JPL SP 43-10, VOL. 3

Not in

Space Science Board Summer Study 1974

Planetary Mission Summary: Mars Surface Sample Return

Jet Propulsion Laboratory
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Foreword

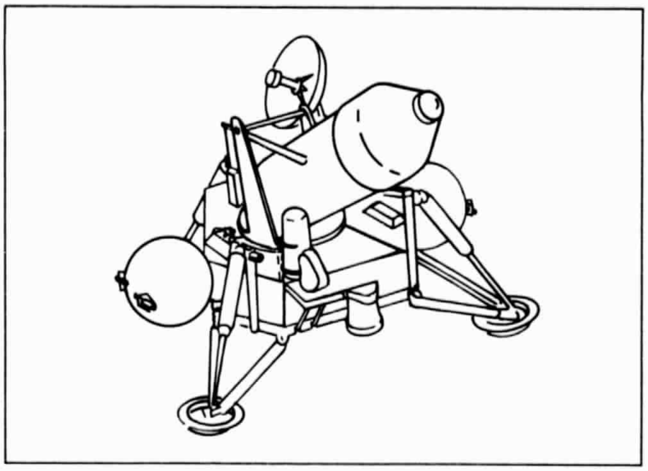
This volume presents one of a collection of planetary mission definitions which summarize what is now known about several future missions of current interest in NASA planning. Since the missions are at various stages in the planning process, the firmness and validity of the information vary. The level of detail presented, however, is uniformly concise and reflects our present best estimate of the likely characteristics of each mission. Most of the information comes from JPL technical studies sponsored by NASA.

For this mission, the choice of baseline reflects our initial judgment as to what level of performance gives a viable combination of scientific potential, development schedule, and cost. Variations from the baseline, such as launching in a later year or using a smaller or larger spacecraft, are included where they have been studied. Our objective has been to compile in brief form the main technical conclusions of recent mission studies in order that these results may interact with the broader questions of scope, pace, and priorities in the planetary exploration program as a whole.

W. H. Pickering
Director, Jet Propulsion Laboratory

Mars Surface Sample Return

Launch Date:	January 1984
Mars Arrival:	October 1984
Mars Departure:	December 1985
Earth Arrival:	October 1986
Injected Mass:	4928 kg
Instrument Mass:	35 kg
Returned Sample Mass:	1 kg
Launch Vehicle:	Shuttle/IUS, two launches



Objectives:

To return to Earth selected samples of the Martian surface for in-depth investigations and analyses.

Typical Science Investigations:

Under the baseline mission, the science is confined to acquisition, processing, handling, and return to Earth of sample. Lander facsimile camera is included to document sample. Sample pressure and temperature are monitored during return. Additional science investigations on orbiter and/or lander are under study.

Mission Description:

The mission spacecraft system has five major elements: a lander-delivery-spacecraft/orbiter, a lander, a Mars-ascent system, and an Earth-return vehicle with an Earth-entry capsule. The orbiter and Earth-return vehicle are placed in Mars orbit. The lander acquires a sample and stows it in a canister in the ascent system about 12 days after landing; the ascent system is launched, and, after docking with the Earth-return vehicle, the sample is transferred to the entry capsule. After more than 400 days in Mars orbit, the flight of the Earth-return vehicle is initiated. Near Earth, the capsule separates; after direct entry, the sample canister is recovered. Mariner, Viking, and Pioneer designs are utilized extensively to implement this 1000-day mission.

Status:

Conceptual mission feasibility established; automated rendezvous and docking at Mars can be achieved. Pre-Phase A mission design studies underway. NASA Mars Sample Return Workshop held June 1974. Sampling strategies, back contamination control, Earth recovery and quarantine procedures, and postflight sample analyses requirements have not been considered in JPL studies to date.

Estimated Funding:

(1) Launch vehicle and DSN-support funding excluded.

(2) $25 million (FY75 dollars) included for additional orbiter and/or lander science.

(3) $100 million (FY75 dollars) included for postflight sample analyses and support facility costs, per NASA/SL guidelines.

(4) Mars sterilization costs included.

(5) Earth recovery operations costs excluded.

(6) Spacecraft back contamination control costs excluded.

(7) Inflated dollars equal 5% annual inflation.

