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Volume I
Summary

Final
Report

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A Feasibility Study of Unmanned Rendezvous and Docking in Mars Orbit

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A FEASIBILITY STUDY
OF
UNMANNED RENDEZVOUS AND DOCKING
IN MARS ORBIT

Final Report

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SUMMARY

July 1974

Approved

William T. Scofield

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ABSTRACT

The technical feasibility of achieving automatic rendezvous and docking in Mars orbit as a part of a surface sample return mission was investigated based on using as much existing Viking '75 Orbiter and Lander hardware as possible. Both 1981 and 1983/84 mission opportunities were considered. The principal result of the study was the definition of a three stage 289 kg Mars Ascent Vehicle (MAV) capable of accepting a 1 kg sample, injecting itself into a 2200 km circular orbit, and rendezvousing with an orbiting spacecraft carrying an Earth Return Vehicle.

The modifications necessary to convert a Viking '75 Orbiter to the sample return mission orbiter are defined. These consist primarily of propulsion system changes and the addition of a rendezvous radar sensor. Required modifications to the Viking Lander are also described; the major ones being the addition of a MAV erector/launcher mechanism and thermal control canopy on the existing equipment platform and converting the terminal descent propulsion to a pressure regulated system.

Digital computer simulations of dispersed MAV ascent and orbit injection and circularization were performed to establish the conditions at start of terminal rendezvous. Flight control laws were then established which would be preprogrammed into the orbiter's computer to effect final closing and docking of the two vehicles in the presence of dispersed as well as nominal conditions at start of rendezvous.

Conclusions are that with state of the art systems plus limited application of new developments in areas where feasibility has already been demonstrated, e.g., solid rocket motor sterilization, it is possible to land a small ascent vehicle capable of automatically ascending and rendezvousing with a modified Viking '75 orbiter spacecraft. The mission can be flown in 1981 or 1983/84, but a dual launch or a larger launch vehicle than the Viking Titan III Centaur, or the use of space storable propellants for Mars orbit injection, would be required in the 1983/84 opportunity.

ACKNOWLEDGEMENT

Martin Marietta wishes to recognize the contributions of the following individuals to this study:

Jesse W. Moore of the Jet Propulsion Laboratory, Technical Representative of the Contracting Officer, for management and direction.

Donald Udlock of the Jet Propulsion Laboratory for supplying data on JPL studies concerning solid rocket motor sterilization.

Daniel H. Herman, Director Advanced Planetary Programs, NASA Headquarters, for general guidance and support.

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GLOSSARY

ACS	Attitude Control System
AZL	Azimuth of MAV Ascent Vehicle
CC	Control Computer
CIRC	Orbiter Circularization Maneuver
CMOS	Complementary Metal Oxide Semiconductors
CST	Coast Time Between MAV 1st and 2nd Stage Ascent Burns
DSN	Deep Space Net
DVM	Magnitude of ΔV
EEC	Earth Entry Capsule
ERV	Earth Return Vehicle
FOV	Field of View
H/S	Heat Shield
G&C	Guidance and Control
GCSC	Guidance Control and Sequencing Computer
HGA	High Gain Antenna
LATL	Landing Sight Latitude
LONG	Landing Sight Longitude
LOS	Line of Sight
LSI	Large Scale Integrated (circuit)
LVMP	Launch Vehicle Mission Peculiars
MIC	Microwave Integrated Circuits
MAV	Mars Ascent Vehicle
MNOS	Metallized Nitride on Silicon
MOI	Mars Orbit Insertion
MOR	Mars Orbital Rendezvous
MSSR	Mars Surface Sample Return

PCM	Pulse Core Modulation
PSK	Pulse Shift Keying
PM	Phase Modulation
PN	Proportional Navigation
PROM	Permanent Read Only Memory
RAM	Random Access Memory
RCS	Reaction Control System
RF	Radio Frequency
RR	Rendezvous Radar
RECIRC	MAV Post Circularization Trim
S/A	Safe/Arm (Device)
SCR	Silicon Controlled Rectifier
STEM	Storable Tubular Extendible Member
TEI	Trans-Earth Injection
TR	Terminal Rendezvous
TRI	Terminal Rendezvous Initiation
THR	Orbiter Thrust
URDMO	Unmanned Rendezvous and Docking in Mars Orbit

LIST OF SYMBOLS

α	right ascension of thrust direction
β	declination of thrust direction
β	ballistic coefficient m/C_DA
γ	flight path angle
γ_E	flight path angle at entry
ΔV	delta velocity (vehicle velocity change)
ΔV_C	closing ΔV (for start of terminal rendezvous)
ΔV_H	ΔV for Hohmann transfer
ΔVLBI	differential very long baseline interferometry
ΔV_{MOI}	velocity change for MOI
V_{PC}	velocity change for plane change
ΔV_{STAT}	statistical ΔV
ΔV_T	terminal ΔV (total ΔV used in rendezvous control law burns)
Δ	change
$\Delta V_1, \Delta V_2, \Delta V_3$	orbiter trim maneuvers, control law burns
$\Delta \gamma_{STAT}$	statistical flight path angle variation from nominal
θ	angle traversed in terminal rendezvous (transfer angle)
θ_{AIM}	angle between B-vector and $\vec{T}$-axis
θ_{MI}	angle between B-ellipse minor axis and T axis
θ_o	initial launch ramp angle
θ_s	cone half angle
$\dot{\theta}$	constant pitch rate after launch
μ	gravitational constant
σ_{BMIN}	standard deviation of B-vector magnitude along minor axis of B-ellipse
$\sigma_x, \sigma_y, \sigma_z$	standard deviations of position (Cartesian components)

$\sigma_{\dot{x}}, \sigma_{\dot{y}}, \sigma_{\dot{z}}$	standard deviations of velocity (Cartesian components)
$\sigma_u, \sigma_v, \sigma_w$	standard deviations of position (orbit plane components)
$\sigma_{\dot{u}}, \sigma_{\dot{v}}, \sigma_{\dot{w}}$	standard deviations of velocity (orbit plane components)
$\sigma_{\dot{\rho}_p}$	standard deviation in projected relative velocity
ϕ_o	initial phase angle at o.e.
$\dot{\phi}$	phase angle catch up rate
Ω	longitude of ascending mode
ω	argument of periapsis, LOS rate
A	base area or reference area (in ballistic coefficient)
a	semi-major axis
B-plane	plane perpendicular to VHE vector
B-vector	center of planet to B-plane impact point
b_p	porjection of baseline vector
C_D	aerodynamic drag coefficient (in ballistic coefficient)
D_o	parachute diameter (deflated)
E	covariance matrix of launch errors
e	eccentricity
HP	orbiter periapsis altitude adjust maneuver
h_p	periapsis attitude
h_a	apoapsis attitude
i	inclination
M	injection sensitivity matrix
m	mass (in ballistic coefficient)
O.D.	Orbit Determination
o.e.	occultation exit
P	period
P_{INJ}	covariance matrix of injection dispersions

P_{ORB}	period of orbiter orbit
P_{MAV}	period of MAV orbit
P_{phase}	period of phasing orbit
R	range
$\dot{R}$	range rate
R_A	actual MAV position vector
R_E	radius of earth entry
R_N	entry capsule nose radius
R_{SB}	entry capsule base radius
r_{EM}	Earth-Mars distance
$\vec{R}, \vec{S}, \vec{T}$	coordinate axes for B-plane coordinates: S along VHE, T in ecliptic, R completes landed system
$\delta t_{12}, \delta t_{23}$	times between trim maneuvers #1, #2 and #2, #3
TA	true anomaly
T_1	thrust of stage #1
T_2	thrust of stage #2
t	time
t_B	burn time
t_{B1}	time of stage #1 burn
t_{B2}	time of stage #2 burn
t_R	time for rendezvous
V_E	entry velocity
VHP	hyperbolic excess velocity
V_∞	hyperbolic excess velocity
W_p	propellant weight
X	generic designation for position
$\dot{X}$	generic designation for velocity

I MARS SURFACE SAMPLE RETURN -- BASIC ISSUES

In all forms of human progress there are routine steps, and there are giant strides. In man's developing understanding of the history of the cosmos and his place in it, there are likewise opportunities for leaps of knowledge. One of these is the correlation of the geological, chemical and biological history, and currently active processes of the Earth, with those of the most Earth-like of our planetary neighbors, Mars.

In the exploration of Mars, one mission, the Mars Surface Sample Return (MSSR), stands above all others in scientific importance--in the potential for answering first order, fundamental questions. The MSSR mission, by providing specimens of Mars material for direct examination in Earth laboratories, will add more to our knowledge of the planet than any other conceivable unmanned expedition.

The value of a Mars surface sample return mission, compared with the delivering of automatic scientific instruments to operate on the planet surface, accrues in four general areas:

1) complex investigations such as age dating, petrological analyses, detailed biochemical analyses and direct observation of biological activity can be performed in Earth laboratories to a precision that would be infeasible technically and economically with remotely operated instruments;
2) a large number of investigations can be performed on a single sample, each designed by the results of previous ones, making a single MSSR mission equivalent to many preprogrammed in situ science missions;
3) Mars samples, once brought back to Earth can be analyzed by instruments representing the latest state of the art whereas remotely operated instruments would be frozen at a technology level at least five years out of date;
4) The full intellectual power of the world scientific community can be brought to bear on the examination and interpretation of returned samples, and in fact part of the returned material can be handed on as a legacy to future generations of scientists whose skills and tools can be expected to exceed ours.

II TYPICAL MSSR MISSION SEQUENCES

There are a number of valid alternatives in designing the MSSR mission. The choice among them will eventually become one involving cost and performance risk. For purposes of illustration, Figure II-1 will be used to define the typical mission phases. It represents the single launch, direct entry, Mars orbital rendezvous, conjunction class mission mode.

Following the numbered sequences in Figure II-1, Step 1 represents the Earth launch and Earth to Mars cruise phase of the total spacecraft. In this case a single launch of a vehicle stack comprising an orbiter, an Earth Return Vehicle (ERV) and a lander capsule is shown. This phase of the mission has been well proven in the Mariner Mars series of flights. Alternatives to this single launch case that offer some particular advantages, will be discussed later.

At Step 2, four hours prior to Mars encounter the lander capsule is separated from the orbiter, performs a deflection maneuver, enters the Mars atmosphere and lands. This direct entry mode was examined in detail in the Alternate Viking '75 Mission Mode Study (Ref. 3) performed under the auspices of the Viking '75 Project in 1970.

At essentially the same time that the lander is entering, the orbiter is performing the Mars orbit insertion (MOI) maneuver to go into an orbiting sequence that will eventually place it in the proper rendezvous orbit (Step 3).

The Mars landing (Step 4) is performed in the same manner as Viking '75 using aeroshell/heat shield, parachute and terminal propulsion systems to control the descent to a final touchdown velocity of approximately 2 to 3 mps. The lander carries to the surface a Mars Ascent Vehicle (MAV) that will be used to deliver the sample back to the orbiting spacecraft.

The principal activities during landed operations (Step 5) will be: 1) imaging of the available sampling area; 2) selection, collection and stowage of the sample(s); and 3) updating of the lander position and attitude and calculation of required MAV launch azimuth and elevation.

At Step 6 the MAV is elevated and rotated to the launch position and commanded to launch.

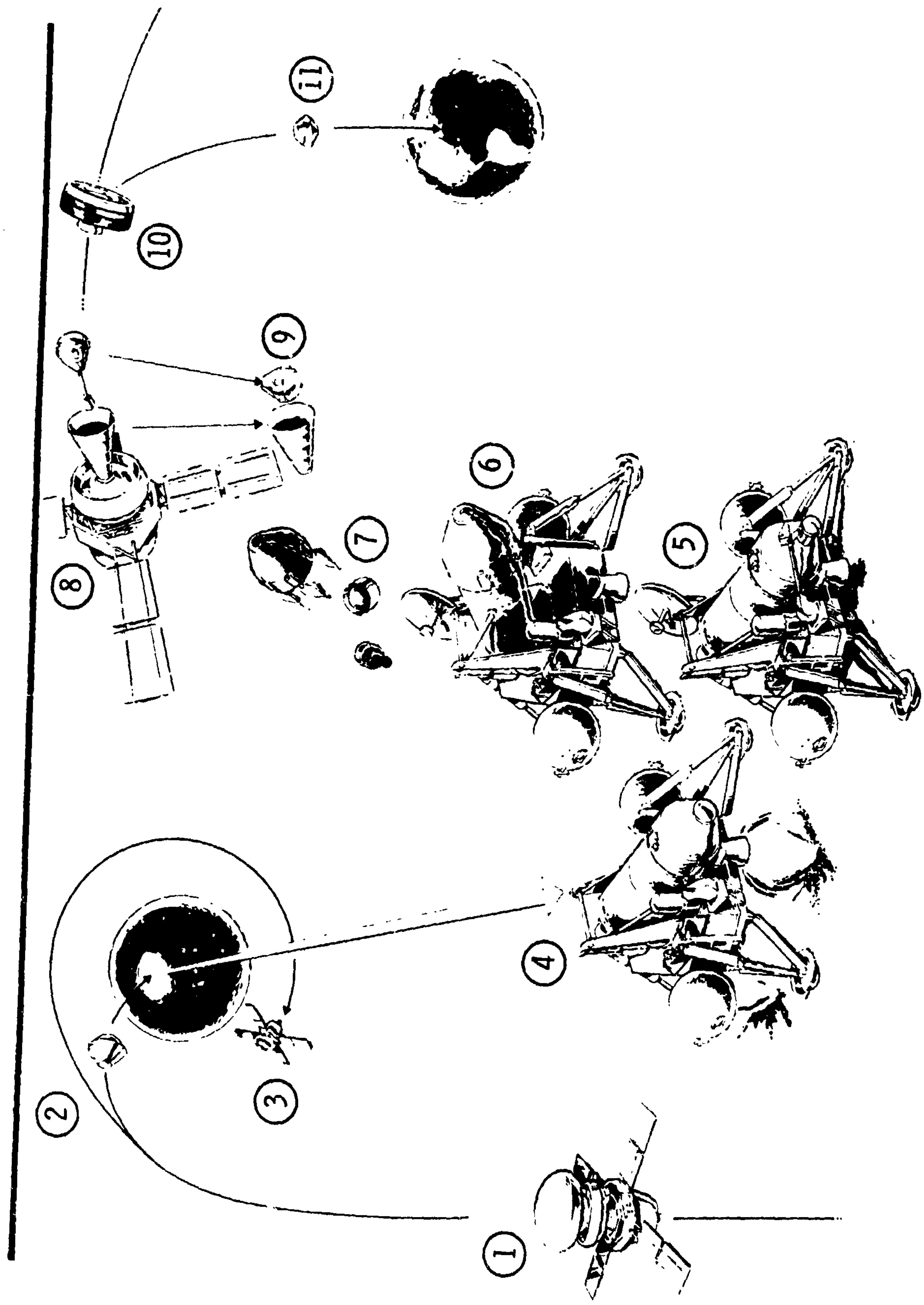

Figure II-1 MSSR Mission Sequence - Mars Rendezvous Mode

The ascent of the MAV (Step 7) involves the firing of two solid rocket stages to achieve an initial, Earth trackable orbit and then a circularization into the rendezvous orbit with a third, liquid propulsion stage.

The rendezvous of the orbiter and sample-carrying MAV (Step 8) is accomplished with maneuvers of the more sophisticated orbiter rather than the MAV in order to keep the latter vehicle as simple as possible.

After rendezvous, docking and sample transfer, the MAV and the docking cone are discarded (Step 9). The sample canister is now safely stowed in the ERV.

In the conjunction class mission the ERV and sample must remain in Mars orbit for approximately 400 days before the planetary geometry will allow the initiation of an efficient Earth return trajectory (Step 10). The ERV could be an adaptation of a Pioneer Venus spin-stabilized orbiter whose interplanetary cruise capability will have been proven in the 1978 flights to Venus. The ERV design was not within the scope of this study.

Upon encountering Earth, the Earth Entry Capsule, carrying the sample, is aimed at the proper Earth entry corridor and separated (Step 11). In the mission mode illustrated here, the capsule will enter directly using a heat shield and parachute for deceleration, and be recovered either by air snatch or after land impact.

Table II-1 summarizes the timing, performance and weight characteristics of the typical MSSR mission profile illustrated in Figure II-1. The baseline mission launch opportunity has been chosen as 1981. The total timeline spans approximately 1050 days from Earth launch to sample recovery and includes allocations of 11 days on the Mars surface, 16 days for rendezvous and docking and 400 days wait time in Mars orbit. The Mars direct entry velocity of approximately 5800 mps ($\sim$ 19,000 fps) compares with the Viking '75 out of orbit entry velocity of 4630 mps ($\sim$ 15,000 fps).

This single launch, direct entry, Mars orbital rendezvous mission mode requires the least amount of spacecraft weight-carrying capability. In this mission profile all the required sequences will have been proven by previous missions except for the Mars ascent, rendezvous, docking and sample transfer. It is also important in minimizing mission cost and risk that in this MSSR mode the proven sequences will be carried out by modified versions of the spacecraft designs that originally performed them.

It is appropriate then that this study was directed primarily at the examination of the mission sequences that are new and untried: ascent, rendezvous, docking and sample transfer.

III MISSION MODE OPTIONS

In addition to the mission mode options described in the preceding chapter, there are others that offer some distinct advantages at the cost of greater spacecraft weights and program funding.

Two options most closely related to the one just covered involve splitting the Earth to Mars phase into two launches; one to carry the lander and MAV and the other to handle the orbiter and ERV. The portions of these two options that differ from the profile in Figure II-1 are illustrated in Figures III-1 and III-2.

Figure III-1 shows the dual launch mode in which the lander still enters the Mars atmosphere directly from the incoming asymptote but is supported during the Earth to Mars transfer by a separate cruise module. After the lander separates from the cruise module (four hours prior to encounter) the latter flies by Mars on a continuing heliocentric trajectory. The advantage of this mode is that the restrictions on total spacecraft weight are not set by the launch capability of one launch vehicle but can grow, theoretically, to the limits of two launch systems. The dual launch mode also offers the potential advantage of clean interfaces in the event responsibility for the MSSR mission was to be divided between two nations.

The other dual launch mode, shown in Figure III-2, uses two orbiters, one of which carries the lander into orbit prior to commitment to a landing site. This option offers the obvious added advantages of out of orbit landings: landing site certification before landing, and the ability to wait out dust storms that might have developed at the landing site. It is interesting to note that for the 1981 mission opportunity the orbiter required to carry the ERV to the rendezvous orbit and the orbiter required to carry the lander to a 24-hour orbit for landing initiation are essentially the same size. This could mean that the dual launch, out of orbit mode would be cheaper than the dual launch direct entry method because it could avoid the cost of developing the new cruise module.

