EXPLORATION OF INTERPLANETARY SPACE BY UNMANNED VEHICLES

by

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ABSTRACT

The United States interplanetary program has been initiated with the Mariner II flight to Venus in 1962 and the Mariner IV flight to Mars this year. The spacecraft used for both missions are similar in many respects, have many similar subsystems, and were both launched by the Atlas-Agena. Scientific measurements in interplanetary space and in the vicinity of the target planets are among the objectives in each case. Later missions will be able to use larger boosters such as Saturn, and will be capable of much more advanced scientific investigations. Primary objectives of Mars exploration will include investigations for evidence of extraterrestrial life, measurements of planetology and environmental characteristics, and acquisition of data useful for manned landing. Landing sites from which can be observed the "wave of darkening" will be selected, as well as sites in desert regions and near the polar caps. A split payload mission offers many operational advantages. Such a system would include a spacecraft bus which carries the landing capsule. The capsule is separated from the bus near the planet and placed on an impact course. The bus can either fly by the planet or be placed in orbit. Both lander and bus will transmit scientific data to Earth. This concept is being studied for the Voyager program, in which the first operational Mars mission is planned for 1971.

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* This paper represents work performed by Avco Corporation under earlier Voyager contracts; the opinions and statements expressed are those of the author and are not necessarily the current NASA Planetary Program policies.

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A. INTRODUCTION

One of the most exciting possibilities of the exploration of interplanetary space is the discovery of evidence of life of nonterrestrial origin. This is one of the major objectives of the United States program. Two other important objectives are to achieve a greater understanding of the characteristics of the planets and interplanetary space itself, and also to pave the way for manned exploration of the solar system. These objectives will be discussed in greater detail later on; first, it is desirable to outline the history of the United States program and to indicate the plans for the future. The bulk of the paper will be devoted to a discussion of the scientific and technical problems, and consideration of several methods of approach which are being studied.

The United States interplanetary exploration program was successfully initiated on 14 December 1962 when the Mariner II spacecraft passed within 35,000 km. of the planet Venus. The next major step is the Mariner IV spacecraft which was launched on 28 November 1964, and will fly by Mars on 14 July 1965. These two flights conclude the Mariner program. The more ambitious Voyager program is now being studied as the possible next step in interplanetary exploration.

B. HISTORY OF THE UNITED STATES INTERPLANETARY PROGRAM

1. Mariner II

The Mariner probes to Venus and Mars have been launched by the Atlas booster combined with the Agena second stage. The launch of Mariner II took place on 27 August 1962 and encounter with Venus occurred 109 days later on 14 December 1962. (Mariner I was launched a month earlier, but the launch attempt was unsuccessful.) The Mariner II spacecraft (Figure 1) was stabilized in space by means of an attitude control system which kept its solar panels facing toward the Sun. A high gain antenna was kept pointed back toward Earth to transmit the scientific data which was obtained while Mariner II was on its journey. The spacecraft contained batteries which were kept charged by electrical power obtained from the solar cells. In addition to a transmitter, Mariner contained a radio receiver to receive commands from Earth. The spacecraft contained a propulsion system in order to perform the midcourse correction. Signals to the attitude control system were furnished by optical sensors which locked on Earth and Sun; at other times these signals were furnished by gyroscopes. The Mariner was also equipped with the necessary programming devices, known as the central computer and sequencer, to control the operation of the equipment and carry out commands received from Earth-based stations. The complete spacecraft weighed 196 kg; of this, 21 kg. was for the scientific instrumentation. These included a detector for counting protons in the solar wind, a crystal microphone for registering impacts by particles of interplanetary dust, three Geiger-Muller tubes, an ion chamber, a magnetometer, a 20 inch parabolic antenna to scan the surface of Venus at microwave frequencies, and two optical sensors to measure infrared radiation from the planet.

The Mariner II was placed in an Earth parking orbit by its Atlas-Agena launch vehicle, and was subsequently injected on an interplanetary trajectory by the Agena rocket. The spacecraft was separated from the burned-out last stage, and its solar panels were deployed. The cold gas attitude control system turned the spacecraft to keep its solar cells facing towards the Sun to assure a supply of electrical power to the Mariner's batteries. By means of Earth-based tracking it was determined that the trajectory of Mariner II would miss Venus by 375,000 km., and nine days after launch a command was transmitted from Earth to cause the spacecraft to perform a midcourse correction. The Mariner turned through the proper angle and fired its hydrazine rocket engine to make a velocity change of 20 meters per sec. As a result of this correction, Mariner II's trajectory was changed so that it would pass 34,900 km. from the planet.

During the interplanetary voyage, the magnetometer determined magnetic-field strengths and fluctuations along its trajectory. The solar-wind detector obtained hitherto unknown data concerning the nature of the solar plasma. The Geiger-Muller tubes and the ionization chamber determined the density and energy spectrum of the high energy cosmic ray particles in interplanetary space.