Fiscal year	80	81	82	83	84	85	86	87	88	Total
FY75 dollars (millions)	35.0	105.0	250.0	265.0	160.0	55.0	50.0	30.0	25.0	975.0
Inflated dollars (millions)	44.7	140.7	351.8	391.5	248.0	89.5	85.5	53.0	46.2	1450.9

Mars Surface Sample Return

I. Science

A. Rationale

The scientific value of a Mars surface sample return (MSSR) is unquestioned. A wide variety of measurements that would be extremely difficult or impossible to make remotely on Mars can be made in terrestrial laboratories. Multiple techniques would be available for chemical and biochemical analysis, detection of life, paleontology, isotope analysis, age dating, mineralogy and petrology, and measurements of physical properties. Results from returned lunar material provide an obvious example. What can be done with a small sample is illustrated by the work done by U. S. scientists on the 5 g of Luna 16 and Luna 20 material supplied by the USSR (Ref. 1). More effort on biology, biochemistry, and atmospheric analysis would be expected for Martian samples.

The following sections discuss what is required to achieve a sample return mission, which includes no additional science, other than that necessary to acquire and handle the surface sample. Obviously, additional desirable scientific observations can be made by increasing the size of the sample, acquiring samples from different areas, or adding remote-sensing and *in situ* instruments; these, however, would add to the total weight and cost of the mission.

B. Objective and Derived Requirements

The objective of the mission is to bring back an unchanged sample of Martian surface material large enough for minimum biological, biochemical, geological, and geochemical examination.

The minimum quantity for these investigations is considered to be 30 g. At least 20 g additional should be provided for follow-up and new investigations, giving a total minimum sample size for the mission of 50 g (Ref. 2). In the mission analysis tradeoffs, the effect of sample size on total mission weight and cost was assessed and it was determined that little impact was measured for sample increases up to 1 kg. Thus, the mission described here has a nominal sample size of 1 kg.

At least 50% of the total should be as fines, ≤ 2 mm; these are needed for both biological and geological work. The remainder should preferably consist of fines and small rock chips (0.5–5 g each).

The sample should be primarily from depths of greater than a few mm but less than 5 cm. Materials of biological interest are most likely to be close to the surface, but a slight cover is desirable to reduce effects of spacecraft exhaust on the sample. The sample should preferably be from a position at least 2.5 m from the exhaust centerline of each descent engine if Viking engines and descent profile are used. The sample site should be near minimum gravimetric elevation (maximum surface atmospheric pressure). It should be possible to select sites up to 45 deg and preferably up to 70 deg latitude.

The sample should be sealed in the ambient atmosphere. It should be taken and sealed during the coldest period of the night to insure maximum content of adsorbed and condensed atmospheric gases at the solid surfaces. Several days should elapse between landing and sampling, to permit dissipation of spacecraft exhaust gases. The sample should be obtained by a mechanical technique, not by aspirating, and should not be transferred by aspirating.

Lander propellants should not include carbon compounds; hydrazine, if used, should be purified to remove carbon. As a design goal, there should be no leakage or venting of propellants or stored gases on Mars until after the sample is sealed. Continued leakage is of great concern; it might be better to flush out lines quickly than to allow slow leakage. One suggestion is to label nitrogen compounds in the propellants with a few percent of N^{15} in order that it could be ascertained whether nitrogen in the sample came from the propellant. There should be no outgassing of volatiles from spacecraft electrical or other components on Mars.

The sample seal should be extremely reliable. The maximum leak rate should be equivalent to 10^{-10} cm^3/sec STP of He under a pressure differential of 1 bar.

The sample temperature on Mars should be at or below ambient. During return flight, it should be below $-30°C$. During Earth entry and recovery, the temperature should preferably be kept below $-30°C$; if this is not practical, it is highly desirable to keep it below $0°C$, and required that it be below $20°C$. These requirements are based on the observation that terrestrial antarctic microorganisms die if stored at $-5°C$ for several months but survive at $-30°C$ (Ref. 3). (Additional work on this problem is needed.)

The sample should not be exposed to greater than 10^2 rem of radiation or to magnetic fields exceeding the Earth's. Sample particles should be kept from rubbing against each other and against the container.

Enhancement of the baseline objective includes sampling variations as well as the increased sample size already mentioned. First, multiple samples (totalling 1 kg) could be selected from different spots within reach of the sampler. Each sample would be required to be sealed separately. In addition, a separate sample of the Martian atmosphere, compressed and sealed, would be desirable.