The full set of potential MSSR mission modes is represented in the sketches of Figure III-3. The modes involving direct return, in which a sample carrying vehicle is capable of ascending from the Mars surface and returning to Earth without Mars orbital rendezvous, have the advantage of

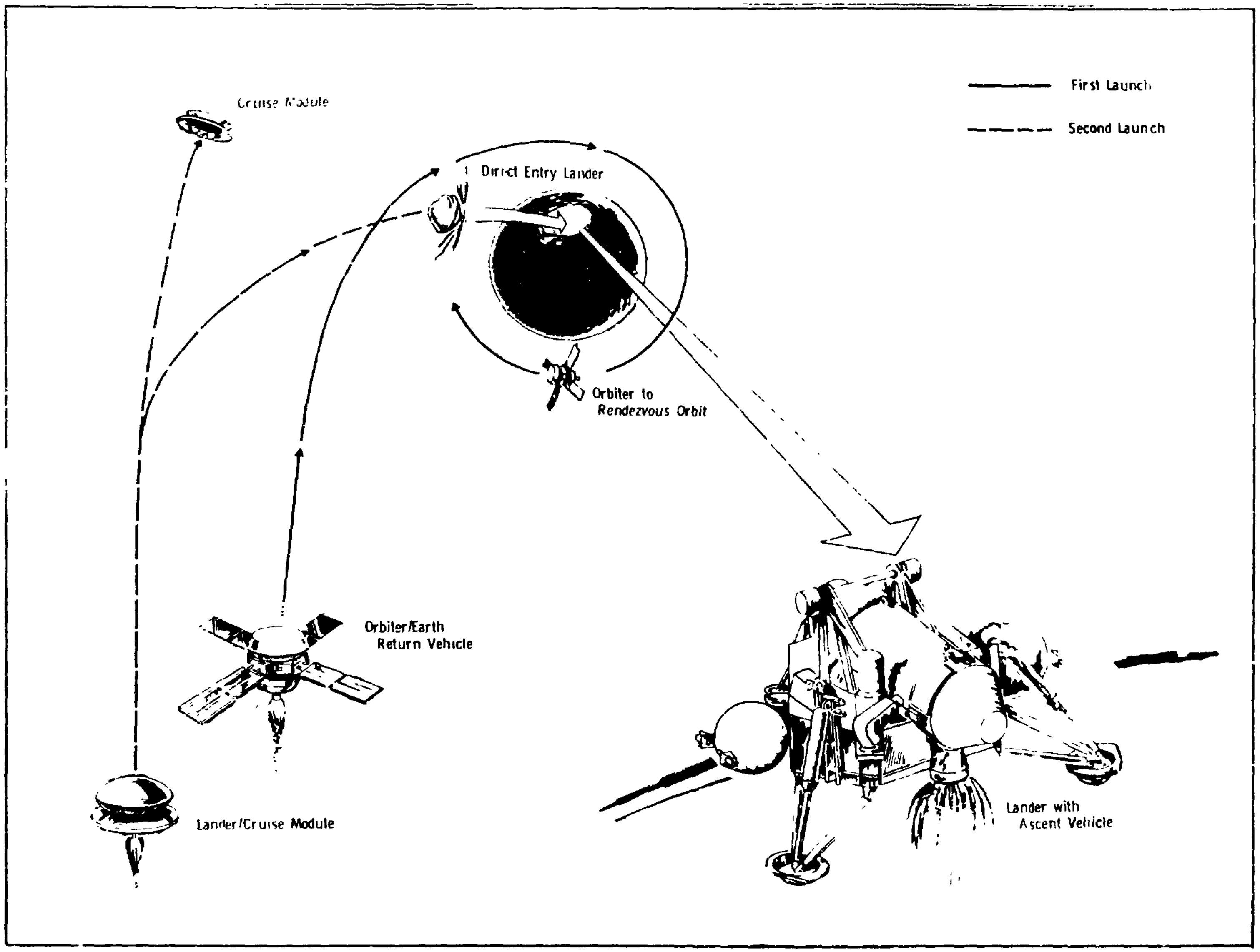

Figure III-1 MSSR Mission Sequence - Mars Rendezvous Mode--Dual Launch, Direct Entry

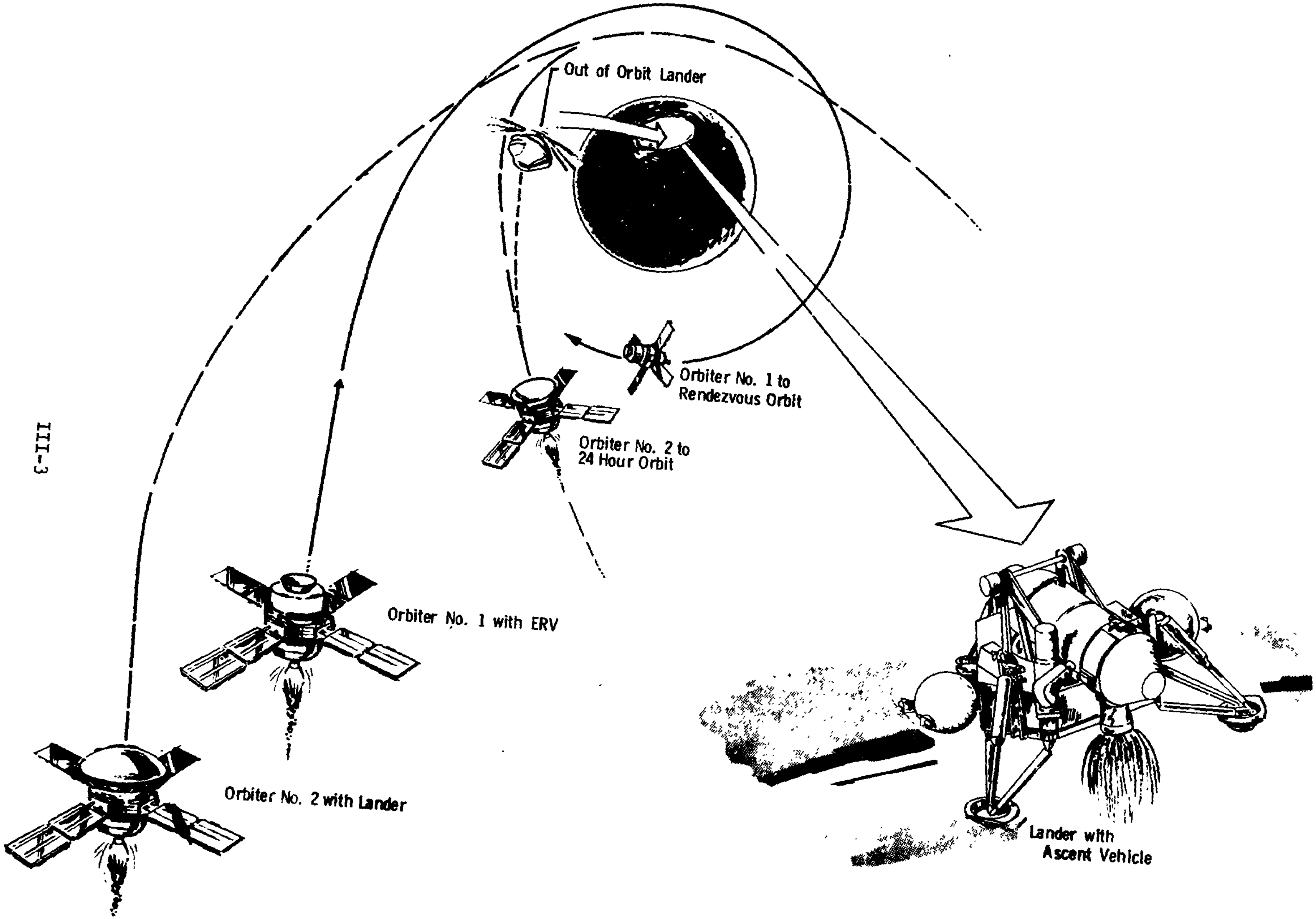

Figure III-2 MSSR Mission Sequence - Mars Rendezvous Mode--Dual Launch, Out-of-Orbit Entry

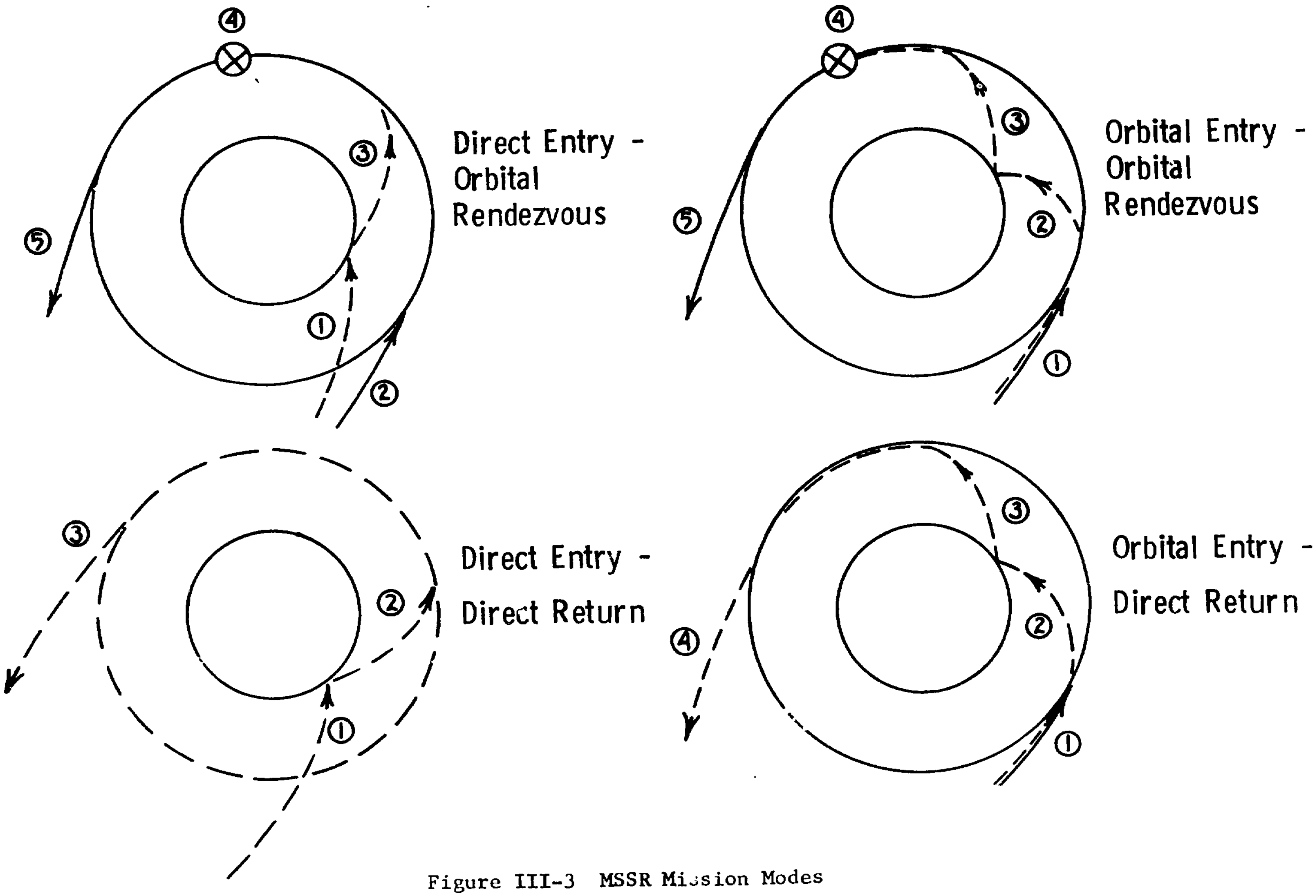

Figure III-3 MSSR Mission Modes

avoiding the complexities of an automated Mars rendezvous. These methods do, however, have a dramatic impact on total spacecraft weights and also make the control of back contamination more difficult.*

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*A MSSR science workshop was conducted at NASA Headquarters on June 11 and 12, 1974 at which the Mars orbital rendezvous mode was endorsed as the favored approach from the standpoint of controlling back contamination.

IV STUDY GUIDELINES

The focus of this study as established by the JPL Technical Manager, J. W. Moore, was to consider the Mars orbital rendezvous mission mode and to then examine in detail the phases of that mode that appear to offer the greatest technical risk; namely, the Mars ascent, rendezvous, docking, and sample transfer functions. The logic was that if the Mars orbital rendezvous can be proven to be feasible and cost effective, then decisions that will define the recommended MSSR mission and estimates of program cost can be more readily developed.

The study approach was to perform a number of technical tradeoffs leading to the definition of a baseline spacecraft set and mission profile. The feasibility of the ascent, rendezvous, docking and sample transfer would then be tested within the framework of this baseline.

The 1981 launch opportunity was chosen for the baseline with the understanding that the mission and spacecraft designs should not be invalidated by the requirements of the 1983/84 opportunity.

Existing spacecraft designs and proven technology were to be used wherever possible in the baseline. Viking and Pioneer Venus spacecraft were considered particularly good candidates for application to the mission.

The sample size was to be in the 0.2 to 5.0 kg range. The baseline was subsequently sized for 1 kg. The impact on the baseline of a 5 kg sample was also evaluated.

Since this was a technical feasibility study, emphasis was not to be given to science strategies or the identification of science investigations that might enhance the basic MSSR mission. However, we did organize a one-day science seminar in Denver, Colorado, at which about 12 members of the planetary science community developed science guidelines for the mission. (See Appendix A to Volume II of this report.) These guidelines are summarized in Table IV-1.

Table IV-1 MSSR Science Guidelines

Sample Use	Sample Amount (per site)	Sample Type	Sample Site	Sample Control
Inorganic Analysis	100 grams	Sieved to 2-10 mm Size; Surface & Trench	Mouth of Stream Bed	Sealed; Temp. < Mars Surf. Max.
Organic Analyses	200 grams	Fines; Surface & Trench		Sealed; Temp. < Mars Surf. Max.
Biological Analyses	200 grams	Fines; Surface		Sealed; Temp. < $0^{o}C$
Pathogenicity	300 grams			
Reserve	200 grams			

V BASELINE MISSION/SPACECRAFT DESCRIPTION

The baseline mission profile chosen to test the feasibility of the Mars ascent, rendezvous, docking and sample transfer was the single launch, direct entry mode illustrated in Figure II-1 and described in Chapter II of this volume. This baseline was selected because it allows the most direct use of existing hardware and technology and therefore is probably the lowest cost mission concept if implemented in the near future. The use of existing systems does, however, restrict the available hardware weights and margins. It was important, therefore, to make certain that weight restrictions were not forcing difficult or unrealistic solutions to ascent, rendezvous, docking and sample transfer problems and consequently clouding an objective evaluation of feasibility. In other words one test applied to each design decision incorporated into the baseline was "could this function be performed significantly better or more reliably if more weight could be added to it?" Except for the obvious approach of adding more and more redundancy, the baseline has not had to sacrifice performance because of weight restrictions to any appreciable degree.

The results of this study should not be interpreted necessarily as a recommendation that this baseline is the optimum MSSR mission mode. Rather the study takes the position that this baseline offers as good a test of the feasibility of the Mars orbital rendezvous mode as any other mission approach.

A. MISSION PROFILE

Moving from the generalized illustrations of mission sequences in Figure II-1 to more detailed descriptions, Figure V-1 shows, in approximately true relative scale, the functions of the direct entry lander and the MAV. The approach deflection maneuver occurs after the lander has separated from the orbiter at four hours (53,500 km) prior to what would have been the closest approach point on a flyby trajectory. The deflection maneuver requires about 84 mps velocity change (ΔV) to insert the lander into a 4° entry corridor ($\pm 2^{\circ}$ about nominal). The 4° corridor was chosen to minimize approach guidance accuracy requirements and can be achieved with DSN tracking alone (no on-board optical navigation aids required).

The lander will begin to sense the Mars atmosphere at approximately 244 km altitude at which time it will have an entry velocity of 5785 mps (18981 fps).

After the landing at Step 2 in Figure V-1, approximately 11 days have been allowed for landed operations in the baseline mission profile.

The Mars landing site accessibility for the 1981 baseline mission is described in Figure V-2. This is a plot of accessibility as constrained only by spacecraft performance capability (i.e., Earth command link or thermal constraints not considered) for a typical launch-encounter day combination. The most efficient MSSR Mars orbital rendezvous mission would locate the approach trajectory (and therefore the rendezvous orbit) and the departure trajectory in the same plane. In the case shown here this condition would restrict the incoming inclination to 43° and constrain the landing latitudes to a narrow band between 37°S and 39°S. The logical way to increase the landing latitude accessibility is to increase the performance capability of the Earth return vehicle so that it can perform a plane change from the rendezvous plane to the departure plane. If sufficient plane change ΔV were available in the ERV, the landing latitudes could be increased to a range of 85°S to 50°N (performance constraints only) for a typical launch-encounter day combination (see page V-25 for sources of added ERV performance).