As Mariner II approached its encounter with Venus, the radiometer antenna scanned the surface of the planet to obtain data which would be useful in determining the surface temperature of Venus. The vast amount of data transmitted by Mariner II has furnished invaluable information on the nature of the portion of interplanetary space between earth and Venus, as well as new knowledge about our sister planet. Also of key importance, the flight provided additional insight which can be used to improve the design of future space vehicles. Details on Mariner II, its design, scientific objectives, and results are contained in References 1-7.

2. Mariner IV

The next step in the United States interplanetary program was the launch of a Mariner spacecraft to Mars. Mariner IV (Figure 2) was launched on 28 November 1964, by the Atlas-Agena*. It had a total weight of 260 kg., and its objective was to investigate interplanetary space between orbits

* Mariner III was launched on November 5, 1964, and was placed on an orbit towards Mars, but communications were lost due to battery failure when the fairing did not jettison.

of Earth and Mars and to perform experiments in the vicinity of Mars. The payload was about 3 m high by 7 m (cruise position, with solar panels and pressure vanes deployed). It consists of a structure which is basically an octagon with four 2 m by 1 m solar panels extended horizontally and four solar pressure vanes extending beyond the panel tips. Power is obtained from 28,224 solar cells and a 33 pound silver-zinc battery. Six experiments will provide data on magnetic fields, cosmic and other particles in interplanetary space and near Mars (solar plasma probe, ionization chamber, trapped radiation detector, helium vector magnetometer, cosmic ray telescope, cosmic dust detector). Other equipment includes a television camera, radio transmitters, control system, and two antennas extended vertically from the octagon. Mariner IV utilizes solar vanes in order to balance the bias torques produced on the spacecraft by solar pressure. The solar vanes cause the center of solar torques to coincide with the center of spacecraft mass maintaining the spacecraft attitude with respect to the Sun (Reference 8). Mariner IV was placed in an excellent interplanetary orbit, with all systems functioning well. The initial trajectory would have caused the craft to pass 243,000 km. ahead of Mars on 17 July 1965; on 5 December 1964, a midcourse correction maneuver successfully altered course so the spacecraft would pass about 9000 km behind Mars at 2 A.M. GMT on 15 July 1965, after a flight lasting 228 days and covering 525,000,000 km. On 7 December 1964, the solar plasma probe failed and on 3 March 1965 the ionization experiment ceased to function, shortly after it passed thru a solar flare (Reference 9), but the other interplanetary sensors continued to return data. In the vicinity of Mars an attempt will be made to take about 20 TV photos of Mars and use radio waves in an occultation experiment to determine composition of the Mars atmosphere and data relating the scale height and pressure (Reference 10). A secondary objective is to provide experience and knowledge concerning the performance of the basic engineering equipment of an attitude-stabilized flyby spacecraft during a long duration flight in space farther away from the Sun than the Earth (Reference 11). The first pictures are to be recorded when the craft is 13,500 km above the surface, starting with the northern Martian desert of Amazonis, (40° north latitude) with subsequent pictures sweeping southeasterly across the equator to record Mare Sirenum, the Phaethontis desert, and finally, Aonius Sinus (in the vicinity of the planet's south pole). At this point, 10,000 km above Mars, Mariner will record its last picture as cameras sweep into the terminator. Its maximum picture resolution will be achieved near the middle of the picture sequence over Mare Sirenum, when its cameras are pointing nearly straight down. Pictures here will measure 240 km. on a side and resolve objects down to 3 km. in diameter. Each picture will contain 250,000 bits of data and will be relayed back to Earth at 8-1/3 bits per second over a distance of 240 million km (Reference 12). The television pictures will be transmitted during a two-week period after passing the planet and Mariner IV is then expected to keep reporting interplanetary data until its communications fade due to sheer distance sometime in September. The spacecraft successfully traversed the Geminid and Ursid meteriod streams in December and in early January reduced its data telemetry rate from 33-1/3 to 8-1/3 bits per second. Additional details on Mariner IV, and particularly the differences between Mariner II and IV, are contained in Reference 7.

3. Voyager

The next stage in the United States interplanetary program after Mariner is known as Voyager, which is now in the study phase. The larger launch vehicle which may be used for Voyager will permit a substantial increase in the size and weight of the spacecraft which can be placed on an interplanetary trajectory, compared to Mariner. The payload capability for the Atlas-Agena for Mars trajectories is in the order of 250 kg., whereas the Saturn has a capability of placing several thousand kilograms on a trajectory to Mars, depending on the selection of upper stages and the particular launch opportunity. The much greater weight capability of Saturn will provide the means for missions to Mars which will make use of orbiting vehicles, landers, and probes.

The Space Science Board of the National Academy of Sciences stated in October, 1964:

> "Accordingly, the Space Sciences Board of the National Academy of Sciences designates the exploration of the nearer planets as the most rewarding goal on which to focus national attention for the 10 to 15 years following manned lunar landing."

Before returning to Voyager, and some of the possible concepts which may be used, the problems associated with interplanetary exploration are worthy of some discussion.