C. Typical Sample Support Instrumentation

As a baseline, the following measurements should be made to document the samples.

During return:

(1) Sample temperature.

(2) Integrated radiation flux received by the sample in transit ("film badge").

(3) Pressure monitoring of sample container for leaks.

On the surface of Mars:

(1) Pictures to document the sample and to aid in sample selection.

(2) Temperature of the soil, at time and place of sampling.

(3) Temperature of equipment contacting sample during sample handling.

(4) Humidity and its variation throughout the day.

(5) Wind velocity.

(6) The local slope.

(7) Mechanical loads on the sampler.

(8) Position of the sampler in three dimensions.

(9) Forces during touchdown (leg loads) and perhaps landing accelerations (to provide information on mechanical properties and therefore on other properties of undisturbed surface).

(10) Atmospheric pressure.

For the MSSR mission studies performed to date, it has been assumed that the sampler system would be Viking-derived. The studies, therefore, have not focussed on the characteristics of the sample acquisition, processing, and handling system. It is recognized that this will be an integral part of future studies. Also, science instrumentation not directly concerned with the sample has been excluded from consideration. Assumed scientific equipment is listed in Table 1.

A major issue in the design of an MSSR mission is contamination of Earth by Martian organisms (so-called back contamination). This summary recognizes back contamination as a major problem; however, scientific and engineering solutions are not discussed here. A NASA-sponsored workshop held in June 1974 (Ref. 4) considered the back contamination issue and other issues such as the scientific impact of sample sterilization, Earth-based quarantine procedures, and the prevention of spacecraft contamination. Results from workshops similar to the above are expected to formulate the scientific and engineering guidelines for addressing these issues in future mission studies.

II. Mission Description

Launch opportunities for Mars round-trip missions are generally associated with Mars–Earth oppositions and precede by 3 to 4 months the opposition dates, which occur on the average every 25.6 months. Because of the eccentricity of the Mars orbit, the heliocentric trajectory profiles and, consequently, the total energy requirements vary from one launch opportunity to the next. Of interest here are the Mars launch opportunities which occur in November 1981, December 1983–January 1984, and May 1986. The total energy requirements for these opportunities progressively increase from 1981 through 1986, which implies that a greater launch vehicle and space-

Table 1. MSSR science instrumentation

Instrument	Mass, kg
Sampler boom and head	13.0
Sample processing and loading	7.0
Sample canister	1.0
Sample sealing assembly	7.6
Facsimile camera	6.0
Temperature and pressure sensors	0.4
Total	35.0 kg

craft performance capability is required as the launch year is extended. For this paper, the 1983/1984 opportunity is considered baseline.

To return a sample from the surface of Mars requires both a long series of critical functions and a high total energy. The minimum energy mission profiles which are within the capability of current and planned launch vehicles require long Earth–Mars and Mars–Earth flight times. Additionally, they require that the spacecraft wait at Mars for about a year for the appropriate heliocentric geometry before starting back. Parking in Mars orbit is preferred to staying on the surface, because the orbital environment is more benign and predictable. This is the so-called "conjunction class" mission; the total mission lifetime exceeds 1000 days.

Two fundamental mission modes exist for the MSSR mission. One is the direct mode, in which all Earth-return systems are landed on the surface of Mars; the second is the Mars orbital rendezvous mode, in which the Earth-return systems are inserted into Mars orbit and the samples are brought to a rendezvous by a Mars descent/ascent spacecraft. The direct mode is simpler in concept while the orbital rendezvous mode can be achieved with smaller, lighter vehicles that demand less of Mars entry, landing, and ascent. Further, the rendezvous mode does not expose the Earth return systems to contamination by possible Martian biota on the surface.

The baseline MSSR mission reported here is launched in January 1984 and is implemented in the Mars orbital rendezvous mode. This mode is described in detail in Ref. 5. The mission sequence of events is illustrated in Fig. 1. Earth launch is performed by a Shuttle/interim upper stage (IUS). A total mass of 4928 kg is required at injection for the 1984 MSSR mission. The Shuttle/IUS must be capable of injecting about 500 kg above the Titan III-E/Centaur capability.