The MAV launch from Step 3 to Step 4 is the only portion of the MAV flight profile that is not under Earth-based control. During this time the

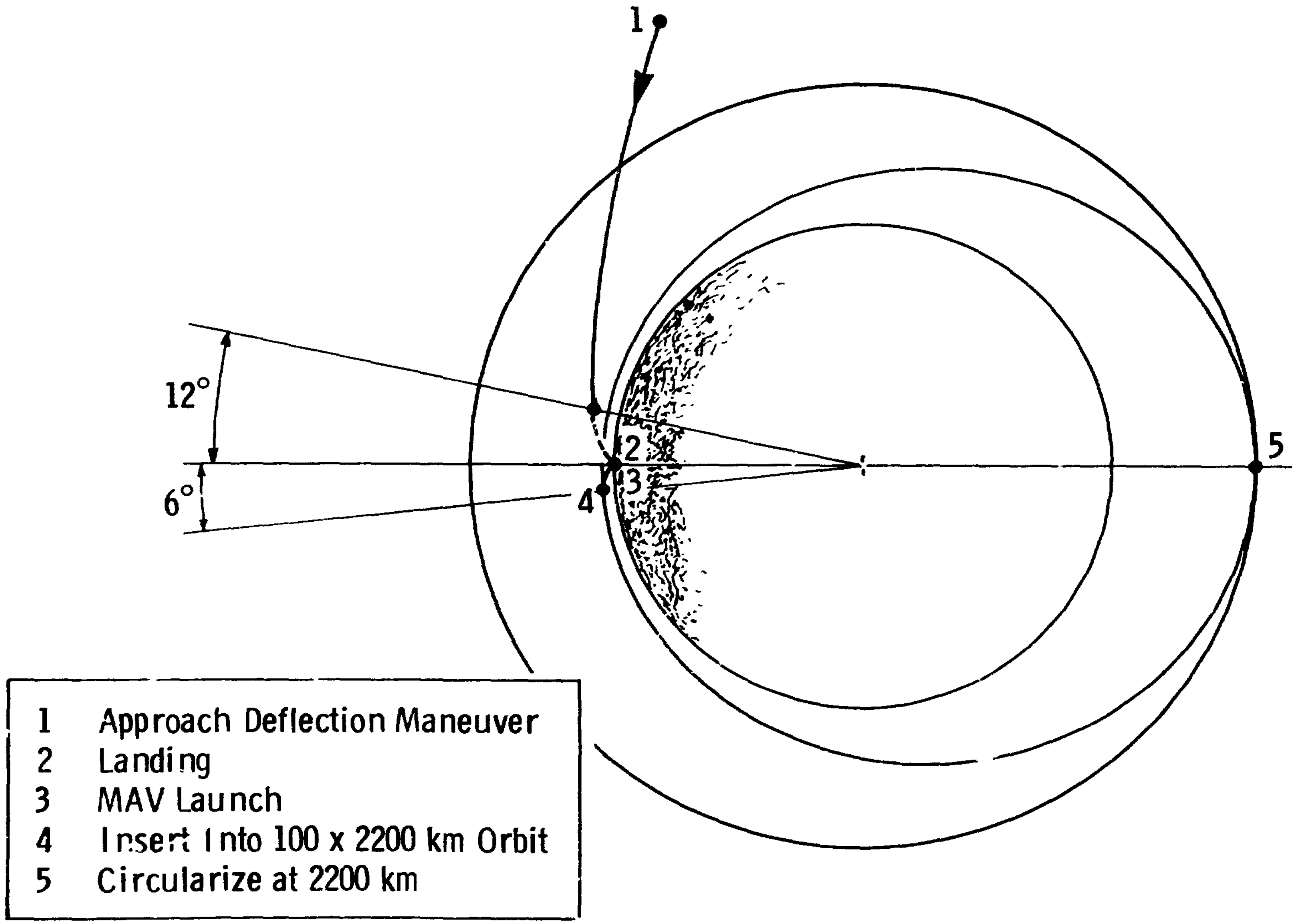


Figure V-1 MSSR Mission Profile at Mars - Lander/MAV

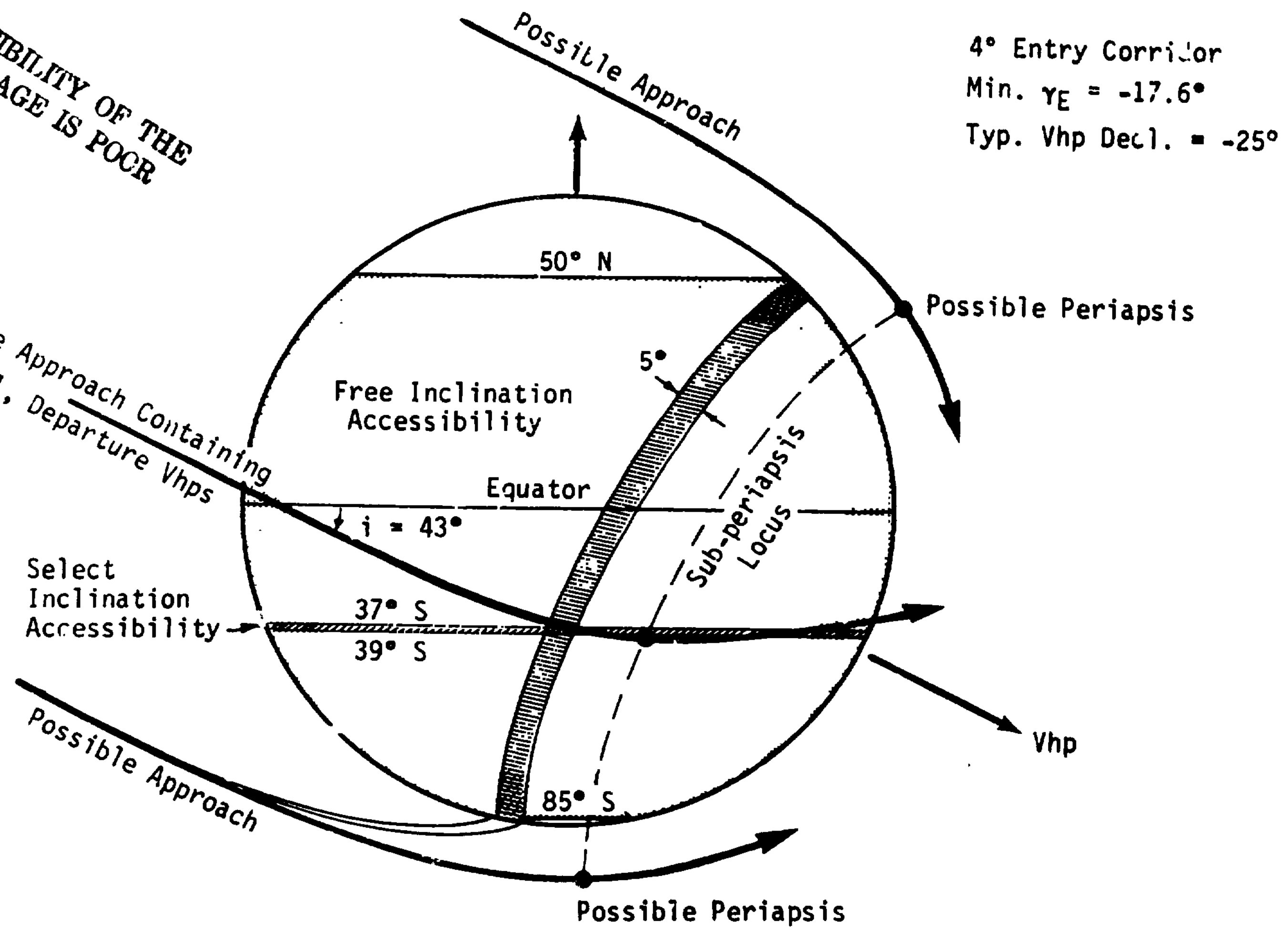


Figure V-2 Landing Site Accessibility

MAV ascends to an altitude of 100 km and inserts into an initial orbit of 100 km x 2200 km. The only real accuracy requirement for this orbit is that it is stable and predictable long enough to allow Earth tracking and a subsequent Earth commanded maneuver to circularize at the 2200 km apoapsis altitude. Stability and lifetime analyses have been made of this low orbit and the required predictability appears to be achievable. These analyses used Viking '75 atmosphere models for the drag terms and Mariner 9 gravity coefficients. Of course, local gravity anomalies are not known for Mars at this time, but it is felt that the orbit determination accuracies required to command a circularization burn to get the MAV safely away from local mascon effects, can be obtained.

After MAV circularization the vehicle is tracked from Earth again and a trim maneuver computed to correct unacceptable dispersions from the desired 2200 km circular (or higher) rendezvous orbit. Essentially the strategy is to let the MAV remain in whatever circular orbit it can achieve and then bring the orbiter down to that orbit.

Figure V-3 shows the sequences followed by the orbiter, ome of which will have been carried out during the same time period of the previously described lander and MAV functions.

The initial capture orbit is a large loose ellipse with a low periapsis altitude (1000 x 100,000 km). This orbit was chosen to minimize the initial MOI ΔV and provide a high apoapsis (low velocity) at which any required orbital plane changes can be made economically. The plane change can be used to adjust the rendezvous orbit plane to a better relationship with the Earth return trajectory, or to adjust the orbiter plane closer to the MAV orbit plane after MAV ascent.

The 5-day period of the initial orbiter orbit has been analyzed for lifetime and will not impact the planet during the approximately 50-year period required by international Mars quarantine protocols.

After the MAV has been put into the rendezvous orbit, the orbiter is brought down to that orbit by a series of maneuvers that are basically no more complex or demanding than those performed by the same vehicle in the Viking '75 program.

The first descent maneuver, at point 4, involves raising the orbiter periapsis to an altitude of 2250 km (50 km above the MAV orbit). This adds a ΔV of 22 mps nominally to the 1098 mps required for MOI.

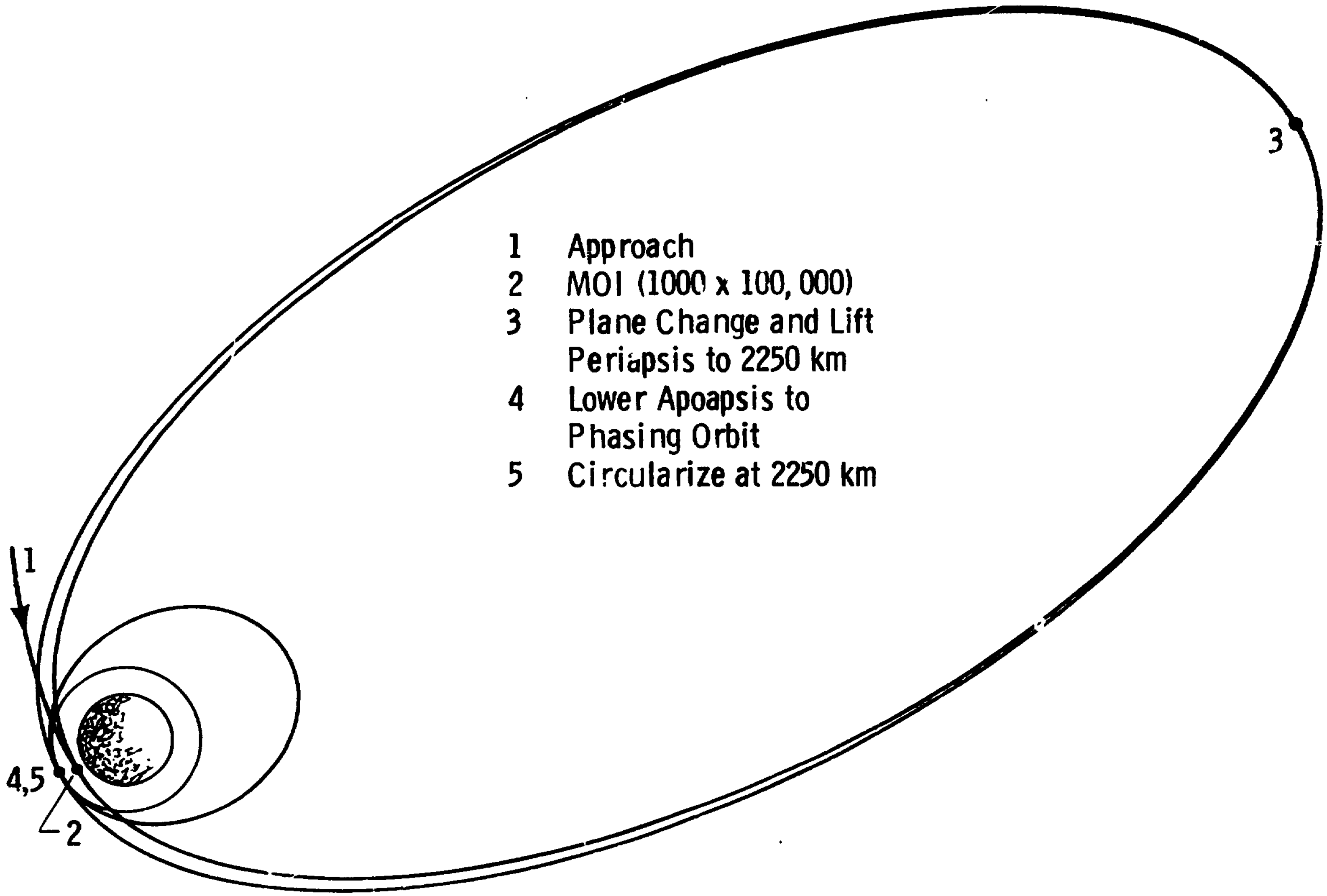


Figure V-3 MSSR Mission Profile at Mars - Orbiter

The next step is to lower apoapsis to an Earth-calculated phasing orbit altitude. The phasing orbit puts the orbiter and MAV in the proper time relationship in their respective orbits so that when the subsequent circularization maneuver (Step 5) brings the orbiter to the appropriate pre-rendezvous orbit, the two vehicles will be approximately 45^o apart.

After the orbiter has been circularized at 2250 km altitude, the MAV and orbiter are tracked, this time using a more accurate ΔVLBI technique. ΔVLBI is an interferometric data type in which both vehicles are simultaneously tracked by two DSN stations and the data double-differenced. This technique will be demonstrated in Viking '75 and Pioneer Venus '78.

A very key feature of the Earth-controlled portion of the rendezvous strategy used in this study is the accuracy with which the location of the vehicles (MAV and orbiter) can be determined. Using conventional DSN doppler tracking the individual vehicle positions can be determined to within approximately 3 km and velocities to within 1.5 mps. With the ΔVLBI tracking technique, relative vehicle positions can be determined to within 0.3 km and relative velocities to within 0.15 mps.

Figure V-4 shows the relative positions of the orbiter and MAV at the completion of the initial rendezvous sequences which is also the end of the Earth-controlled portion. The MAV is in the nominal 2200 km circular rendezvous orbit and the orbiter is 50 km higher and, at the completion of its final Earth-controlled trim maneuver, is 3.4^o ahead of the MAV. The difference in periods of these two orbits (3.528 hrs vs 3.575 hrs) is such that the MAV will "creep up" on the orbiter at a rate of approximately 1.35^o per hour.

Figure V-5 amplifies the relative positions shown in Figure V-4 and summarizes the results of an extensive navigation simulation that was one of the major features of this study effort. It shows that the predicted relative dispersions from the nominal 50 km in altitude and approximately 340 km down track, are contained in a rather small ellipsoid approximately 142 km x 16 km x 52 km in size (3-sigma).

The simulation that produced these predicted dispersions was built around the following features: 1) a maneuver and timing strategy that made conservative allowances for tracking and occultation periods, Earth-based

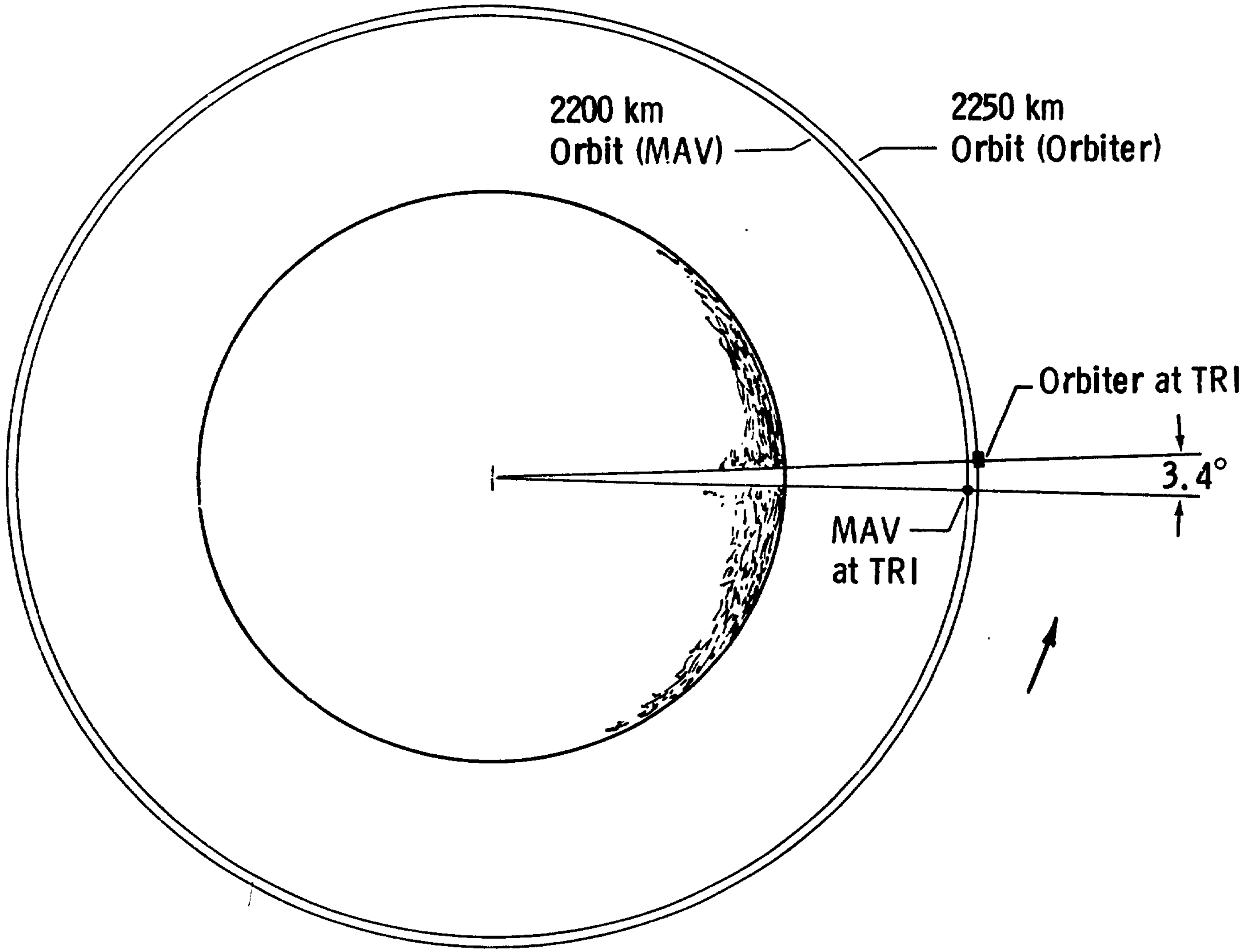


Figure V-4 MAV/Orbiter Positions at Terminal Rendezvous Initiation (1)

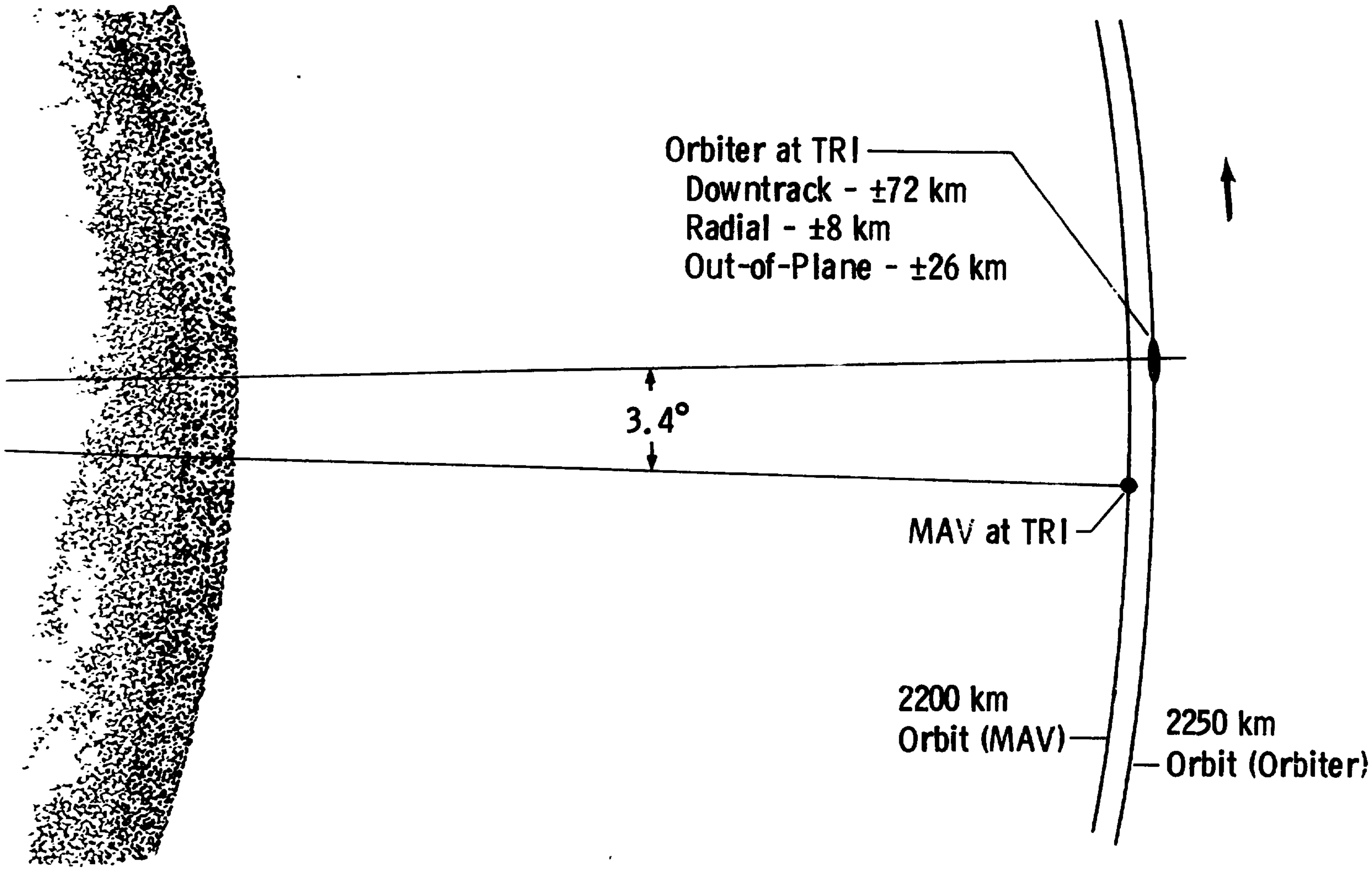


Figure V-5 MAV/Orbiter Positions at Terminal Rendezvous Initiation (2) data reduction and command calculations, and vehicle reorientations; 2) proven DSN doppler tracking accuracies and predicted ΔVLBI capabilities; and 3) demonstrated or conservatively predicted vehicle execution errors. The simulation was constructed so that the sensitivity of the vehicle dispersions and the required propellant budgets to correct them (ΔV_{stat}) could be measured in terms of the assumed error and uncertainty sources.