C. SCIENTIFIC OBJECTIVES

1. Mission Objectives

Justification for the exploration of interplanetary space rests ultimately on the scientific knowledge to be gained. Consequently the first factor to be considered in any analysis is the scientific mission and specifically, the scientific objectives which are expected to be achieved. Three primary objectives, in broad terms, can be stated as follows:

1. Exobiological investigations

2. Geological and geophysical measurements

3. Acquisition of data useful for manned landing

A fourth objective, while not strictly in the same class as the first three, is to obtain information which will contribute to the success of subsequent interplanetary missions. That is, measurements should be made which will furnish data useful in the design of future spacecraft.

Exobiological investigations are given first priority for several reasons. First, the observations which have been made of Mars, and current knowledge of that planet, indicate that of all planets in the solar system, Mars is the one most likely to provide conditions necessary for the development and existence of lifeforms. Second, information concerning the possible existence of extraterrestrial life should be obtained on the first missions, particularly prior to manned landings on Mars, to minimize the possibility of contamination of the Mars environment with lifeforms which are transported there from Earth. Finally, and perhaps most significant, the discovery of a life mechanism which is different from those available for study on Earth may provide greater insight into the various evolutionary paths and the origin of life. Similarly, new bodies of descriptive geophysical and geological data from other planets will be very helpful in understanding the vast body of data which has been gathered concerning Earth.

To achieve these objectives, measurements which would be given high priority for Mars are shown in Table 1. They are divided into two categories; those measurements which would be made from a flyby or orbiter and those which would be made by a capsule as it descends through the atmosphere and after it has landed on the surface of the planet (Reference 13).

It is worthwhile to comment further on that fact that valuable scientific information can be obtained from planetary orbiters as well as from landers. This fact makes very attractive an approach in which both types of missions can be combined. It has already been pointed out that Voyager plans to follow this approach, and the concepts which are described later in this paper all make use of a dual mission capability in some form. That is, the initial spacecraft consists of two parts which are separated when they approach the vicinity of Mars. The spacecraft acts as a bus to carry the landing capsule. After the capsule is separated to perform its separate mission, the bus flys by the planet or is placed in orbit about the planet and continues to gather scientific information.

Table 1

High Priority Measurements--Mars

Flyby or Orbiter	Landing Capsule
Television mapping	Television Mapping
Magnetic fields	Biological Detection
Infrared Spectra of Surface	Atmospheric Pressure
Infrared Radiometry of Surface	Wind Velocity
Spectral Albedo	Atmospheric Temperature
Radio Absorption (Lander to Orbiter)	Atmospheric Composition
	Solar Optical Absorption
	Microscopic Examination of Soil
	Magnetic Field
	Density of Atmosphere
	Chemical Structure of Soil

2. Constraints on Landing Site

Before discussing specific landing sites which are suggested for the first Mars missions, it will be helpful to review some of the constraints which influence the selection. These constraints are not related to the scientific value of a landing site choice, but rather they arise from the physical aspects of the problem, the nature of the trajectory, and the desire to use a dual purpose orbiter-lander spacecraft concept. The major constraints can be summarized as follows:

a. The direction of the relative velocity vector of the approaching spacecraft determines the 90-degree entry angle* impact point on the planet.

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* Entry angle is defined as the angle of the velocity vector at the time of entry into the planet atmosphere measured downward from the local horizontal. Thus a 90-degree entry angle corresponds to vertical entry.

b. Circular areas around the 90-degree impact point form the loci of impact points achievable with lower entry angles. The minimum entry angle of about 20 degrees (below which skipout occurs) determines the maximum available landing areas.

c. The requirement to take television pictures from the capsule during its descent through the Mars atmosphere necessitates an impact point on the sunlit of the planet.

d. In order to monitor the performance of the capsule and to receive transmitted scientific data from the capsule instruments via direct link to Earth during descent and immediately after touchdown, it is necessary to land on the side of the planet which is facing Earth at the time of impact. Landing sites in the vicinity of the pole facing away from Earth would be unsatisfactory at any time of day.

e. In order to maintain communications between orbiter and lander for the maximum period of time during lander entry and impact, the lander impact point cannot deviate too far from the locus of sub-orbiter points.

For a particular date and time of arrival, the constraints listed above define a zone of acceptable landing sites on the surface. On some trajectories, there may be no area which meets all requirements, and some relaxation of the requirements will be necessary. Alternatively, considerable variation in the trajectory parameters is possible with only a small penalty in payload weight, so that such things as time of arrival, direction of the approach velocity vector, and plane of the spacecraft orbit can be changed to increase the regions in which acceptable landing areas can be located.

The acceptable landing area can be thought of as a region fixed in inertial space for any given day. As Mars rotates on its axis, the surface of the planet revolves "under" the fixed landing area. In effect, the restrictions on the landing area serve to determine only latitude limits on the available landing sites. Any desired longitude of landing can be selected by adjusting the time of arrival.