During cruise to Mars, the total MSSR spacecraft consists of an orbiter, lander, Mars ascent vehicle (MAV), Earth-return vehicle, and an Earth-entry capsule. After lander separation at encounter minus 4 hours (E − 4 hr), the orbiter (with Earth-return vehicle and entry capsule) is inserted into a "loose" capture orbit with a 1000-km periapsis and a period of 105 hr. This orbit is held for 10–15 days while the Mars surface landing and sample acquisition take place, followed by MAV ascent and placement in a 2200-km-altitude, circular rendezvous orbit.

A modified Viking lander carries the MAV to the Martian surface on a direct entry trajectory. This vehicle is capable of automatically ascending to a 100 × 2200 km orbit and thereafter being commanded to circularize at 2200 km into the rendezvous orbit. The design approach used was to keep the MAV as simple as possible and keep its maneuvers under Earth or orbiter control whenever feasible.

Flexibility in the choice of landing sites is achieved by timing the entry appropriately and performing orbit plane changes. Typically, ±65-deg latitude and all longitudes are possible.

Critical mission performance functions are the orbiter navigation and maneuver execution and the MAV ascent guidance and control. The MAV uses open-loop guidance with a constant pitchover rate during the ascent phase. In orbit, the MAV and orbiter are both tracked. The orbiter maneuvers to a 2250-km-altitude circular orbit in a phase appropriate for starting rendezvous.

The orbiter's rendezvous radar acquires the MAV, and an active radar loop is established between the two vehicles. Proportional navigation is used to drive the orbiter down to dock with the MAV. Then the surface sample canister is transferred from the MAV to the entry capsule in the Earth-return vehicle. The total time from landing to sample transfer is about 30 days.

After remaining in Mars orbit for more than 400 days, the Earth return vehicle separates from the orbiter and is injected on an Earth trajectory. Midcourse maneuvers are employed to target the Earth-return vehicle for entry capsule separation. About six hours before entry, the capsule is separated from the spinning Earth-return vehicle at an attitude resulting in a zero angle of attack at entry (nominal angle of −10-deg). One hundred seconds after entry, at 15,000 m altitude and Mach 0.3, the parachutes and beacon antenna are deployed. Twenty minutes later the capsule reaches 3,000 m altitude and 7.5 m/sec, at which time the recovery aircraft engages the chute with its trailing grappling hook system and winches it into the aircraft. The air-snatch load is ~25 g.

In the event of parachute failure the capsule impacts the ground or water at slightly above 30 m/sec and about