At the completion of the initial rendezvous phase, accomplished under Earth control, the MAV will be within range of the orbiter rendezvous radar (maximum range of the radar sensor is 750 km) and the relative positions will be accurately enough known to command them to point at each other well within the beamwidths of the orbiter radar and the MAV transponder.

Details of the terminal rendezvous, docking and sample transfer phase, in which the two vehicles are brought together by on-board control, will be discussed in the next chapter.

Figure V-6 summarizes the sequences in the baseline mission profile after sample transfer. The orbiter and the Earth Return Vehicle, now carrying the sample canister, will remain in the 2200 km circular orbit for the approximately 400 days required for the Earth return geometry to be established. The sequences the ERV will follow in maneuvering to the Earth return trajectory are essentially the reverse of those performed by the orbiter to reach the rendezvous orbit. After raising apoapsis to 100,000 km and lowering periapsis to 1000 km the ERV is in an efficient energy state to transfer to the trans-Earth trajectory with a burn at periapsis.

The mission profile sequences for landing site targeting, entry and recovery at Earth are described in Chapter VII.

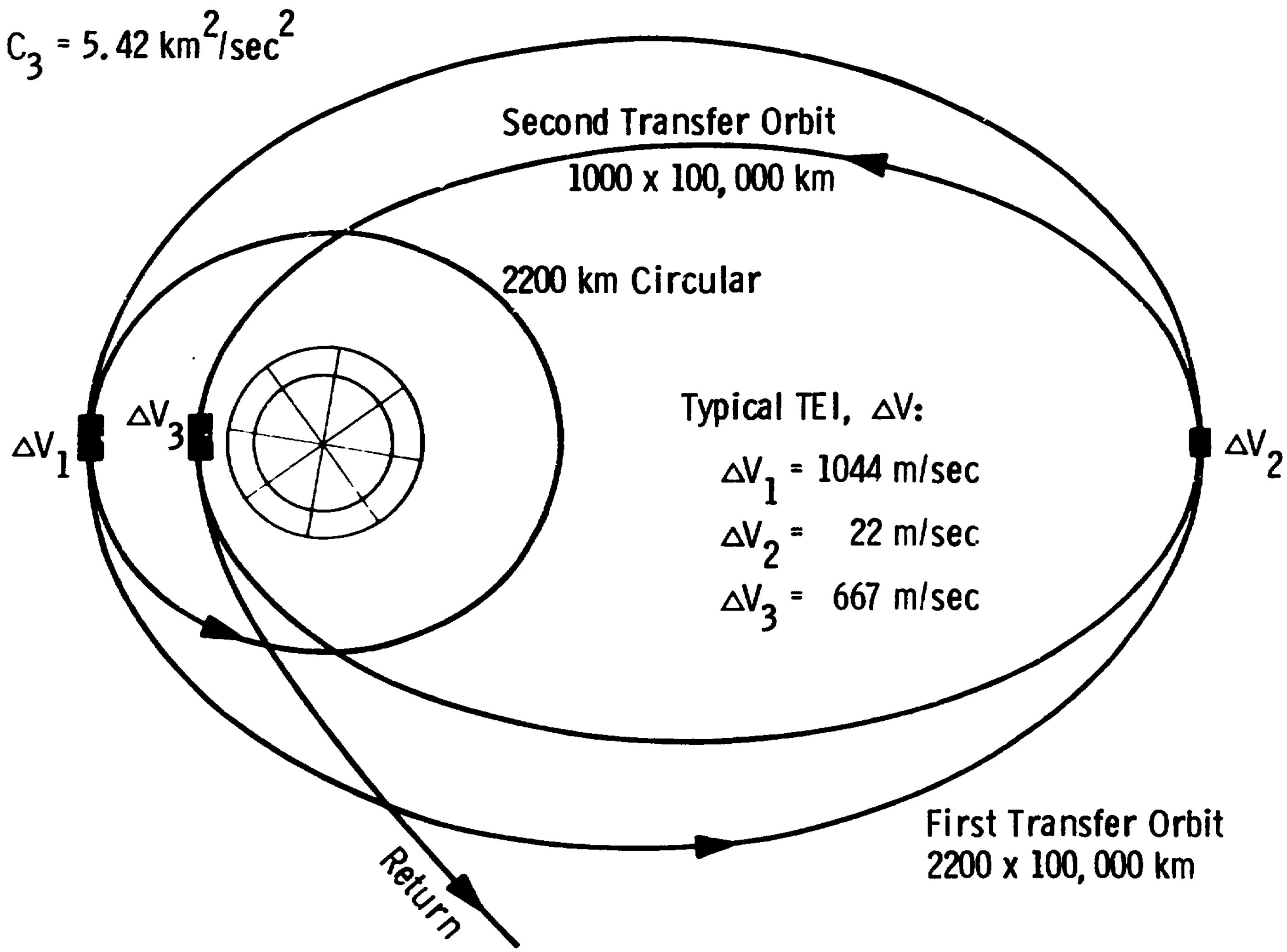


Figure V-6 TEI (Earth Return) Profile for 1981 MSSR

B. BASELINE SPACECRAFT

The total spacecraft for this baseline MSSR mission comprises five separately functioning vehicles. The spacecraft in its Earth launch configuration is diagramed in Figure V-7.

The total spacecraft weight at launch will be 4409 kg distributed as shown in Table V-1. This compares with an equivalent Viking '75 launch weight of approximately 3500 kg. The overall spacecraft length will be approximately 180 cm (71 in) longer than the Viking '75 launch configuration (6.92 m vs 5.12 m). The dynamic envelope within the Titan IIIE Centaur shroud will be adequate without modification.

Three out of the five MSSR spacecraft vehicles will have been proven prior to their application to this mission. The orbiter is a minimally modified Viking '75 orbiter with the propellant tank capacity increased by approximately 15%. The rendezvous radar is the only new subsystem added. The rendezvous radar has been designed to parallel the performance characteristics of the proven Apollo system. A comparison of the two is shown in Table V-2.

With deletions of unneeded equipment, the MSSR orbiter dry weight becomes 792 kg compared to the equivalent mass of 918 kg for the Viking '75 configuration. A summary of the orbiter mass derivation from Viking '75 is outlined in Table V-3.

The Earth Return Vehicle has not been studied in detail but for this baseline is assumed to be a modified Pioneer Venus spin-stabilized orbiter. A major objective of the modification from the Venus configuration will be the reduction of dry weight and the addition of a bipropellant propulsion system capable of providing the required ΔV of approximately 1800 mps.

The lander capsule will be a modified Viking '75 lander that integrates the MAV. Figure V-8 shows the impact of the MAV integration on the lander capsule indicating the 59 cm increase in clearance under the parachute canister that must be provided compared with Viking '75. This will necessitate the redesign of the parachute canister truss, the aeroshell aft body, and the bioshield base. The heat shield and supporting structure must also be increased to accommodate the increased entry weight and the direct entry mode.

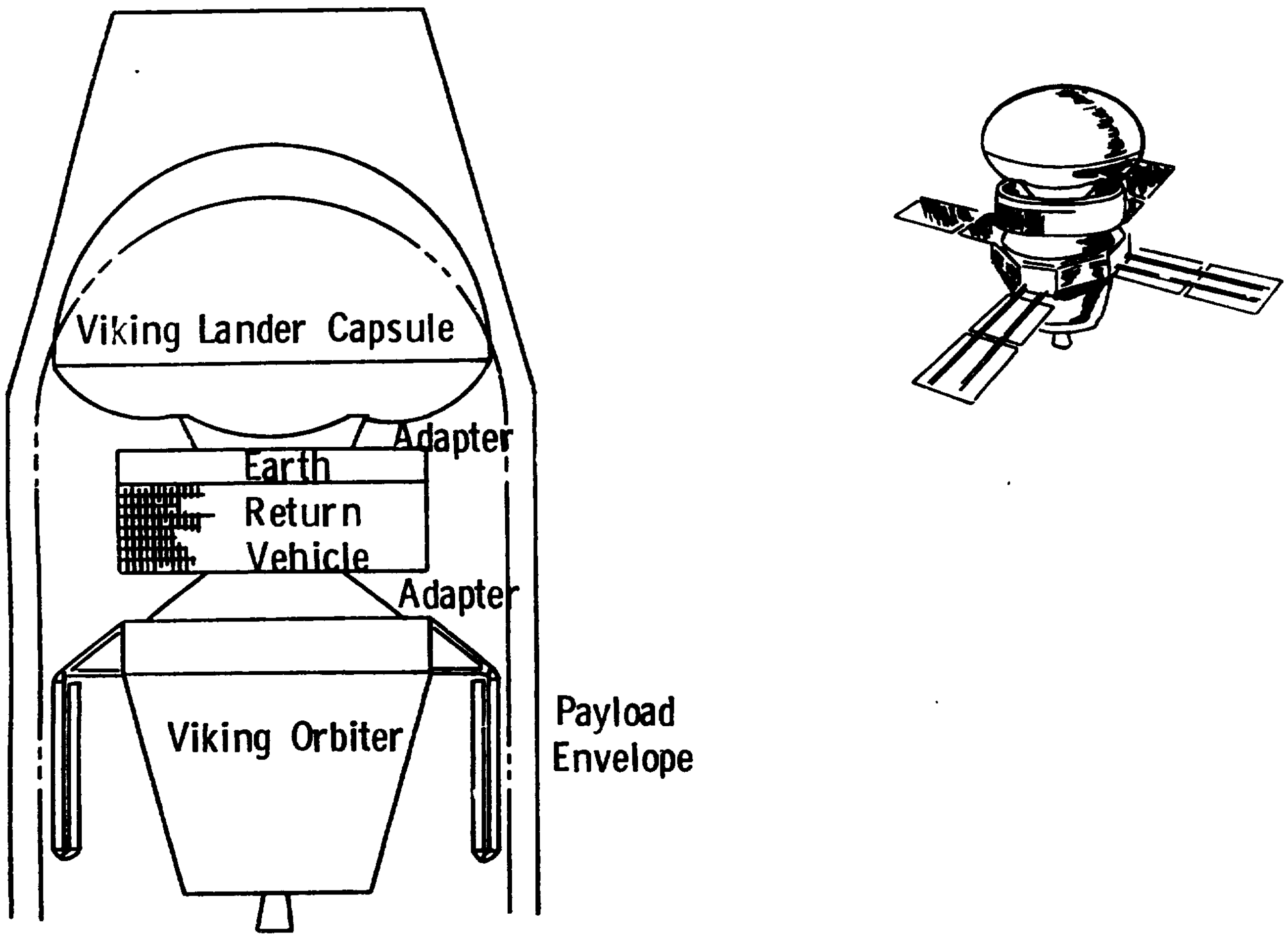


Figure V-7 Earth-Launched Payload

Table V-1 Masses of Major Elements - 1981 Baseline (kg)

Mars Orbiter	2818	Mod VO'75 Bus	530				
		Propulsion Inerts	262				
		ERV (Earth Return Vehicle)	261			ERV Spacecraft	105
						Earth Entry Capsule	26
						Propellant	130
		Propellant (incl. Midcourse)	1631				
		Margin	134				
Mars Lander System	1285	Entry Vehicle	1205	Aeroshell	242		
				Parachute	116		
				Landing Prop	71		
				Lander	776	Mod Lander	446
						MAV	290
						Launcher	40
		Deorbit Prop & Misc.	80				
Items Jettisoned	306	Adaptors					
		Bioshield					
		Launch Vehicle Peculiar					
	4409	Throw Weight					

Table V-2 Comparison of Apollo and URDMO Rendezvous Sensors

System Parameter	LM/CSM Rendezvous	Orbiter/MAV Rendezvous/Docking
Frequency	X-Band	S-Band
Radar Type	CW	CW
Radar Mode	Automatic and Manual	Automatic
Modulation	PM: 3 Tones	PM: 819 kc - Subcarrier 4 Minor Tones
Radar Power	0.3 W (Solid State)	0.3 W (Solid State)
Maximum Range	750 km	750 km
Minimum Range	24 m	3 m
Radar Antenna	Cassegrain	Travelling Wave Array
Angle Track Method	Amplitude Monopulse	Phase Monopulse
Transponder Power	0.3 W (Solid State)	0.15 W (Solid State)
Transponder Antenna	Horn	Cassegrain
Coherence Ratio	240/241	220/239

Table V-3 URDMO Orbiter Mass Derivation

	Kilograms
Viking Orbiter (Dry)	926.9
Remove Science	-87.0
Remove Scan Platform	-32.8
Remove Data Storage	-31.4
Remove Cold Gas RCS (incl. Gas)	-46.2
Add Combined Rendezvous/RCS (Dry)	+22.8
Tank Stretch	+23.9
Docking Cone and Rendezvous Radar	+15.3
URDMO Orbiter (Dry)	791.5
RCS Propellant	5.0
Rendezvous Propellant	33.0
Main Propulsion Propellant (1 kg Sample Case)	1542.0
Midcourse Propellant	51.0
URDMO Orbiter Loaded	2422.5
RECAP (DRY WEIGHT)	
Orbiter Bus	529.5
Propulsion Inerts	262.0

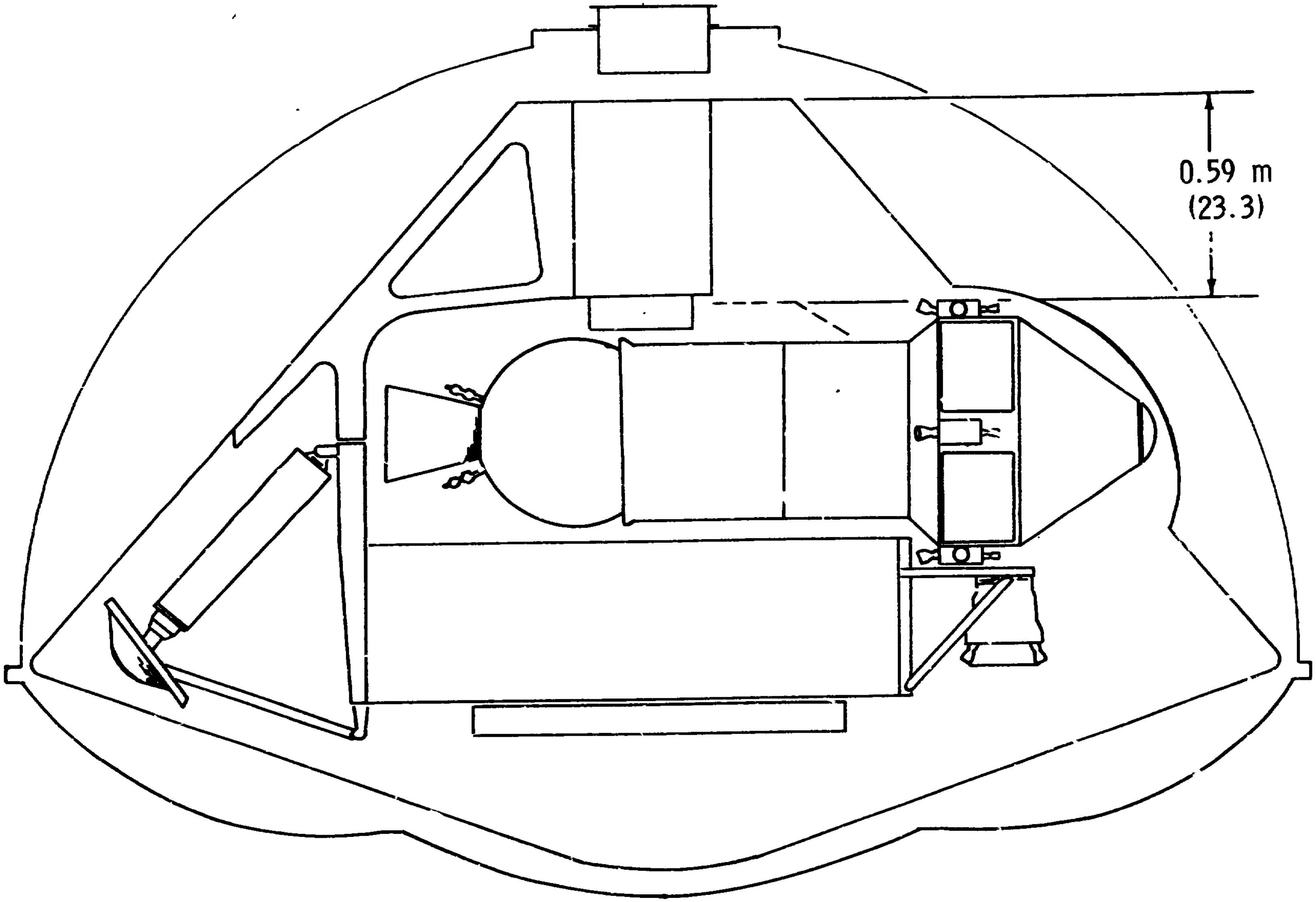


Figure V-8 MAV Impact on Viking Lander Capsule

Aerodynamic analyses conducted during this study indicated that the lander capsule shape and mass properties will provide for a stable entry and safe heating conditions.

The required modifications to the Viking '75 lander in the landed configuration are summarized in Figure V-9. The Viking '75 landed weight of approximately 594 kg as shown on the left, is reduced to approximately 485 kg as indicated in the center sketch, and then increased to 776 kg with the addition of the MAV as seen on the right. The details of this weight derivation are indicated in Table V-4.