3. Landing Site Selection

Having determined the areas where landing is possible, it is next necessary to select those areas where landing is desirable from a scientific point of view. Since several missions to Mars are planned, it would be desirable to select landing sites for each mission which are as different as possible; that is, sites located in regions having different geological and climatological environments, in order to maximize the information returned. Also, in view of the high priority given to exobiological investigations, first choice of landing sites should be given to those areas which are considered most likely to have the conditions required for the development and existence of life. Broadly speaking, the topological features of Mars can be divided into two general areas. These are the maria, or dark areas, which because of their changing appearance with the season, give the impression of being characterized by vegetation. The maria cover a large portion of the southern hemisphere, with fewer isolated areas in the northern hemisphere. Second are the vast desert-like areas which, because of their color, account for the reddish appearance for which Mars has long been known as the "red planet." The best known features of Mars are the extensive network of so-called canals and oases which cover the planet surface.

The seasonal changes on Mars are not unlike those found on Earth. At the autumnal equinox, clouds begin to form in the polar region, and grow steadily in extent until by mid-winter they form a continuous cover over middle and high latitudes. Shortly before the vernal equinox, this cloud cover lifts and exposes the surface polar cap which then begins to recede toward the pole. As this recession takes place, various areas begin to turn darker, this darkening effect progressing as a wave towards and across the equator (Reference 14).

Based on the characteristics of Mars which have been observed, it is possible to select landing sites which should offer the most information in support of the scientific objectives. The dark areas are generally considered the most likely locations for the existence of some type of life process, and would in all probability be the first choice for a landing site. A large, prominent site such as Syrtis Major would be a likely candidate for the first landing. Since the wave of darkening is indicative of dynamic behavior of some sort on the surface of Mars, it is particularly desirable to locate a landing capsule at a suitable location just prior to the peak of the darkening effect, and observe the darkening through its peak and subsequent lightening.

The polar caps are regions of interest, particularly the changes which occur in the dark collar which hugs the caps as they recede each spring.

Finally, the desert areas and so-called canal features should be studied.

When the desired landing sites are compared with the constraints placed on latitude of the landing sites and the arrival dates, it is possible to make a logical selection of sites to be given first priority during each of the launch opportunities. Figure 3 shows a map of Mars with several landing sites identified for each launch opportunity from 1969 through 1975. A more extensive discussion of this selection is contained in Reference 15, from which the map in Figure 3 was taken.

D. MISSION ANALYSIS

1. Typical Mission Profile

The sequence of events for a typical mission to Mars is shown in Figure 4. After injection by the launch vehicle into an interplanetary transfer orbit, the antennas and boom-mounted sensors are deployed. The spacecraft then turns to acquire the Sun and keeps its solar panels facing the Sun to insure a supply of power for the vehicle electrical systems. The craft next rolls around the Sun-line until its star tracker locks on the star Canopus, which together with the Sun, defines the frame of reference for the spacecraft attitude during most of its flight. The spacecraft is held in its prescribed orientation by a reaction control system which uses the signals from Sun sensors and star tracker. The communication antenna is turned through the proper angle to acquire Earth. Scientific data is transmitted to Earth throughout the interplanetary journey.

Shortly after launch the first midcourse correction is made based on trajectory information obtained by tracking from Earth. To perform this maneuver, the craft turns through prescribed angles, using gyroscopes on board the vehicle for reference, and fires its engine for the necessary velocity change. The craft is once again oriented to its cruise attitude. A second midcourse correction may be made one or two weeks after launch, and a third correction a few days before encounter is possible. If the spacecraft contains a landing capsule, then at a range of about one million km. from the planet, lander-orbiter separation will occur. After the lander is placed on its impact trajectory, the spacecraft bus returns to its original orientation, and as it approaches Mars, may make on-board position fixes for terminal guidance. The landing capsule will begin collecting data at entry into the planetary atmosphere. Scientific and engineering data from the lander obtained during entry, descent, and impact will be transmitted directly to Earth, relayed via the spacecraft, or both.

At planet encounter the orbiter is reoriented, and retrothrust is applied to achieve the proper orbital injection velocity. After establishing the desired orbit, the spacecraft will once again be turned to its Sun-oriented attitude, and will begin taking television pictures and other scientific measurements from instruments which will be pointed at the planet surface.

2. Interplanetary Trajectories

If the orbits of Mars and Earth were coplanar, the minimum energy interplanetary trajectory would be a Hohmann transfer, in which the Earth at departure and Mars at encounter would be separated by a heliocentric angle of 180 degrees, and flight times would be in the order of 7 months.

Since the orbits are not coplanar, the actual trajectory varies from this, and may require flight times as long as one year, but 6 to 9 months is more typical. The minimum energy launch opportunity occurs once every 25 or 26 months for Mars. For Venus, trip times of 4 to 5 months are typical, with launch opportunities occurring every 19 months. The dates for several opportunities are listed in Table II.

Table II

Minimum Departure Velocity Launch Dates

Mars	Venus
5 January 1967	11 June 1967
3 March 1969	13 January 1969
24 May 1971	19 August 1970
30 July 1973	25 March 1972
15 September 1975	7 November 1973
	6 June 1975

Since the Earth and Mars orbits are neither coplanar nor circular, the injection energy requirements vary with each launch opportunity, as shown in Figure 5. The figure shows the injection velocity, or burnout velocity, for type I trajectories. These are trajectories which traverse a heliocentric angle less than 180 degrees.