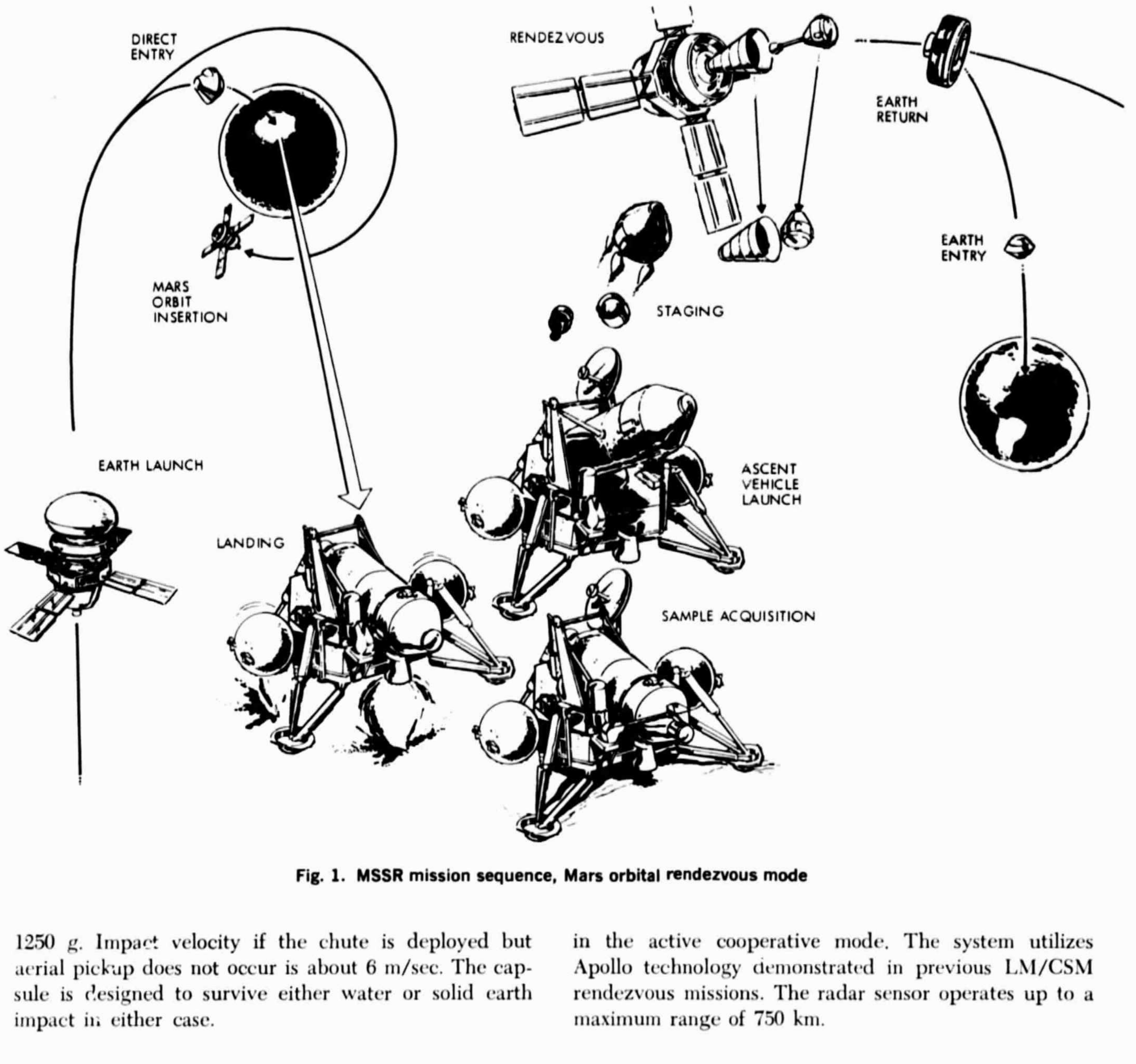


Fig. 1. MSSR mission sequence, Mars orbital rendezvous mode

1250 g. Impact velocity if the chute is deployed but aerial pickup does not occur is about 6 m/sec. The capsule is designed to survive either water or solid earth impact in either case.

III. General Spacecraft Characteristics

The MSSR Orbiter is derived from the Viking Orbiter by removing orbit science and related items, replacing the cold gas reaction control system with a combination reaction-control/maneuver system to provide braking thrust during final rendezvous and docking, and stretching the main propulsion system. Orbiter guidance and control must be modified to utilize an additional sensor, which is the rendezvous and docking radar.

The rendezvous and docking radar provides range, range rate, and angle data to the orbiter computer. An S-band, all solid-state CW system is employed, operating in the active cooperative mode. The system utilizes Apollo technology demonstrated in previous LM/CSM rendezvous missions. The radar sensor operates up to a maximum range of 750 km.

Several changes to the Viking Lander configuration are required to mount the MAV and its launcher. All lander science, except one camera and the sample acquisition system, is removed. A MAV launcher is mounted on the lander equipment plate with 360 deg of azimuth rotation and 79 deg of elevation. The lander terminal descent propulsion system is pressure-regulated and propellant storage is increased.

The Mars ascent vehicle (MAV) is a three-stage three-axis-stable launch vehicle (Fig. 2) weighing 290 kg. It is the only entirely new vehicle in the MSSR spacecraft configuration. Salient features of the MAV subsystems include:

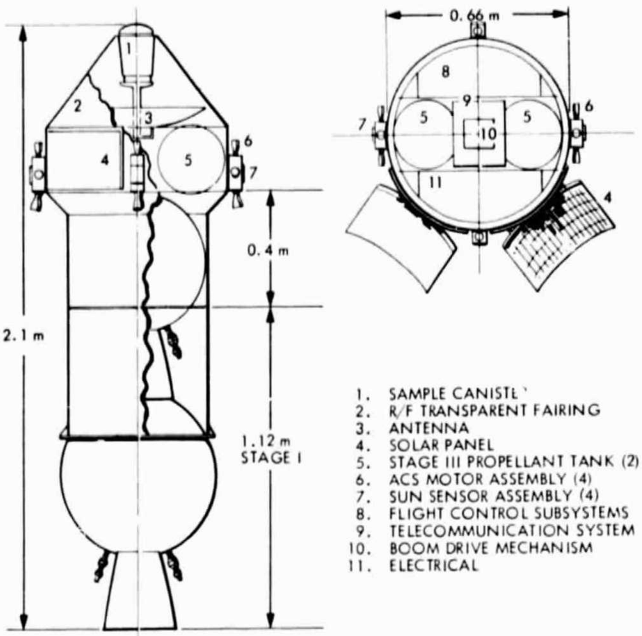


Fig 2. Mars ascent vehicle

(1) Guidance and control. Open-loop, constant pitch-over-rate with rate-gyro reference during ascent and sun-sensor/Earth-pointing reference during orbital operations.