The MAV launcher is mounted on the lander equipment plate and provides 360° of azimuth rotation and 79° of elevation.

The change to the lander that accounts for most of the increased landed weight capability is the addition of a regulated pressurization system for the terminal propulsion subsystem which allows the engines to operate at full thrust throughout their burn time.

The MAV is the only completely new vehicle in the baseline concept developed in this study. As seen in Figure V-10, it is a small combined launch and orbiting vehicle. Its sole purposes are to carry the sample to the rendezvous orbit and to participate, in a semi-passive way, in the rendezvous, docking, and sample transfer operations.

Propulsion consists of two stages of sterilizable solids to achieve the initial 100 x 2200 km orbit and a third monopropellant hydrazine stage for thrust vector control, circularization at 2200 km, and final rendezvous orbit trim.

Power is provided by two deployable solar panels charging a nickel-hydrogen battery.

A single dual-frequency ratio S-band transponder supports both the Earth-based tracking link and the orbiter-to-MAV rendezvous radar link.

Guidance and control features a simple open loop rate gyro system for ascent trajectory control and a Sun-Earth referenced system for on-orbit operations.

The weight limitations on the MAV and particularly on its third stage are the most critical in the entire baseline spacecraft. The multiplying

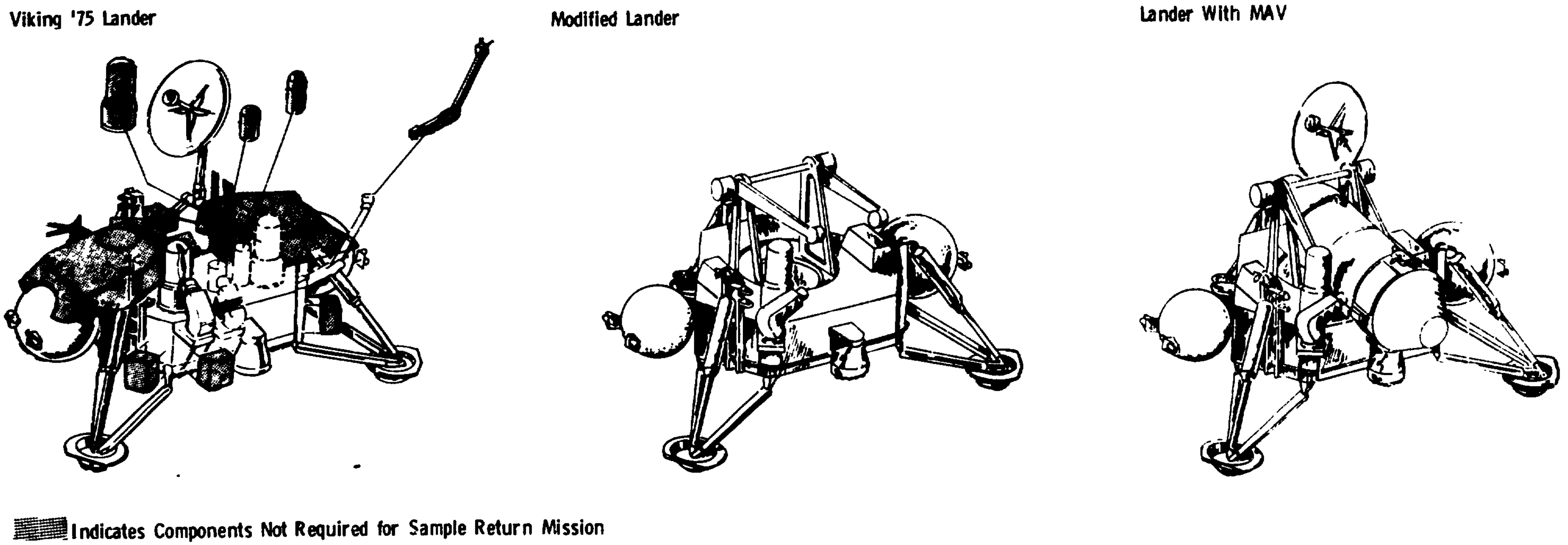
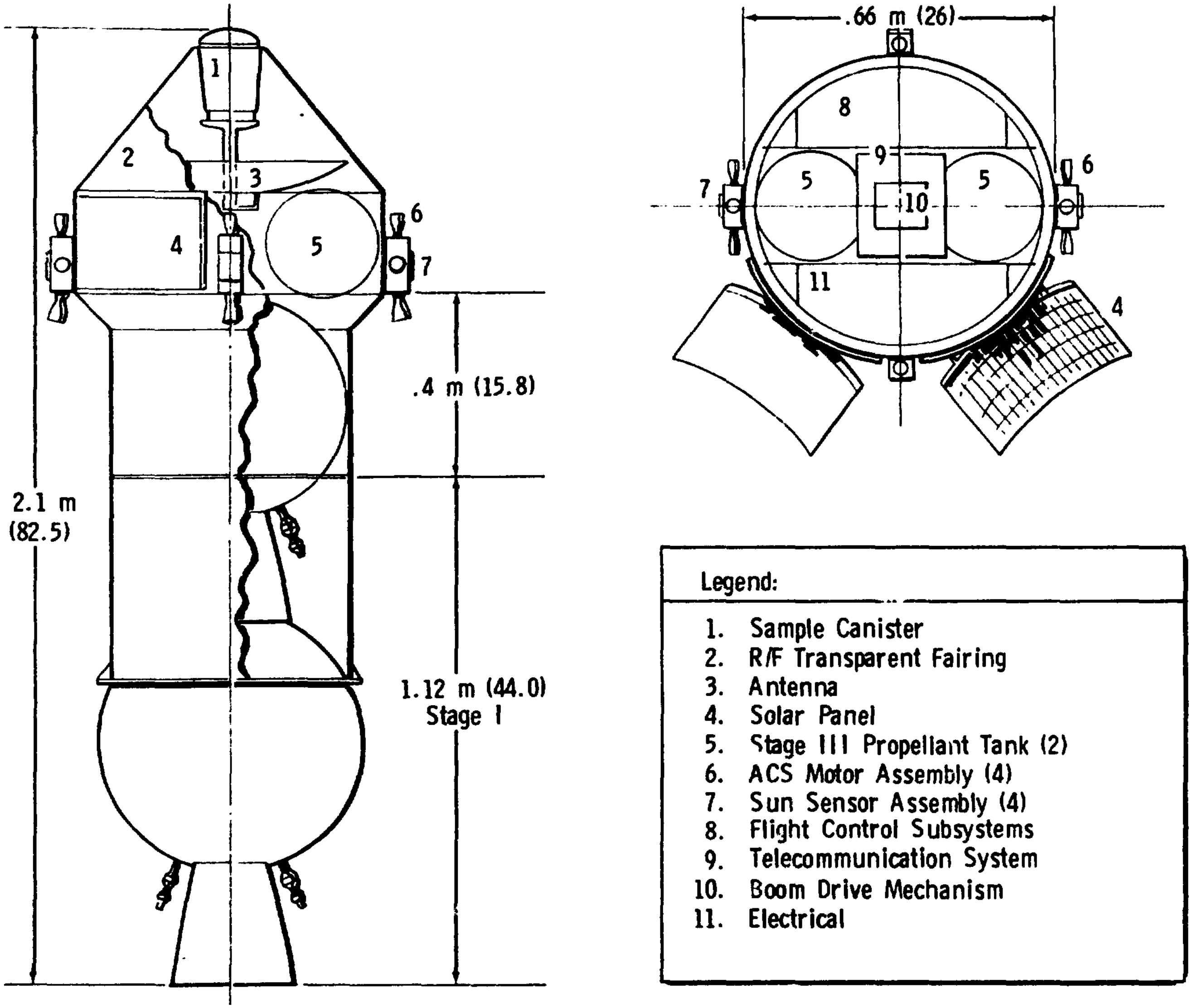


Figure V-9 Lander Modifications

Table V-4 URDMO Lander Mass Derivation

	Kilograms
Viking Lander - Landed (2/19/74)	594.2
Remove UHF	- 5.85
Reduce RTG Size	-22.54
Remove One Battery (1/2 Package)	-11.47
Remove Data Storage	-13.83
Modify Thermal System	- 5.35
Remove Science (except one camera & soil sampler)	-60.55
Add Regulated Pressure System	+ 1.45
Modify Telemetry	- 6.58
Modify S-Band to MAV Components	-15.15
Remove Cabling	- 9.98
Beef-up Landing Struts	+ 2.07
Add MAV	+288.93
Add MAV Launcher (incl. Thermal Protection)	+ 41.05
URDMO Landed	776.40



Legend:

1. Sample Canister
2. R/F Transparent Fairing
3. Antenna
4. Solar Panel
5. Stage III Propellant Tank (2)
6. ACS Motor Assembly (4)
7. Sun Sensor Assembly (4)
8. Flight Control Subsystems
9. Telecommunication System
10. Boom Drive Mechanism
11. Electrical

Figure V-10 Mars Ascent Vehicle

factor between MAV launch weight and MAV third stage weight in orbit is approximately 10. This means that any excess third stage weight has a very costly impact on the rest of the spacecraft. Table V-5 summarizes the MAV weight breakdown.

The sample canister is mounted in the nose fairing of the MAV and is a single-seal unit with self-contained opening and closing actuator as shown in Figure V-11. This particular canister concept assumes the sample will be a single bulk loading into the drawer-like inner container. Future requirements could lead to the possibility of segregating and separately sealing samples taken from a number of different sites.

Table V-5 290 Kilogram MAV Mass Summary (kg)

Stage III		
Structure & Mechanism	8.85	
Equipment	9.39	
Propellant Inert (incl. residual)	11.29	
Contingency 10%	2.90	
Propellant	8.30	
Total Step 3		40.73
Sample		1.00
Stage III at Liftoff		41.73
Stage II		
Skirt	3.95	
Propulsion Inert	11.11	
Propellant	81.55	
Total Step 2		96.61
Stage II at Liftoff		138.34
Stage I		
Skirt	5.67	
Propulsion Inert	17.51	
Propellant	128.41	
Total Step 1		151.59
Stage I at Liftoff		289.93

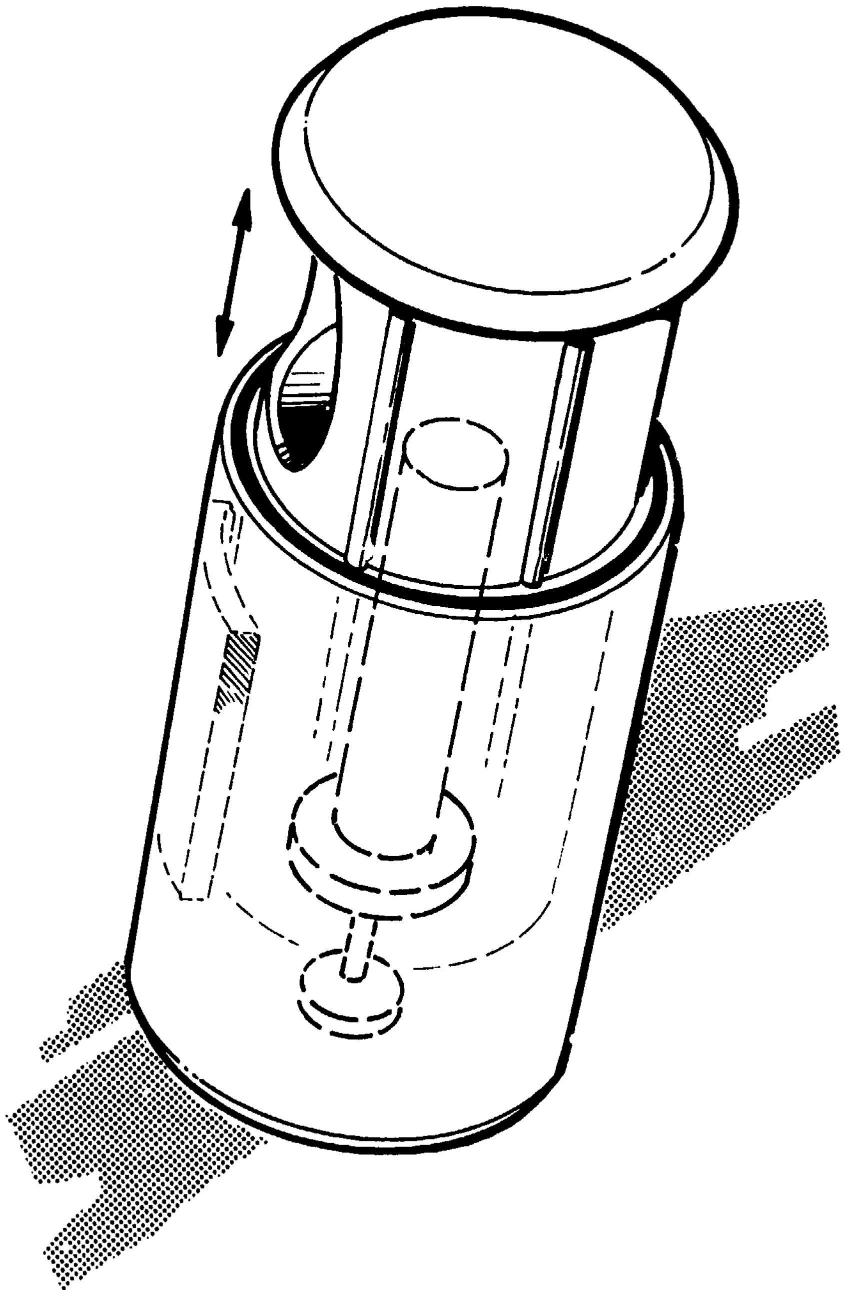


Figure V-11 Sample Canister Concept

C. MAJOR SYSTEMS LEVEL TRADES

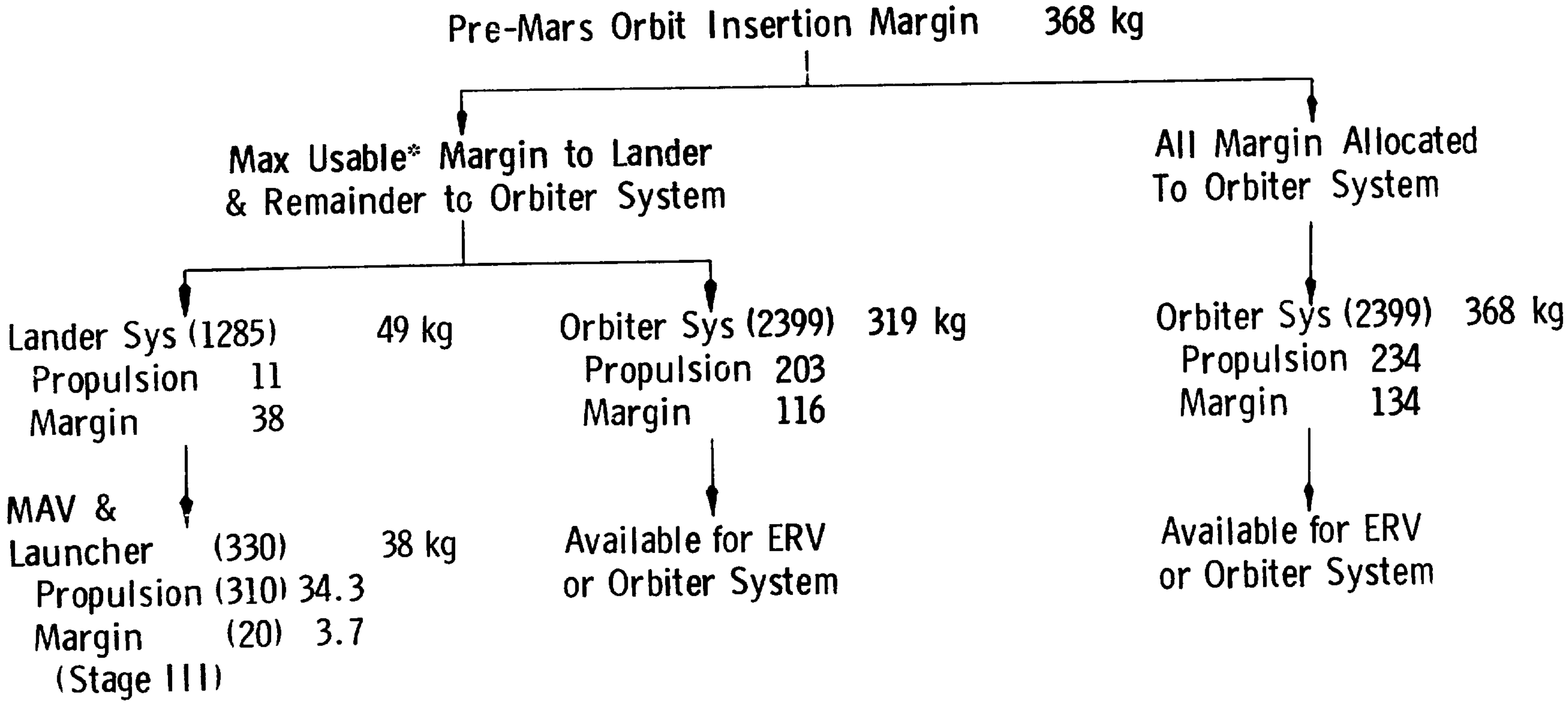
With four separately functioning flight vehicles and almost three years of mission operations, it is not surprising that a large number of configuration and mission profile alternatives are available in selecting a baseline MSSR system. During the course of this study, trade studies were conducted on such options as spin vs three-axis MAV stabilization, solid vs liquid MAV propulsion, and circular vs eccentric orbit rendezvous. Three other comparative studies that have major impacts on the overall mission and spacecraft design involve distribution of performance weight margins, sample size, and mission opportunity.

Figure V-12 diagrams the possible distribution of the weight margin available in the baseline mission concept described in this report. It shows that 368 kg of unallocated mass exist prior to Mars orbit insertion. This amount could be put entirely into the orbiter/ERV in which case 134 kg would be available in orbit. This could be used to increase the performance of the ERV and thereby open up a wider range of accessible landing latitudes, for example.

Alternatively, mass could be added to the lander, up to the landed weight limits of the Viking '75 parachute, and achieve a landed weight increase of 38 kg. If the 38 kg were added to the MAV it would increase the MAV payload in orbit by 3.7 kg. Such a lander increase would still allow an in-orbit mass increase of 116 kg.

Figure V-13 is a repeat of the MSSR baseline showing the impact on systems weights of increasing the sample weight from 1 kg to 5 kg. The most significant change comes in the mass of the MAV at liftoff which must increase by almost 50 kg to handle the extra 4 kg of sample. The landed weight of 830 kg shown can be handled by the Viking '75 system if the entry corridor is moved to a steeper nominal value or its width is reduced (probably by means of on-board optical guidance) from 4° to 2°.

Figure V-14 indicates one approach to modifying the Viking '75 orbiter to handle the increased performance requirements of the 1983/84 mission opportunity compared with the mods required for a 1981 launch. The 1981 mission requires approximately a 15% stretch over Viking '75 while the 1983/84 opportunity calls for a 35% stretch, and increased launch vehicle capability.



