- THERMAL CONTROL SYSTEM WEIGHT

	Weight, lb
Capsule	
Multilayer insulation	18
Coatings	14
Lander	
Isotope heaters (includes installation and actuators)	28
Mars surface insulation	12
Phase change material (includes packaging)	8
Thermal switches	5
Multilayer insulation (on nonsurvivable equip.)	4
Support module	
Multilayer insulation	1.5
Heaters and thermostats	.5

Conclusions

1) Thermal control of both the capsule and support module system designs can be achieved by relatively simple passive means;

2) No long-lead items are required for the thermal control system design if the following action is taken;

   a) The Mars surface insulation R&D work planned by JPL is done,

   b) The radioisotope heater development program in progress by NASA Houston continues and meets the technical requirements for the Mars lander mission.

CONCLUSIONS

The soft lander/support module configuration is a design that fits all the important ground rules and constraints -- cost, meaningful science, extended lifetime on Mars, Titan IIIC performance, and availability for the 1973 launch opportunity.

The configuration uses much existing equipment. The vernier engine is under development by Walter Kidde and Company. The accompanying engine throttle valve has been developed by LTV. The cruise attitude control system, sun and Canopus sensors, and the cruise solar array cells are taken directly from the Mariner '69 program. After certain modifications, the LM radar is adaptable to the 1973 Mars missions. Several other components ranging from digital computers to parabolic antennas are available.

All subsystem components are either present state of the art or can be developed for the 1973 launch opportunity. There are several long-lead items, but none are so critical that they require FY 69 funds to support the Mars 1973 soft lander mission.

The long-lead items that must start during the Phase C are terminal descent and landing radar (TDLR), inertial measurement unit (IMU), structural and thermal test models, and structural landing test models.

The following items must continue their development cycle to be available to this program -- sterilizable battery, isotope heaters, and science payload instruments.

Several areas have been identified that appear to merit further investigation as a result of this study effort. Use of the Kidde engine/LTV throttle valve combinations could significantly reduce program cost and risk. A test program that will demonstrate the engine/valve compatibility is desirable. A detailed six degree of freedom simulation of the selected TDLR should be considered. Work should proceed in simulating the radar/parachute phase, including wind and gust effects.

Martin Marietta Corporation
Denver, Colorado, November 11, 1968

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NASA CR-66661	- Volume III	- Appendix A - Launch Vehicle Performance and Flight Mechanics;
NASA CR-66662	- Volume IV	- Appendix B - Entry and Terminal Phase Performance Analysis;
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