(2) Telecommunications. S-band transponder. Earth tracking provides command, telemetry, and two-way coherent doppler links. Orbiter tracking provides pointing reference during rendezvous.

(3) Propulsion. Sterilizable solid-propellant Stage I and II. Monopropellant hydrazine Stage III for maneuvers and attitude control.

The Earth return vehicle is a spin-stabilized vehicle which carries the Earth entry capsule. Pioneer Venus, currently in the early design stage, is a candidate for return vehicle design. Preliminary evaluation indicates high commonality with Pioneer Venus.

A cutaway view of the Earth entry capsule is shown in Fig. 3. The soil sample canister is located on the vehicle centerline, for ease in sample transfer. The beacon and flotation systems surround the sample receptacle and all systems are supported by the crushable honeycomb, which allows for survival over water or land if the parachute fails. The sample canister receptacle can be either aluminum or titanium (for thermal purposes) and does not require heavy gage construction because of the presence of the crushable material. The beacon system will be adapted from the AF satellite recovery program, with additional batteries provided to yield a 30-day survival time.

A mass summary for the MSSR spacecraft is shown in Table 2.

IV. Mission Options

The mission concept described previously is somewhat mass limited. The Mars ascent vehicle, a critical mission element, has little margin due to the limited capacity of

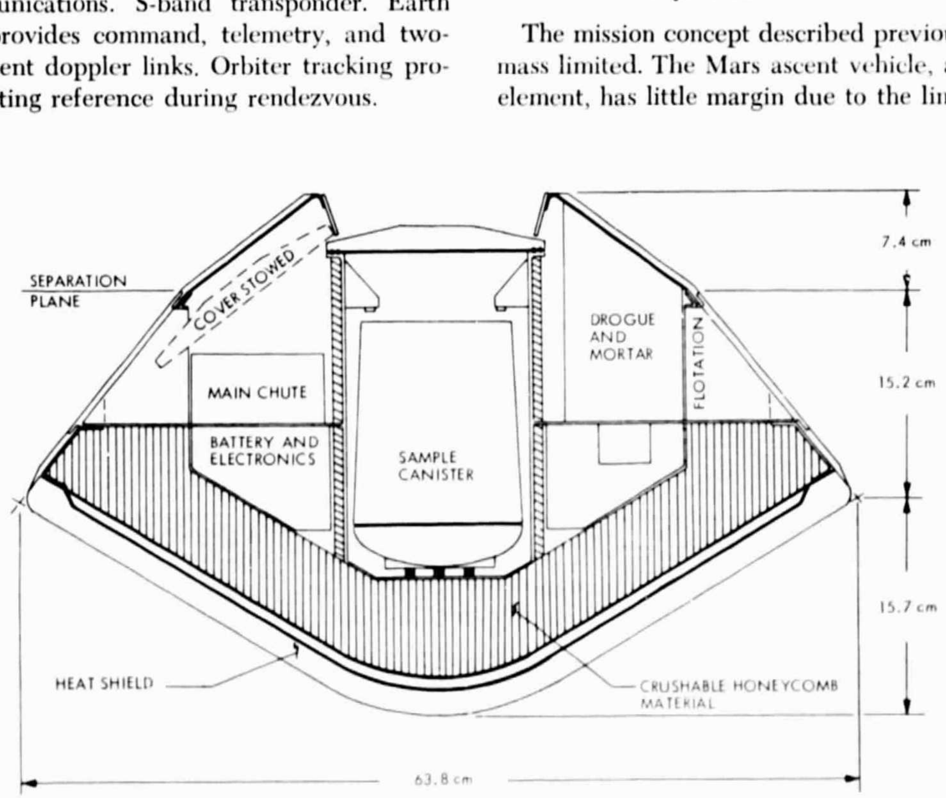


Fig. 3. Earth entry capsule (1-kg sample)