*Based on Viking '75 Lander as modified for baseline URDMO (parachute dia, aeroshell dia, terminal engines, and lander body size not changed).

( ) Current mass estimate without margin; however, a 10% contingency is included in new hardware estimates.

Figure V-12 URDMO Margin

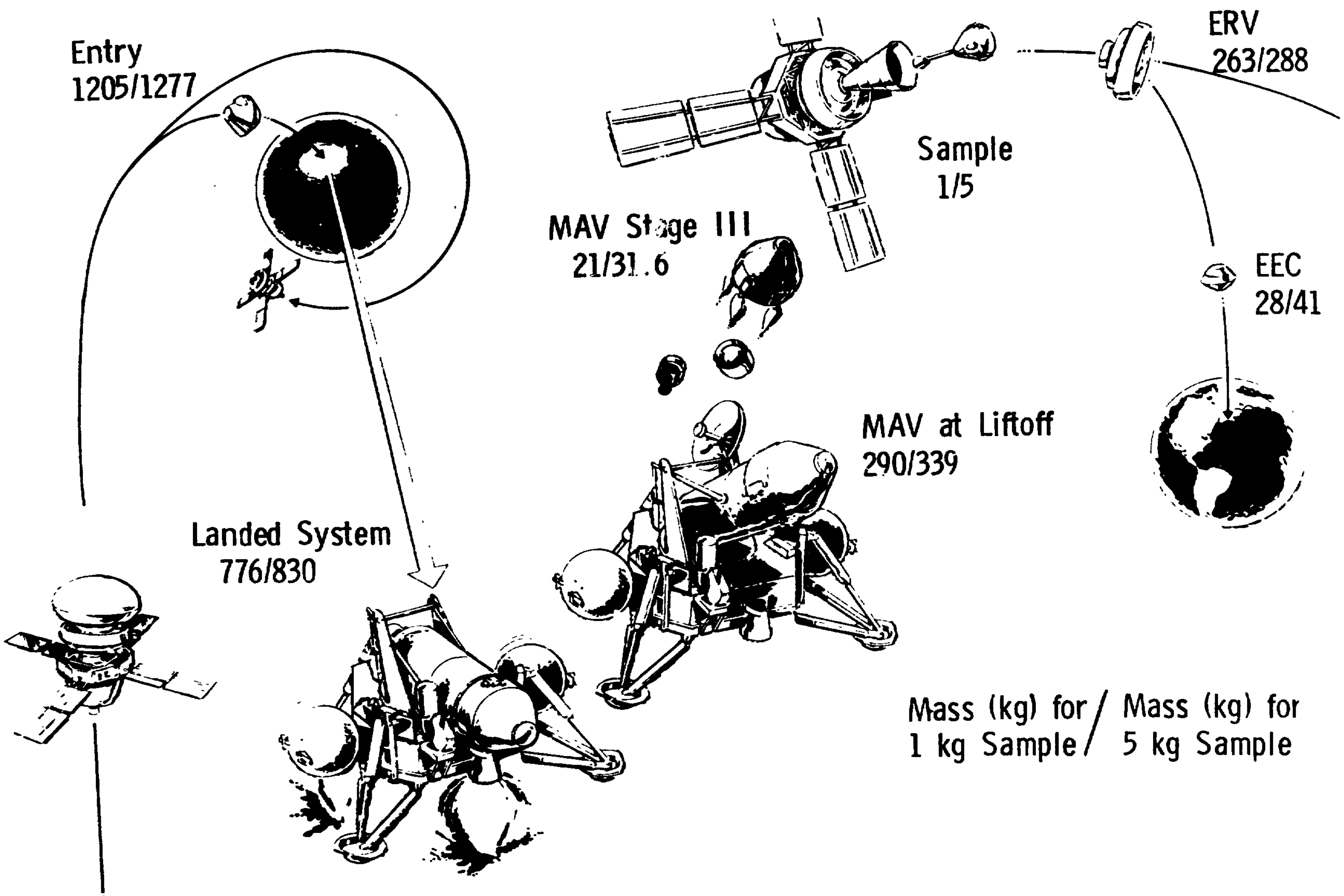


Figure V-13 Impact of Sample Size on System Design

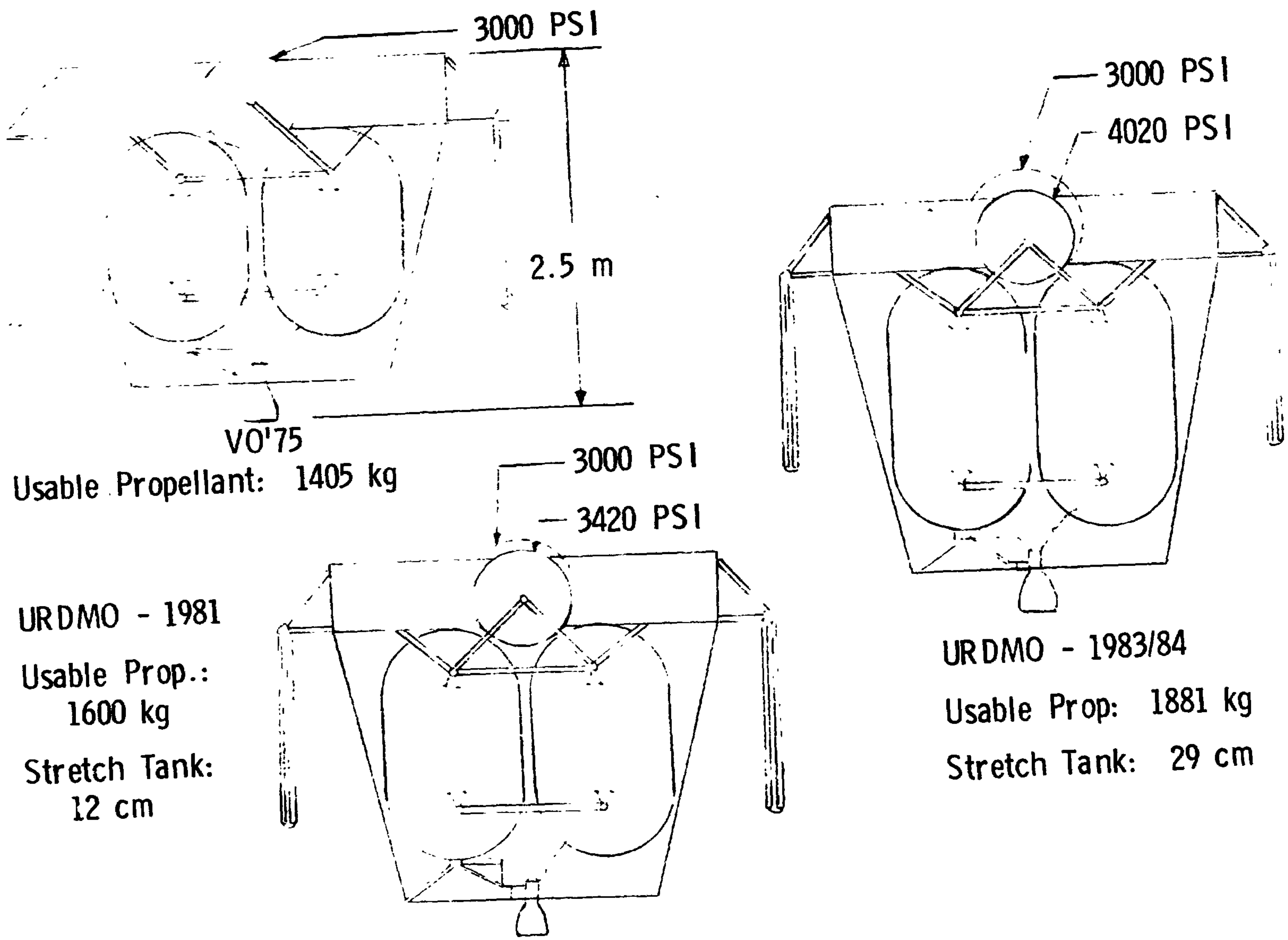


Figure V-14 Orbiter Propulsion Options

VI MARS ASCENT, RENDEZVOUS, DOCKING, AND SAMPLE TRANSFER OPERATIONS

The primary objective of this study was to assess the technical feasibility of the Mars orbital rendezvous mode for MSSR. Of the missions sequences required to support orbital rendezvous, there are five that have not been performed under conditions equivalent to the MSSR mission and therefore were given special attention. They are 1) the ascent of the MAV from the Mars surface to the rendezvous orbit; 2) initial rendezvous, in which the orbiter is brought to the MAV orbit under Earth-based control; 3) terminal rendezvous, in which the orbiter closes on the MAV under the automatic control of the orbiter rendezvous radar; 4) docking, in which the orbiter and MAV are brought into physical connection; and, 5) sample transfer, in which the sample canister is handed over to the Earth return vehicle.

The ascent of the MAV requires that an orbit, within predictable tolerances, be achieved by a small, self-controlled vehicle, launched from a remotely pointed platform. Figure VI-1 identifies some of the important MAV ascent sequences and tolerances.

The position of the lander on Mars prior to launch is determined by Earth-based tracking and the orientation is sensed by the lander inertial reference system. The equired MAV azimuth and elevation angles are calculated on Earth and the launch is commanded into a preset clock system. The nominal baseline sequence requires an initial ramp angle of 54.8 $\pm$ 0.5 degrees.

The goal in the design of the MAV has been to keep its hardware as simple as possible and its performance tolerances as large as possible. In line with this approach, the first stage is controlled with a simple open-loop rate gyro guidance system to a constant pitch-over rate of 0.16 $\pm$ 0.004 degrees/second. This ascent trajectory approximates a gravity turn.

After 54.8 seconds of first stage burn, the vehicle coasts for 200.8 seconds before the second stage burn of 31.2 seconds injects it into the initial 100 x 2200 km orbit. After the insertion maneuver the vehicle acts on a prestored command that points it toward Earth to establish the Earth tracking and command links and one of the MAV pointing references. The other reference is the Sun, detected by the MAV sun sensors.

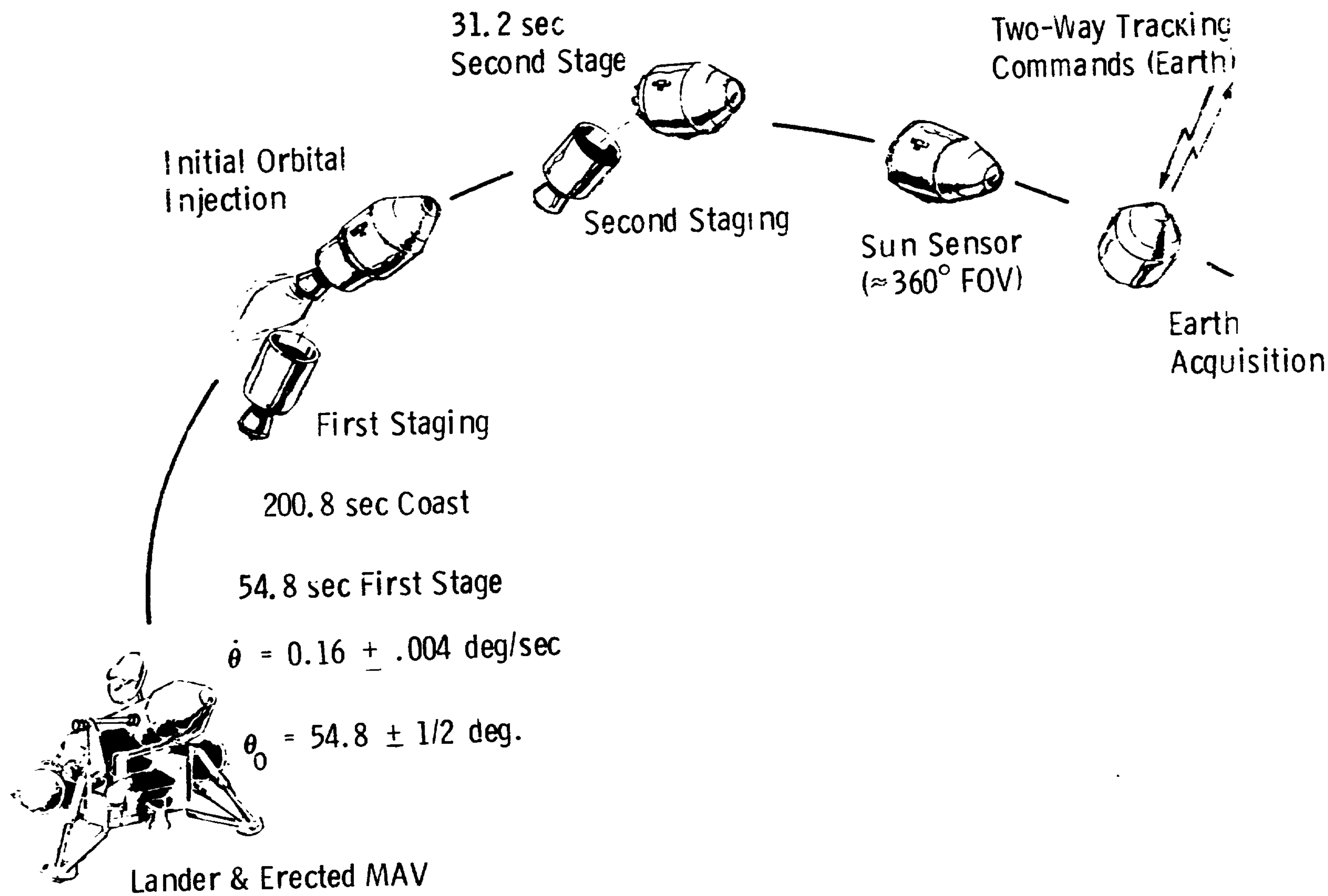


Figure VI-1 MAV Launch, Ascent, and Earth Acquisition

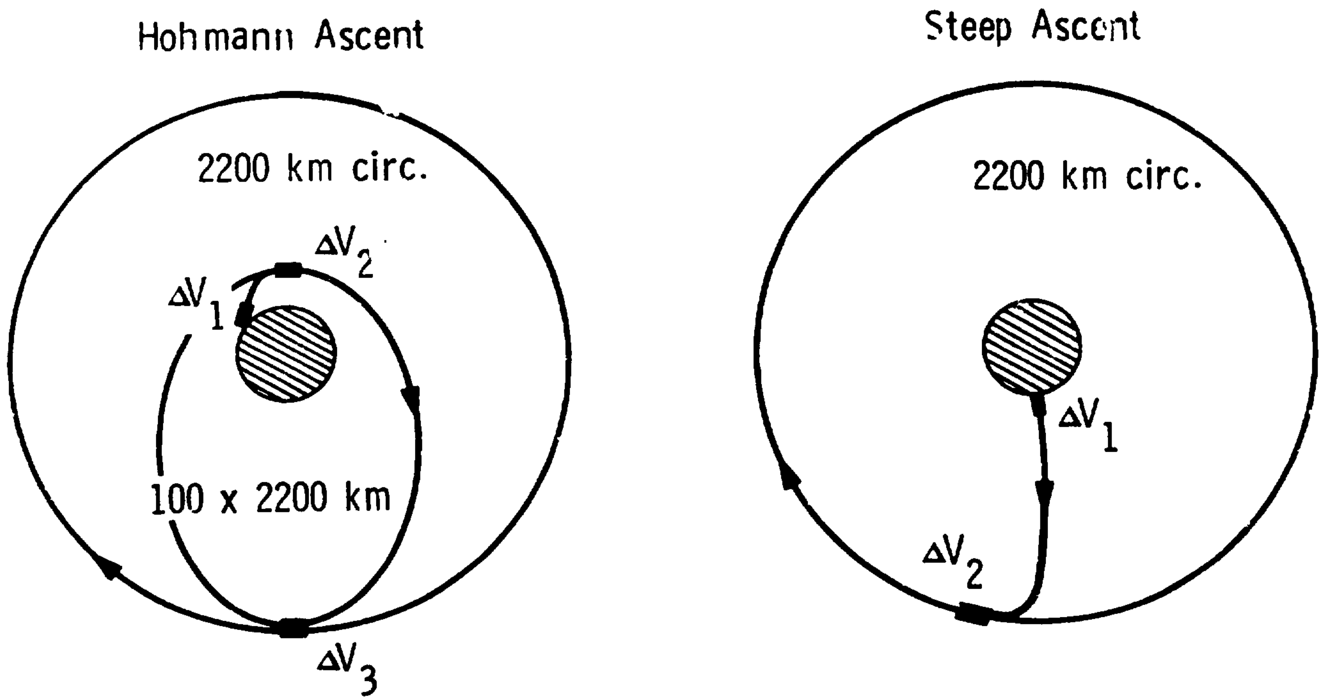
The ascent of the MAV to the 100 x 2200 km orbit was simulated by a Monte Carlo program that sampled realistic errors and uncertainties in all the MAV pointing, propulsion and timing functions. The results indicated that a stable, trackable orbit would be achieved with maximum dispersions in periapsis altitude of 12 km.

After doppler tracking of the MAV in the initial orbit the vehicle is commanded to circularize at apoapsis to the 2200 km rendezvous orbit.

The choice of the three stage Hohmann ascent to the rendezvous orbit was made after a trade summarized in Figure VI-2. Two and three stage configurations were compared in the Hohmann and steep ascent modes. A fixed 290 kg launch weight was assumed and relative performance measured by the amount of non-propulsive usable payload inserted into the rendezvous orbit. As can be seen in the figure the three stage Hohmann ascent was clearly the best.

The sequences and simulated performance of the orbiter and MAV in the initial rendezvous phase have been discussed in Chapter V of this volume. Figure VI-3 shows the position of the two vehicles at the conclusion of that phase when the orbiter is 50 km higher and 3.4° ahead of the MAV. At this time the maximum slant range between the two vehicles can be as much as 460 km (including predicted dispersions). By means of Earth calculated commands the vehicles are pointed at each other and the orbiter rendezvous radar locks on to the MAV transponder.