Many other factors must be taken into consideration, however, in selecting the launch date and trajectory. A very significant constraint arises due to range safety limitations. The maximum Earth parking orbit inclination that can be achieved from Cape Kennedy, Florida, is approximately 34 degrees. On some dates, the minimum launch energy interplanetary trajectory requires that the Earth parking orbit from which it departs have an inclination greater than that consistent with range safety requirements. In such cases a dogleg or plane change maneuver is necessary, resulting in significant payload reductions. Injection from a parking orbit is preferred to direct ascent into the interplanetary trajectory to permit greater flexibility in launch azimuth and launch time. Figure 6 shows the injected payload capability for several launch vehicles. The shaded region indicates the range of typical Mars injection velocities.

The minimum launch energy trajectory permits the greatest payload to be placed on an interplanetary course. However, if the objective is to achieve orbit around Mars with the maximum payload, the approach velocity relative to Mars must be considered. The maximum weight injected into a planetocentric orbit occurs when the hyperbolic approach velocity is a minimum, for a given departure velocity. Since the minimum approach velocity is not, in general, associated with the minimum departure velocity, the optimum trajectory requires a compromise between these two parameters. The analysis is further complicated when a split payload (orbiter-lander) is utilized. Figure 7 shows the approach velocity corresponding to the minimum injection velocity and also the minimum achievable approach velocity, which of course corresponds to a higher injection velocity. Consequently the optimum date for the launch opportunity for any particular mission may vary a few months from the dates listed in Table II. Typically the opportunity lasts for only a month or two; before or after that time the injection energy requirements increase rapidly, as can be seen from Figure 5.

Another important consideration in selecting the orbit is the direction of the approach asymptote; this is important because of the previously mentioned desire to approach the sunlit side of the planet and also to be within sight of Earth at that time.

Still another factor to be considered is the relative location of Mars, Earth, and the Sun at the time of encounter. If the angle between Mars and the Sun as viewed from Earth is too small, tracking of the spacecraft and communications may become impossible because of the interference from the Sun's electromagnetic radiation. Fairly wide variations in all of these trajectory parameters can be achieved without large payload penalties by varying the interplanetary launch date and flight time.

3. Selection of Orbit or Fly-By Path

For either a flyby or orbiter mission, the prime consideration in selecting the plane of the trajectory relative to Mars is the latitude of areas of interest, since the longitude can be controlled by adjusting the time of arrival in the case of a flyby, or simply by waiting to pass over the desired longitude in the case of an orbiter. Many of the considerations which lead to the selection of landing sites qualify those same sites as points of interest for observation from orbit. Since Mars rotates on its axis once each 24 hrs. and 37 minutes, a polar orbit will permit observation of all surface features over a sufficiently long period of time. Also, the declination of the approach asymptote determines the minimum possible orbiter inclination, so that equatorial orbits in general are not possible except with large fuel expenditures. Consequently a high inclination orbit is generally desirable, but for a flyby mission, the plane of the trajectory might have a somewhat lower inclination to observe more points of interest in the vicinity of the equator.

The orbit altitude and minimum flyby distance must also be chosen. It is at first glance desirable to pass as close as possible to the planet. On the other hand, because of the importance of avoiding contamination of Mars and its atmosphere by microorganisms from Earth, it is necessary to insure that the spacecraft does not pass through the atmosphere of Mars. (As will be explained later, the spacecraft will probably not be sterilized, but the landing capsule will have to be.) Because of imperfect guidance on board the spacecraft it is necessary to bias the nominal aiming point away from Mars to insure that accumulated errors will not result in an orbit which is too close. The minimum periapsis altitude to insure a lifetime of 50 years for the unsterilized orbiter may be as high as 4,000 km., depending on the ellipticity of the orbit. An elliptical orbit, with its lower periapsis, permits better resolution from TV pictures of the surface, in the vicinity of the subperiapsis point. (This point will change in latitude due to the effect of Mars oblateness.) An elliptical orbit will also require less energy for planetocentric orbit injection, thus enabling the use of larger orbiter payload. This can be seen in Figure 8 which shows the retrograde velocity required to achieve orbit, as a function of apoapsis altitude, for several approach velocities.

4. Lander-Orbiter Relay Geometry

In split payload missions it may be necessary to relay information from the lander to Earth by way of the orbiter. Therefore the orbit selection and lander location are inter-related. It is particularly important that this communication link be available during descent of the lander and for its first minutes of lifetime on the surface. To insure that this is possible, it is necessary for the lander to lead the spacecraft sothat the orbiter will not pass out of view over the lander horizon too soon. This can be achieved by either slowing down the orbiter or accelerating the lander at the time of separation; more will be said about this later. Lander-orbiter communication requirements restrict landing sites to an area within approximately 30 degrees central angle of the orbiter trajectory plane.