Table 2. MSSR spacecraft mass summary

Spacecraft element	Mass, kg
Orbiter	2757
Lander	1010
Ascent vehicle	290
Earth return vehicle	235
Earth entry capsule	30
Mass margin	300
Adaptors, bioshield, launch-vehicle-peculiar	300
Total injected weight	4928 kg

a relatively unchanged Viking system. To improve the margin, a dual-launch Mars-orbital-rendezvous mode has been defined (Fig. 4). The first launch consists of the orbiter and Earth-return vehicle (with the Earth entry capsule), which are inserted into orbit about Mars. The second launch consists of a cruise module to support the lander to Mars and the lander. The lander is separated and enters Mars on a direct trajectory, while the cruise module flies by. The remaining mission sequence is as discussed previously.

Several advantages accrue from this concept. First, the total injected mass per launch is considerably reduced. Second, raising the injected mass will provide the opportunity to improve the lander delivery performance: the aerodecelerator system capability (parachutes, aeroshell, retros) will be increased. Part of the margin, in turn, can be used in the mass-critical ascent vehicle. Third, additional science can be added to the orbiter and/or lander. The dual launch concept will be considered in studies during 1974/1975 and evaluated in comparison to the single launch concept.

Preliminary option studies for this mission have also considered alternate launch opportunities and the use of space-storable propulsion systems. Compared to Earth-