The first maneuver of the terminal rendezvous phase occurs when the MAV has moved up to a slant range of 100 km at which point the orbiter executes a closing ΔV maneuver down the line of sight toward the MAV that is calculated to produce an approximate rendezvous. As the orbiter closes on the MAV it executes a number of retrothrusting burns that control the closing rate and the rotation of the line of sight. This control is provided by range rate vs range relationships built into the orbiter computer. A typical set of these programmed control curves is shown in Figure VI-4 which also shows the results of a simulated rendezvous sequence. The control curves are converging pairs that indicate the conditions for "retrothrust on" (upper curve) and "retrothrust off" (lower curve). The curves are switched to a higher sensitivity pair when the range decreases to, in this case, about 4.5 km. The figure shows the final portion of a simu-



Comparison for 290 kg MAV to 2200 km Circular:

<u>3 Stages, Sol-Sol-Liq</u> Final Non-prop. P/L = 21 kg <u>2 Stages, Sol-Liq</u> Final Non-prop. P/L = 8 kg	<u>2 Stages, Sol-Liq</u> Final Non-prop. P/L = 9 kg

Figure VI-2 Mars Ascent Vehicle - Staging Considerations

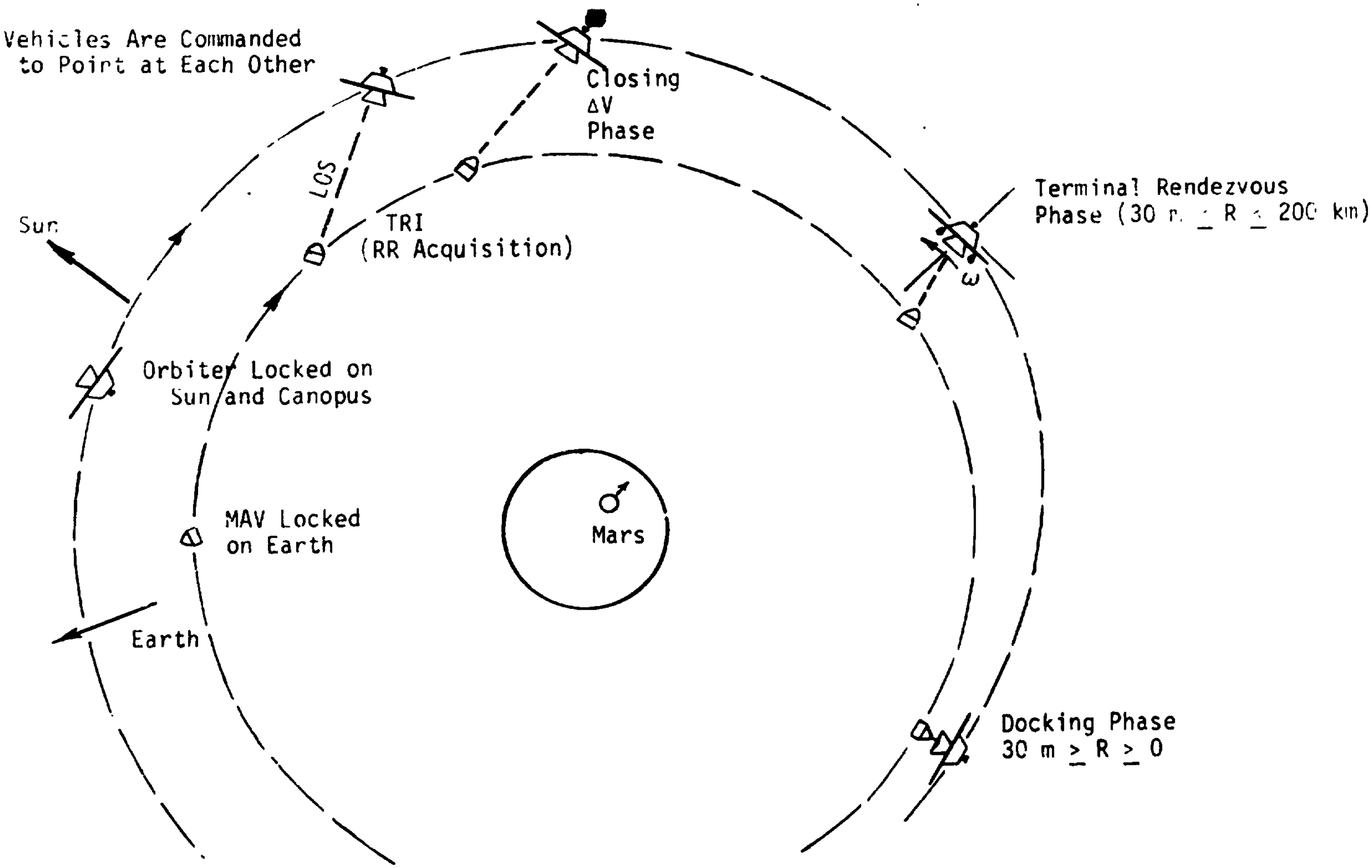


Figure VI-3 Terminal Rendezvous and Docking Phases

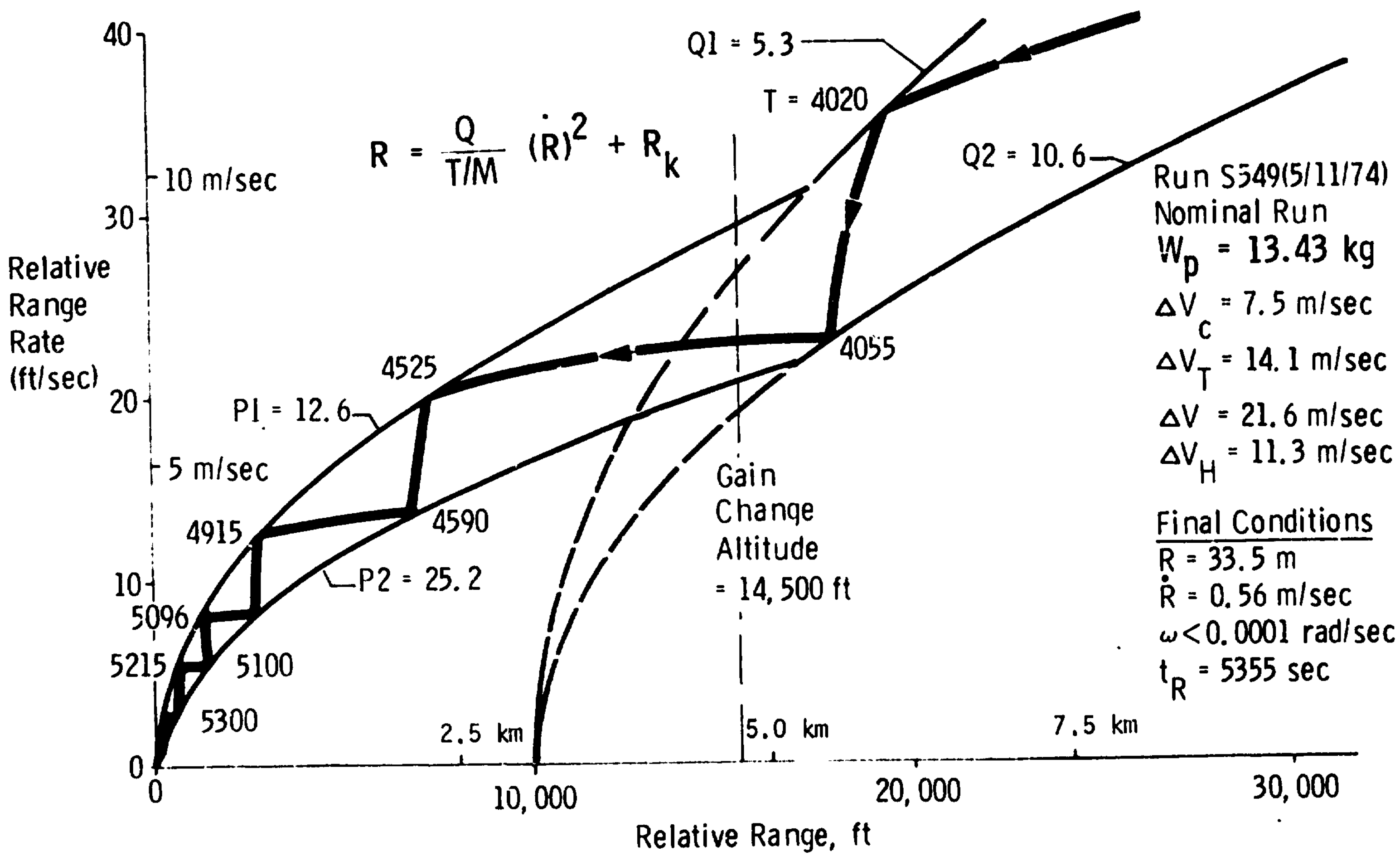


Figure VI-4 Axial Thrust Control Curves (Short Range) lation that starts with the range rate vs range relationship on the upper right corner and proceeds along the dark arrows. When the conditions corresponding to the upper control curve are hit, the orbiter propulsion system is commanded to retrothrust until the lower curve is hit and the thrusters shut off. The situation proceeds between the curves until the range and range rate are simultaneously reduced to zero resulting in a rendezvous. The rotation of the line of sight between the vehicles is also sensed by the orbiter inertial reference system and an appropriate vector offset given to the retrothrust direction to reduce the LOS rate to zero.

As can be seen in the results of this simulation which assumed a nominal separation between orbiter and MAV (50 km altitude and 340 km downrange), that closure took 5355 seconds and consumed 13.43 kg of orbiter propellant. For comparison purposes, an ideal Hohmann transfer from this separation distance would have consumed approximately 7 kg thus indicating the inefficiencies of this type of automatic rendezvous algorithm.

The docking phase begins when the orbiter has approached to a range of approximately 30 meters from the MAV and the range rate has been reduced to essentially zero. At this point the orbiter goes into full three axis control and approaches the MAV at a rate of 0.3 $\pm$ 0.1 mps as shown in Figure VI-5. The sample canister has been extended from the nose fairing of the MAV so as to mate with the docking cone of the orbiter.

Figure VI-6 indicates the details of the docking and sample transfer concept developed for this baseline.

The pointing accuracies of the orbiter rendezvous radar and the MAV transponder will allow the two vehicles to hold line of sight pointing to within $\pm$ 0.5^{o} of vehicle axes. This accuracy should keep the offset between the sample canister and the canister receptor cavity in the Earth return vehicle very small, certainly within the 1.2 meter diameter of the docking cone.

After the canister slides by the spring-loaded retainer clips in the canister receptor and activates the sensor in the receptor bottom, the MAV is commanded by the orbiter to separate the canister and back away.

Several provisions have been designed into the sample transfer concept used in the baseline to minimize the possibility that Mars biota, that might

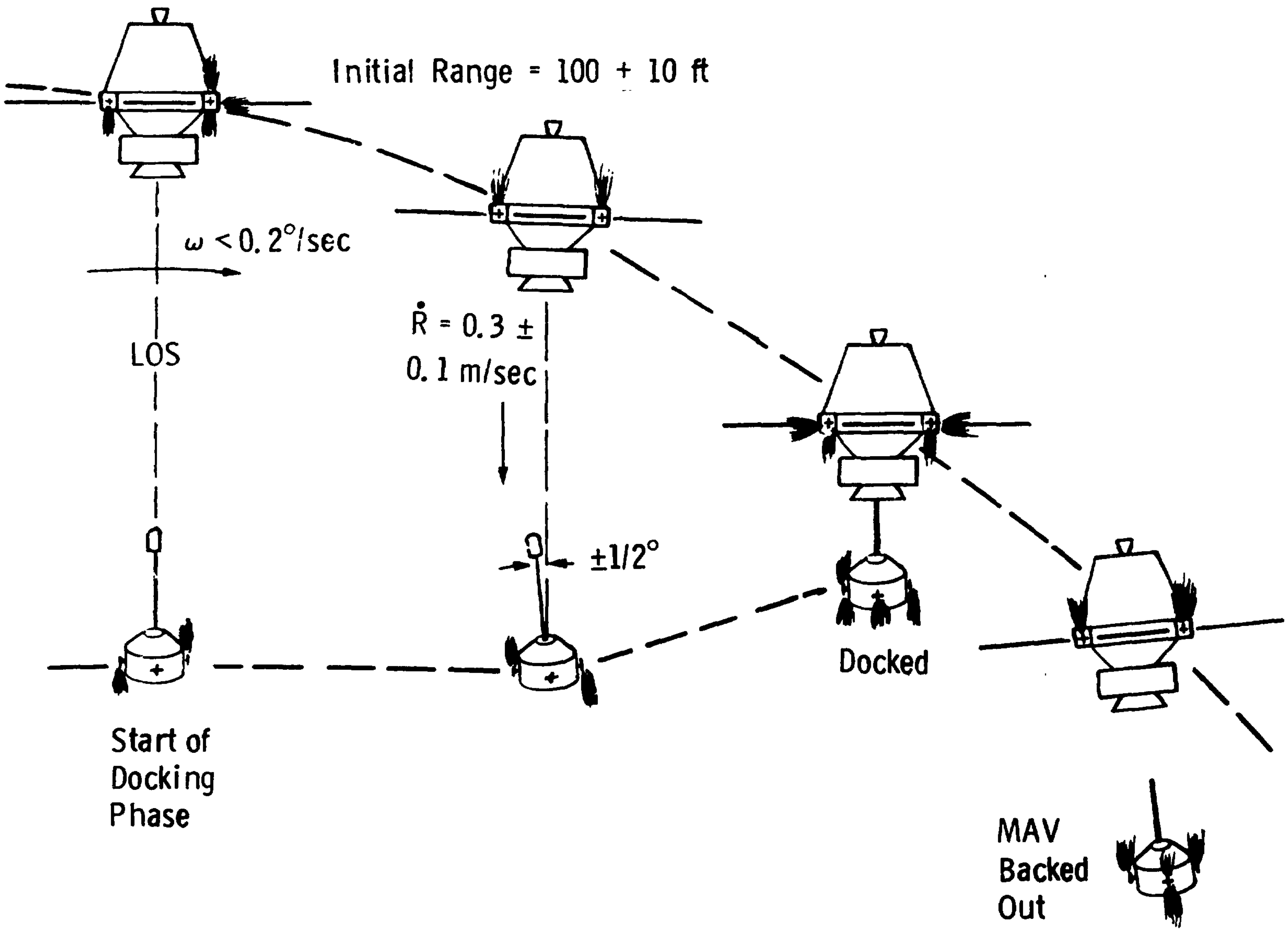
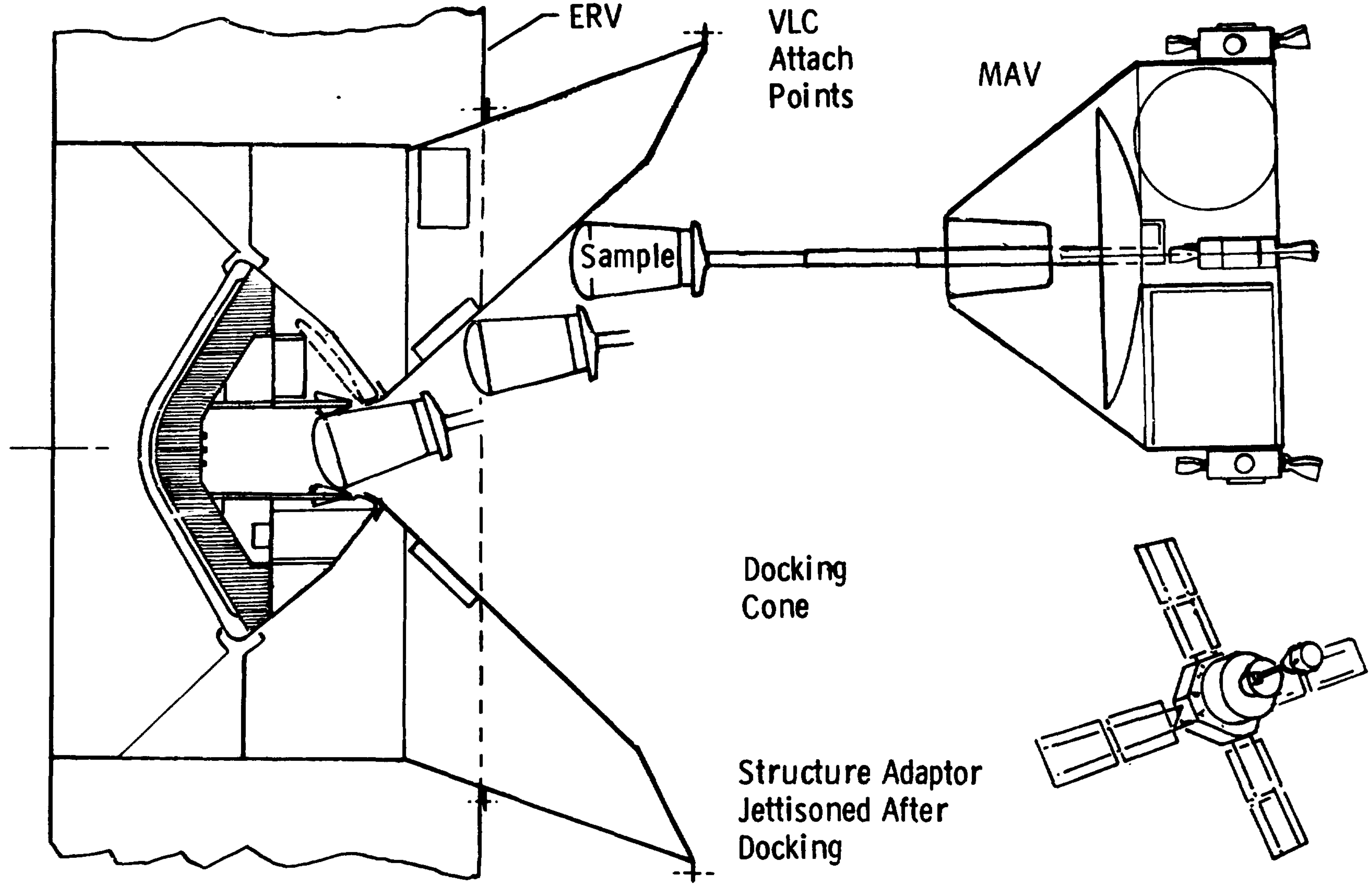


Figure VI-5 Docking Phase



: ure VI-6 Rendezvous and Docking Implementation be contaminating the MAV, will be transferred to the Earth return vehicle. These provisions are summarized in Figure VI-7.

While the MAV is on the surface, the only parts of the sample canister that are exposed to Mars contaminants are the canister nose cap and the inner slide. The nose cap can be designed so that it will be heated to approximately 650°C peak, and remain above 500°C for approximately 15 seconds, by the passage through the Martian atmosphere during ascent. At the time of docking the canister will be extended from the possibly contaminated MAV. The docking cone on the orbiter will protect the ERV from biota still on the canister cap or dislodged from the MAV in a trajectory heading toward the ERV.

After the canister has been captured and sealed inside the ERV, the MAV and the docking cone are jettisoned.