5. Choice of Single or Split Missions

Valuable information can be obtained from both a lander and an orbiter, as has been seen; lack of knowledge of the Mars surface features and atmosphere make the design of a very large lander a high risk endeavor. Availability of planned launch vehicles will permit spacecraft weighing several thousand kilograms to be placed on trajectories to Mars. For early missions a split mission spacecraft consisting of an orbiter and lander has the following advantages:

a. The ability of an orbiter and stationary lander to acquire more information together than either one alone.

b. The ability of the orbiter that would make scientific measurements to serve also as a relay for the lander, thus increasing the reliability of the lander-Earth link by adding a redundant path to the direct transmission link.

c. The capability of a split payload to maximize the chance of obtaining both an orbiter and lander for a given launch opportunity.

d. The greater utilization of the components of the spacecraft. Since the orbiter will serve as its own bus, it can also serve as a bus for a lander. Alternately, the lander requires a bus, but the bus with added propulsion capability can also serve as an orbiter.

Considering the booster payload capabilities and the energy requirements as a function of launch date, the split payload concept is quite attractive for 1971 and 1973. For later missions, the option exists of using basically the same orbiter as a flyby and going to considerably larger landers; since the flyby requires no orbit injection propulsion the weight savings can be applied to the lander. The use of larger landers in later missions is logical, since by then better information about the atmosphere and surface of Mars will permit a design which can use the additional weight in a more nearly optimum fashion for scientific exploration.

6. Separation of Lander and Orbiter

Since the lander must follow an impact trajectory to the planet and the orbiter must pass the planet at an altitude of at least 4,000 km., one or the other must change its trajectory at the time of separation. Since the orbiter with its propulsion and guidance is well equipped to perform such a maneuver, it seems reasonable to aim the orbiter-lander combination at the desired landing site, then change the orbiter trajectory at the time of separation. There is one major difficulty with this approach, and that is the possibility of contamination of the planet by the unsterilized orbiter. If the orbiter maneuver fails, it would be doomed to impact the planet along with the lander. To avoid contamination of Mars, the orbiter as well as the lander would have to be sterilized. While this may be possible, it is not attractive, and most studies have adopted the technique of placing the orbiter-lander on a flyby trajectory which is suitable for the orbiter, and at the time of separation changing the lander trajectory to aim for the desired impact point. Not only must the lander aiming point be changed, but the lander must also be accelerated to achieve a suitable lead time between lander and orbiter for communications purposes, as mentioned earlier. An alternative to accelerating the lander is to slow down the orbiter. Depending on detail design concepts, each approach has its advantages and disadvantages. The main effect is the accuracy with which each maneuver can be achieved, and this in turn depends on the nature of the guidance, control, and propulsion system contained on both lander and orbiter. Separation must occur in the vicinity of 1 million km from Mars to avoid an excessive propellent weight penalty for the separation maneuver, but lander accuracy will necessarily be degraded as the separation range increases.

7. Spacecraft Orientation

Since solar cells will provide the power for future spacecraft for some time to come (at least until radioisotope power sources of larger size than are now available become competitive in size, weight, and reliability), the solar panels must be oriented towards the Sun at all times except during maneuvers and other orientations. On the other hand, many of the scientific instruments, such as TV cameras, must be pointed at the planet. The question is, during orbit, should the solar panels be rigidly mounted to the spacecraft with scientific instruments mounted on a gimbal, or vice versa? Most studies have resolved this in favor of fixed solar panels, primarily because of the large solar panel area required and the difficulties of mounting these on a suitable gimbal, whereas the TV cameras and other instruments are more conveniently packaged on a compact gimbal structure. Much is to be said for each approach, however, and the final choice must involve detailed design considerations of weight, performance and reliability.

8. Direct Versus Relay Communications

The large quantity of scientific information obtained by the lander requires a high-capacity information channel to Earth. This can be provided with reasonable power by a direct link using a high gain antenna on the lander. Power requirements are also reasonable if the information is relayed via the orbiter using a low gain antenna on the lander and placing the high gain antenna on the orbiter. (Actually, the high gain antenna is already required on the orbiter to transmit TV data.) The disadvantage of the direct link is the uncertainty in being able to design a lander which can be erected after impact so that its large antenna can be pointed toward Earth. The low gain lander antenna (which is used to transmit to the orbiter) can be designed to operate in any orientation, and so provides a more reliable technique for communications, at least until more knowledge is obtained about the atmosphere and terrain of Mars.

9. Reliability

Perhaps the single most important factor in the design of interplanetary vehicles is the stringent reliability requirement. Interplanetary craft must function reliably for many months, during a period when they are subjected to a series of hostile environments, including the vacuum of space, solar radiation, and micrometeriod impact, to mention just a few. Aside from this, the long operating time of all systems, followed by the critical events which occur at the time of encounter, place formidable demands on the entire system. In those areas where the component or subsystem reliability is not adequate, the use of redundancy is a possible solution. For example, in the reaction control system, two completely redundant systems might be employed to guard against leakage of gas from a single system. This approach to reliability by means of redundancy must be used with caution, however, or the weight available for payload is rapidly diminished. It is more desirable, where possible, to design the equipment (or to modify its design) to achieve the desired reliability without redundancy.