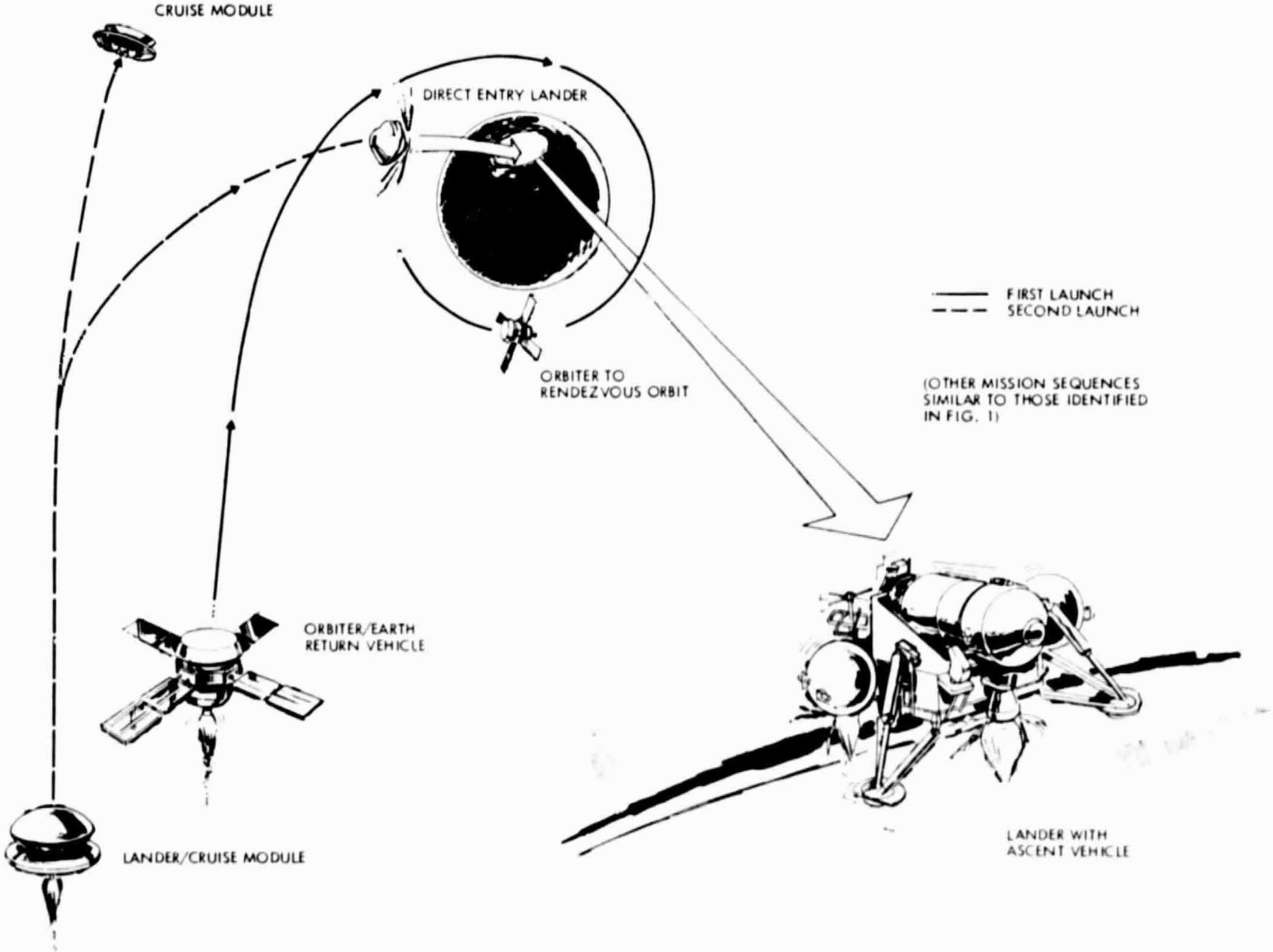


Fig. 4. Alternate mission concept, dual-launch Mars orbital rendezvous mode

storable systems (I_{sp} = 291 sec), space storables (I_{sp} = 375 sec) offer considerable mass relief at Mars orbit insertion.

Options for a 1981 launch (Earth-storable) and 1984 and 1986 launches comparing the two propulsion sets are compared in Table 3.

Another fundamental mission option that will be explored in further studies is the retrieval of the sample in Earth orbit, as contrasted with direct Earth entry recovery. Here, the capsule would be placed in orbit around Earth and the Shuttle/Tug would be used for retrieval. The Tug would be required to rendezvous and dock with the capsule and return to the Shuttle. The sample could then be evaluated in the Shuttle/Spacelab or returned to Earth-based laboratories.

The direct mode MSSR mission has been studied previously (Ref. 6). These results will be comparatively evaluated with the rendezvous mode results during 1974–1975 to quantify the advantages of each mode.

V. Program Assessment

The 1984 MSSR mission can be achieved without major advances in technology. The principal area requiring advancement in existing technology is solid propellant sterilization. Sterilizable solids are used in the Mars ascent vehicle for Stages I and II. A technology program in this area is currently underway and is expected to provide a technology readiness in sufficient time for this program. Although complex in nature, the rendezvous and docking sequence can be implemented with essentially existing systems. Other areas requiring further emphasis are the lander system performance, ascent vehicle performance, rendezvous and docking, and the impact of back contamination control procedures on the MSSR spacecraft. The above areas will be addressed in future efforts.

The cost of an MSSR mission is strongly dependent on the scientific scope of the mission. The preliminary estimates below are only valid for the baseline mission described, and changes in objectives or implementation are likely to result in considerable cost growth.

The cost study performed for this mission was based on the following assumptions:

(1) Launch vehicle and DSN-support funding excluded.

(2) $25 million ($FY75) included for additional orbiter and/or lander science.

(3) $100 million ($FY75) included for postflight sample analysis and support facility costs, per NASA/SL guidelines.

(4) Mars sterilization cost included.

(5) Spacecraft back contamination control costs excluded.

(6) Earth recovery operations costs excluded.

The program cost breakdown is as follows, in millions of dollars (FY75):

Science		$ 45 million
Spacecraft elements		550
Orbiter	167	
Lander	210	
Ascent vehicle	68	
Earth-return vehicle	60	
Earth entry capsule	45	
Rendezvous/docking simulation		30
Mission operations		65
Postflight data analysis/facility		100
MCCC		15
Project management and integration		81
Contingency		89
Total		$975 million

Table 3. Launch and propulsion options

Launch year	1981	1984		1986	
Propulsion system	Earth-storable only	Earth-storable (baseline)	Space-storable	Earth-storable	Space-storable
Spacecraft mass at Mars orbit insertion, kg	2571	3022	2426	3175	25[illegible]5
Separated lander/ascent vehicle mass, kg	1285	1300	1300	1320	1320
Margin, kg	300	300	300	300	300
Total injected mass, kg (includes 306 kg for adapters, bioshield, etc.)	4462	4928	4332	5101	444[illegible]

Funding requirements as a function of time are given in Table 4.

Table 4. Mission funding spread

FY	80	81	82	83	84	85	86	87	88	Total
FY75 dollars (millions)	35.0	105.0	250.0	265.0	160.0	55.0	50.0	30.0	25.0	975.0

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