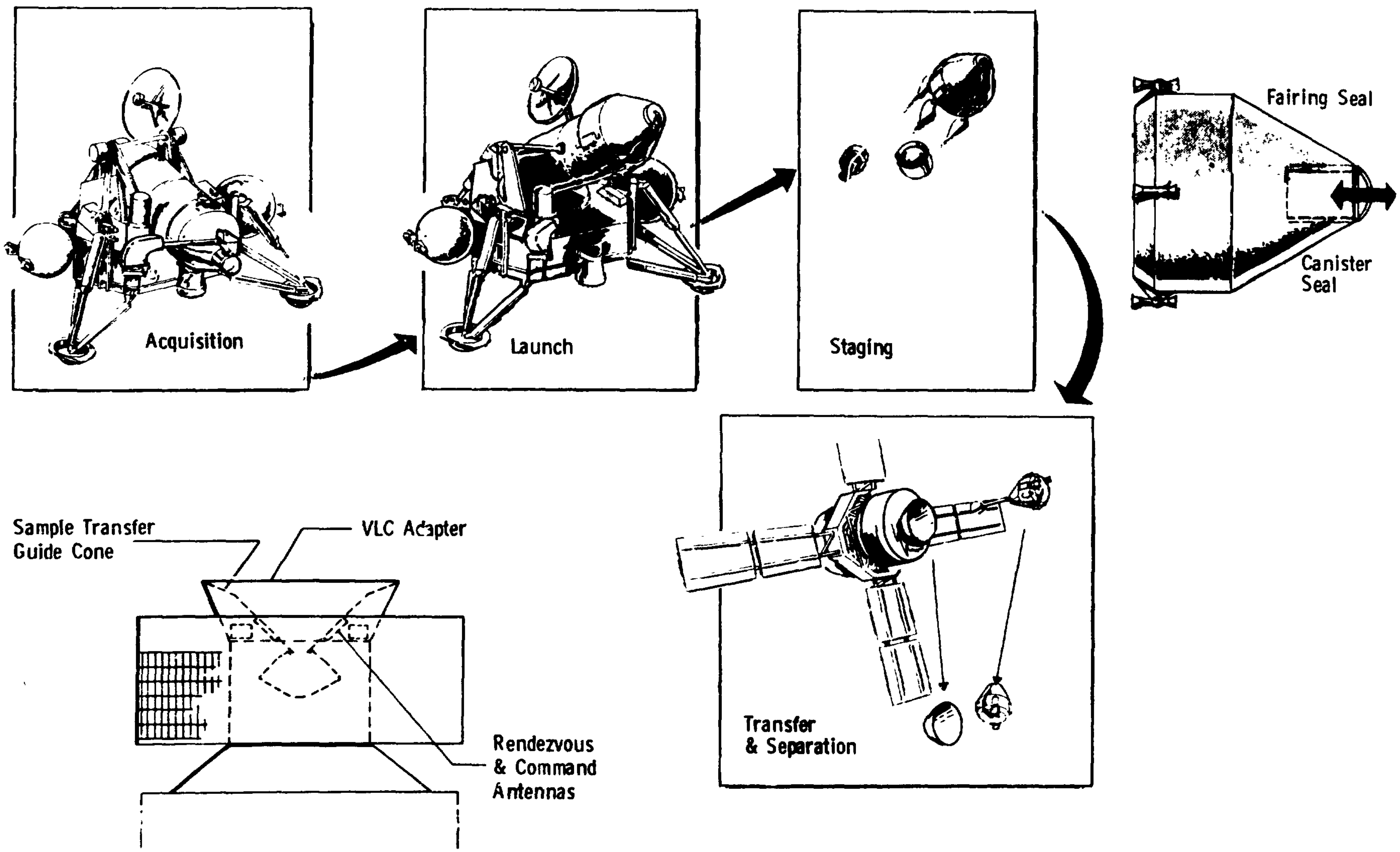


Figure VI-7 Sample Transfer and Contamination Control

VII EARTH RETURN OPERATIONS

The method employed to recover the sample, once it has been returned to Earth, will depend to a great extent upon the quarantine regulations adopted to prevent back contamination of the Earth's biosphere with Mars biota. Two basic recovery options are available: 1) direct entry into the Earth's atmosphere with air snatch or surface recovery; or 2) capture in Earth orbit with subsequent delivery to a shuttle-launched orbiting laboratory.

Direct entry recovery was assumed for the baseline developed in this study.

The Earth return vehicle will be targeted by Earth-based commands to an entry corridor that can vary from -6° (skipout) to -15°. At approximately 6 hours prior to entry the Earth Entry Capsule is separated from the ERV. It has a 5 rpm spin rate as imparted by the ERV and its attitude at release results in a zero angle of attack at entry. After separation, the ERV is deflected to a flyby trajectory. The probability that the ERV will have Mars contaminants on board is very low, making this deflection maneuver a reasonably safe one from a back-contamination probability point of view.

Figure VII-1 describes the Earth entry and recovery sequence. One hundred seconds after entry, at an altitude of 15,200 m (50,000 ft) and at Mach 0.3, the drogue chute opens. Twenty minutes later the capsule reaches 3050 m (10,000 ft) on the parachute and is sinking at the rate of 7.6 mps (25 fps). At this point aerial recovery can occur which will impose a load on the capsule of approximately 25 gs.

In the event of parachute failure, the capsule will impact the surface at about 30 mps and will experience approximately 1250 gs. Impact velocity if the chute deploys but aerial pickup does not occur will be about 6 mps.

Figure VII-2 indicates the landing site accessibility at Earth for the direct entry capsule in the 1981 Earth launch opportunity. Because of the 10° entry corridor and the $+35^{\circ}$ declination of the incoming asymptote, landing sites will be available from approximately 40°S to essentially the north

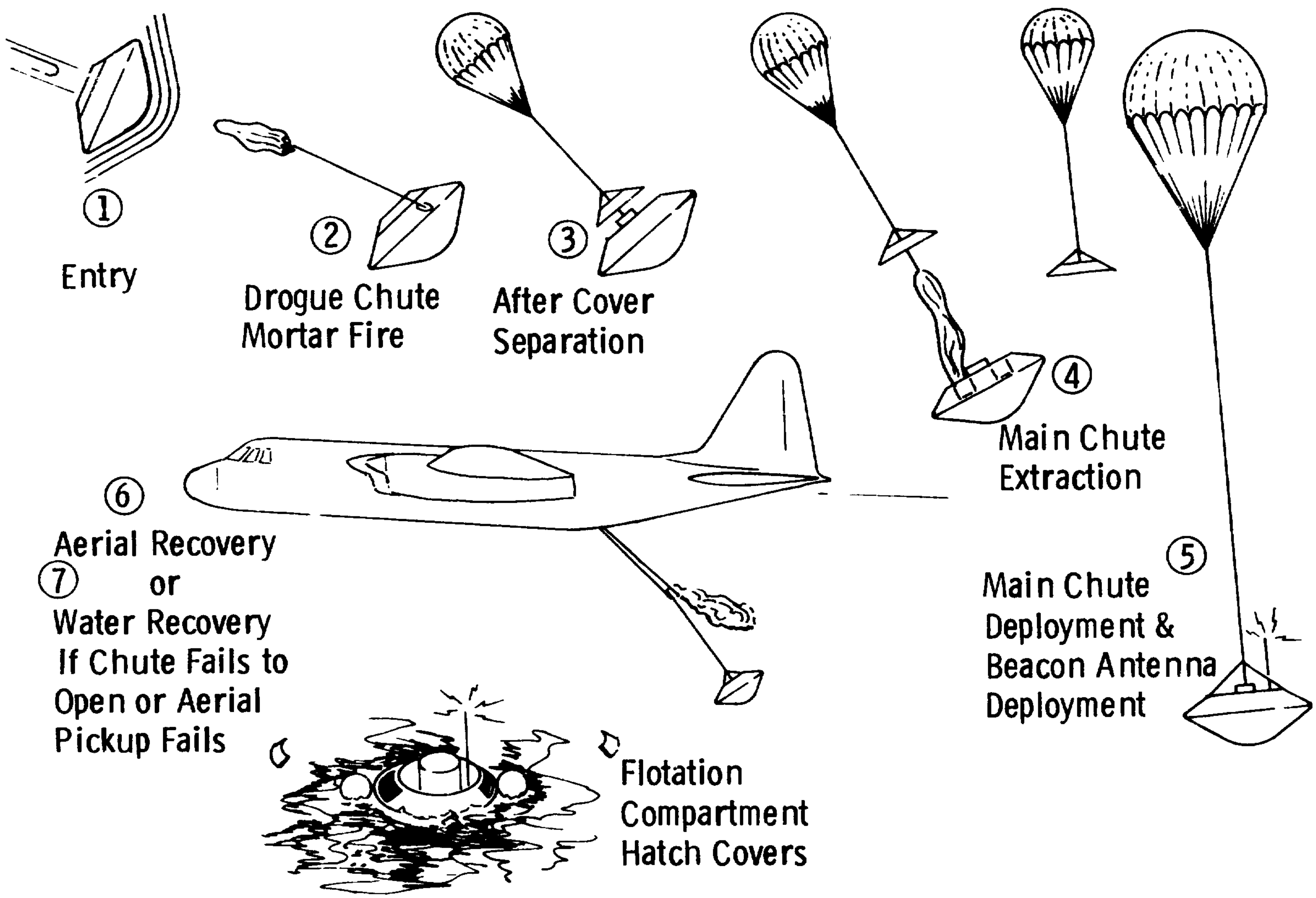


Figure VII-1 Earth Entry Capsule Recovery Sequence

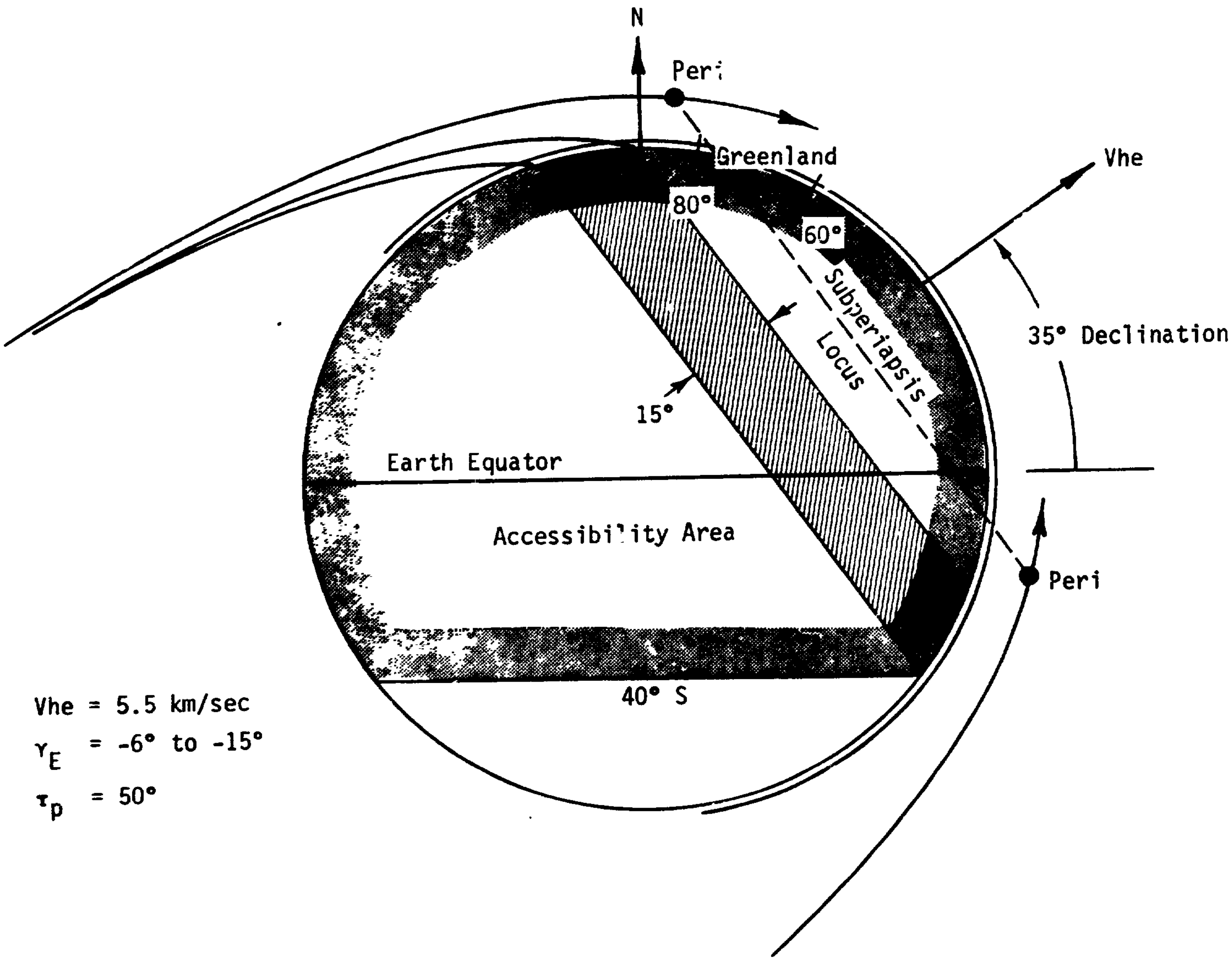


Figure VII-2 Earth Return Landing Accessibility pole. For the 1983-84 mission opportunity the equivalent landing site accessibility range extends from $50^{o}S$ to $75^{o}N$. This will provide a wide variety of land or water landing options.

The Earth Entry Capsule included in the baseline is shown in Figure VII-3. It weighs 28 kg (61 lbs) and features a 60^{o} half angle blunted cone. This shape was chosen to combine the advantages of low heat shield weight (typical of the blunt Apollo shape) and passive stability (characteristic of the narrower cone).

The capsule is designed to enter from a Mars trajectory either posigrade or retrograde and at any latitude. Structural margins will permit surface impact in the event of parachute failure without rupture of the sample container and without destruction of the tracking beacon. The beacon is a modified version of a standard Air Force recovery beacon utilizing dual antennas.

The critical design objective in the development of an acceptable direct entry capsule is to guarantee an extremely low probability of a failure mode that would result in contamination of the atmosphere or surface. Failure probabilities for structural systems are difficult to predict. Therefore, success probabilities are best enhanced by adding design margins and then exhaustively testing real hardware specimens to realistic loading conditions.

Table VII-1 outlines an approach to increasing the probability of successful sample recovery through a combination of design margins and test program additions over the baseline capsule system. Enhancements to the probability of success will increase system weights and cost ratios as shown, with the baseline system weight starting at 28 kg.

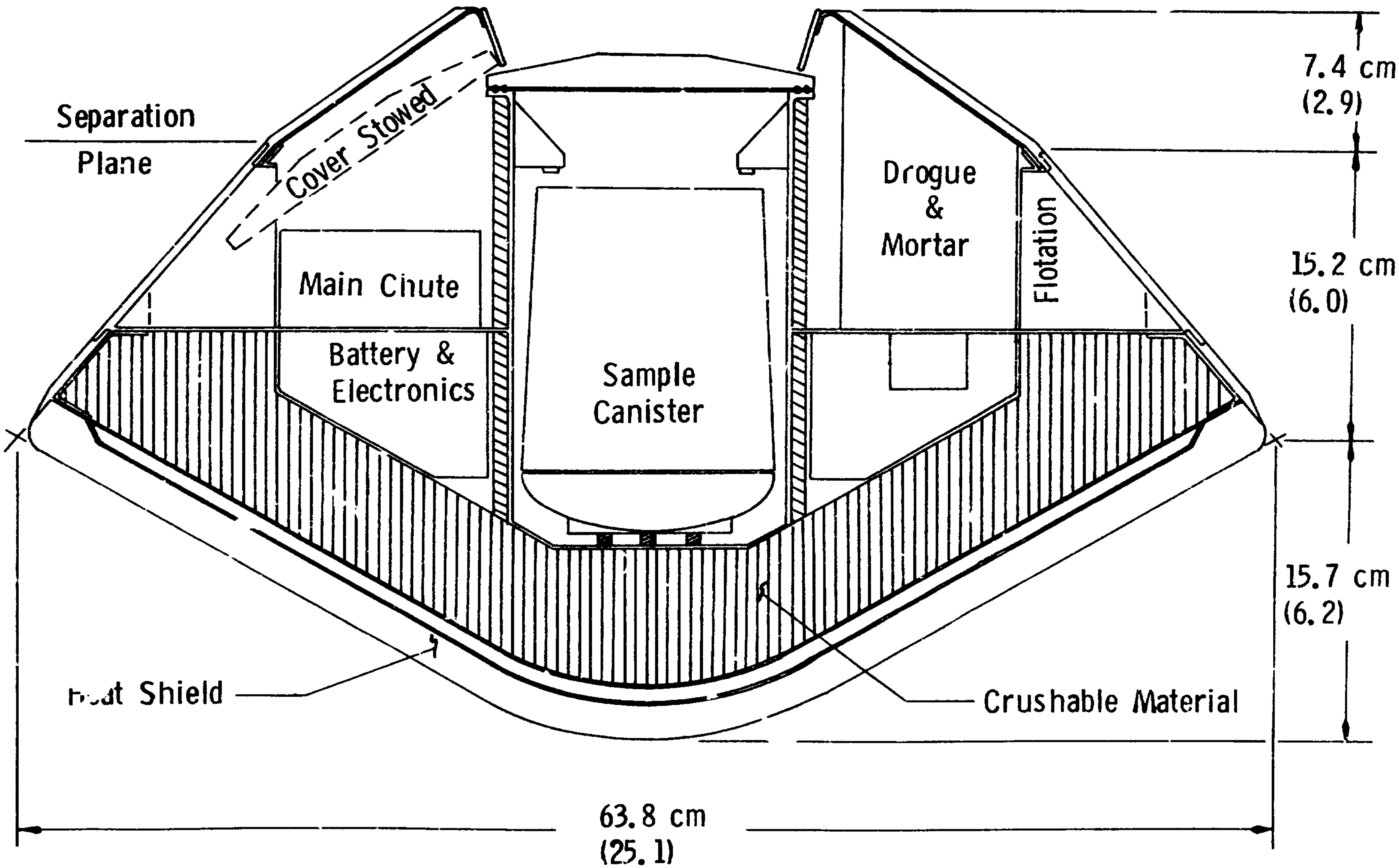


Figure VII-3 Earth Entry Capsule (1 kg Sample)

Table VII-1 Cost Trends as a Function of Probability of Successful Sample Recovery

	I Baseline System	II Enhanced Success Probability System ($\Delta Weight_I$ 100%)	III Ultimate System ($\Delta Weight_{II}$ 50%)
<u>Design Aspects of Earth Entry Capsule</u>			
System Characteristics	Single Parachute with Most Compact Configuration. Passively Stable Vehicle Normal Factors on Structure and Heat Shield 30 Day Beacon Life	Change Chute to Conventional Packaging (Requires Larger Capsule) Add Active ACS & Crushable Material on Afterbody Double Factors on Structure and Heat Shield	Two Independent Chute Systems Two Independent Beacon Systems Extend Beacon Life to 150 Days
<u>Development and Qualification Aspects</u>			
Heat Shield	Plasma Arc Coupon Tests	Large Scale Component Plasma Arc Tests	Full Scale Earth Entry and Descent Flight Test - All Systems Functioning
Parachute and Capsule Aerostability	Wind Tunnel - Aircraft Drop Tests	Same as I	
Structure	Static/Dynamic Lab Tests	Aircraft Drop Tests	
Beacon/Flotation	Functional/Environmental Lab	Same as I	
Cost Factor - Capsule System Only	1.0	1.7	3.0

VIII CONCLUSIONS

The general conclusions derivable from the results of this study can be summarized as follows:

1. Mars ascent, rendezvous, docking and sample transfer are technically feasible within present state of the art, and can in fact be performed with spacecraft derived, in most cases, from currently approved planetary programs.

2. The feasibility of automatic rendezvous and docking makes the Mars orbital rendezvous (MOR) mode the preferred approach for accomplishing the MSSR mission.

3. Using these techniques, based on existing technology and spacecraft, the MSSR mission becomes a logical next step in Mars exploration, after the Viking Landers. It represents a performance challenge that is no greater than those already taken in progressing from Ranger to Surveyor, from Gemini to Apollo, and from Mariner 9 to Viking. From a new technology point of view, the advancement required is a good deal less than that successfully demonstrated in many other space programs.

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