10. Sterilization

Since the primary scientific objective in the exploration of Mars is the performance of exobiological investigations, it is of utmost importance that all possible care be taken to avoid introduction of viable organisms into the Mars atmosphere and planet surface which may be carried there on the spacecraft from Earth. If organisms are introduced which have their origin on Earth, then the possibility exists that these earthly organisms would be identified by the exobiological experiments. Thus the value of the experiments would be nullified, and the most promising opportunity in history to gain new understanding about our evolutionary process would be lost.

The current goal in the United States space programs is to achieve a probability of less than one in ten-thousand (10^{-4}) of landing a viable terrestrial microorganism on Mars during any single launch attempt. This probability number was selected on the basis that it gave a reasonable confidence over the next 15 years of being able to carry out exobiological missions without contamination, taking into account the number of failures, and the fact that ultimately man will go to Mars, making the introduction of earthly organisms a virtual certainty at some date in the future (Reference 16).

To achieve this objective, there are two approaches one can take. First is to intentionally fly wide of the planet, thus avoiding it and also avoiding the problem. The second solution is to sterilize the spacecraft. Before discussing sterilization, it is worthwhile to make a few brief comments on the first approach. On early missions, such as the Mariner IV, and even for later missions as well, all or part of the spacecraft is intended to fly by or orbit the planet. In that case it is clearly an unnecessary complication to sterilize the spacecraft and it is necessary only to insure that the guidance of the craft is such as to insure that it will not hit the planet when the objective is to fly by or orbit. Since the spacecraft guidance is not perfect, it is possible that when aiming for a close approach to the planet, an error in trajectory could result in an impact. Therefore, the aiming point must be intentionally offset far enough to insure that the probability of accidental impact is less than 10^{-4}. For an orbiter the initial aiming requirement is the same; in addition, the orbit selected must be high enough to ensure that it will not decay for at least 50 years.

For the lander which is intended to enter the atmosphere and land on the surface of Mars, there is no alternative to sterilization. The method of sterilization which is preferred at the present time is the use of dry heat to bring the capsule and all its components to a temperature of 135°C for at least 24 hours. The effect of this temperature cycle is to reduce the burden or population of viable organisms by a factor of about 10^{-13}. If the population is sufficiently small at the onset of the sterilization cycle, it can confidently be predicted that the possibility of a single viable organism remaining at the end of the cycle is less than a specified amount. The fact to be noted is that the vehicle must have some minimum biological burden at the outset; this level, while not unmanageable, still requires that all manufacturing, assembly, and test operations must be conducted in "clean rooms," where biological contamination can be controlled. Prior to terminal heat sterilization the entire landing capsule would be placed in a sealed container which would not be opened until shortly before the capsule is placed on its impact trajectory. Some components may be unable to sustain the high temperature for sterilization without failure. If these components cannot be replaced with equipment which can tolerate the temperature requirements, some other method must be used for their sterilization, such as gas (ethylene oxide), radiation, chemicals, etc. These components would then be assembled to the sterilized lander before its sterilization shroud is sealed.

E. DESIGN CONSIDERATIONS

Some of the critical design problems in both lander and spacecraft are discussed in this section. For more thorough discussion the reader is referred to the references; space limitations prevent more than a superficial treatment here.

1. Communications

Communications equipment on the spacecraft are required to transmit scientific and engineering data to Earth, to respond to Earth-based tracking signals for range and range rate information, to receive commands from Earth, and to act as a relay between lander and Earth if that mode of operation is employed. The major constraint is the high data rate required to send TV information during the 6 months lifetime of the orbiter; rates of 4500/sec may be required.

Communications for the lander will require even higher data rates for the relay link--as high as 10,000 bits/sec. However, these rates would be required for only that time when the orbiter passes over the vicinity of the lander. During the remainder of its orbit, the spacecraft would relay the data to Earth at a lower rate. For direct communications between lander and Earth a data rate of 1500 bits/sec might be required for early landers. This would require a steerable antenna on the lander which could be deployed after impact and point towards Earth. The lander must also have the capability of receiving commands directly from Earth.

2. Power Sources

Of major importance to the design of both lander and spacecraft is the source of power. Early missions will undoubtedly use solar cells for the spacecraft because of their proven reliability. They will be combined with batteries to handle peak loads. On the other hand, solar cells are not suitable for the lander, because of its operation on the surface and the problems of deployment of solar panels, atmospheric attenuation, and restriction to daytime operations. In order to achieve long life (6 months or more) batteries are out of the question, although they may be used for early small probes and capsules having limited lifetime. Power limited devices are more suitable for long missions, and the most promising candidate is the radioisotope thermo-electric or thermionic generator (RTG). However considerable development work is required to bring these devices to the status required for reliable lightweight spaceflight applications at the power levels which would be required (upwards of 100 watts.) The heat dissipation of the RTG will require careful design considerations, particularly during the transit phase of the mission. Looking far into the future, the RTG will also be a promising candidate on the spacecraft, since it will eliminate the deployment problems of solar panels and the degradation of solar cells due to effects of solar radiation. Particularly for missions to the distant plants, the RTG will become essential.

3. Guidance and Control

The use of solar panels for power immediately dictates the primary attitude reference of the spacecraft; namely, the Sun. The control of the spacecraft attitude around the Sun line can most conveniently be accomplished by reference to a star near the ecliptic pole. Canopus, which is near the south ecliptic pole, and also the second brightest star in the heavens, is well suited for this purpose. Orientations would be carried out with the aid of gyro references, and commands for midcourse maneuvers would be determined from Earth based tracking and transmitted to the spacecraft. This method is far more accurate than on-board celestial navigation, at least with current and projected state of the art. The exception may be in the immediate vicinity of the planet, and particularly during orbit injection. Here on-board astro-inertial guidance will permit more precise control of the approach trajectory and final orbit injection. The reaction control system for attitude control may be the biggest single item of the guidance and control system, in terms of weight, if current thinking in favor of a cold gas system prevails. Much of the weight requirement results from the necessity to provide large factors of safety to provide for the contingency of a valve sticking open during the long mission. The use of the resistojet for attitude control (Ref. 17), perhaps combined with subliming solid fuels, offers promise of substantial weight reduction.

The lander will probably be spin stabilized during the firing of its rocket motor; it is not likely to require any other attitude control. After landing on the planet, it must have a means of determining its position in order to point its antenna to Earth. This will require a local vertical sensor and sun tracker, as well as a clock and ephemeris data for Earth and Mars. (More advanced landers may require terminal guidance to adjust the approach trajectory to achieve precise landing point control. They may also require guidance during descent in order to achieve soft landings.)

4. Propulsion

The spacecraft requires propulsion for midcourse corrections as well as for placement into the planetocentric orbit. The propulsion system must therefore be capable of multiple restarts, high reliability, and high performance. Systems using liquid space storable propellants appear to be the better choice, but advances in the state of the art of solid propellants still make them a potential candidate. Thrust levels are not particularly critical for this application, and engines developed for other missions may very well be adapted to meet the spacecraft requirements. For the lander, a propulsion system is required to produce the velocity change needed to place the lander on an impact trajectory after separation from the spacecraft bus. A fixed impulse engine will simplify the design, but the choice is complicated by the necessity to select a propellant which can successfully withstand dry heat sterilization. Here again, the choice between solid and liquid propellants is still not clear cut.

5. Landing Capsule Design

The external design of the landing capsule is influenced almost entirely by the atmospheric entry requirements. For flexibility in operational use and landing site selection, entry angles between 20 and 90 degrees below the horizontal may be employed. To obtain atmospheric measurements, the capsule must have sufficient deceleration at high altitude. If the final landing is to be made by means of a parachute, the design must provide for its deployment.

The attempt to design a vehicle to meet these requirements is greatly complicated by the lack of knowledge of the Mars atmosphere. Models having surface pressures between 10 and 40 millibars are being used, but some experts believe the pressure may be over 100 mb. (Ref. 18) If the atmosphere is as tenuous as the lowest surface pressure would suggest, the vehicle must have a very small weight to drag ratio (W/C_DA), in the order of 30 kg/m^2. Since C_D (drag coefficient) cannot change over very large ranges, this means that A (frontal area) must be increased; this reaches a limit imposed by launch vehicle diameter. Consequently very severe restraints are placed on the maximum weight (W) which can be placed in the lander. It is quite possible, for example, that large launch vehicles will be unable to use their full weight lifting capability because they are size limited, rather than weight limited. Some of the earliest landing capsule missions may not be expected to survive landing at all. Most capsules, however, in order to obtain any useful information on the surface must have a reasonably soft touchdown. This means that a deceleration system other than just the drag of the entry body must be provided. Because of the large diameter parachute required for final descent, an auxiliary drogue chute will probably be deployed at supersonic velocities, followed by deployment of the main chute at subsonic velocity. Impact attenuation will still be required to cushion the landing shock and protect delicate scientific instruments.

More advanced landers may make use of retropropulsion for a completely controlled soft landing, such as is planned for Surveyor in landing on the Moon. One disadvantage of this technique is the effect which the rocket exhaust will have on the surface in the vicinity of the landing site; it may seriously limit the usefulness of experiments which depend on taking samples of the surface near the touchdown point. Of course the ultimate goal will be to equip the lander with a roving vehicle which can move away from the landing point for more extensive investigations. (References 19, 20, & 21)

6. Thermal Control

Of major importance to the design of both spacecraft and lander, is the provision of means for insuring that internal temperatures of components will stay within the operating range. The solar radiation input will change as the spacecraft changes its distance from the Sun, during interplanetary flight. Also, the part of the spacecraft which is normally in the shadow of the solar panels will be exposed to solar heating during reorientation maneuvers. The lander will have to be designed for the thermal environment it experiences while it is still attached to the spacecraft, and also after separation during its approach to the planet. The lander will also require thermal control during its operation on the planet surface, when both diurnal and seasonal temperature changes will occur. Passive thermal control using properly selected coatings, and active techniques using movable louvers, together with careful packaging of the components, are among the methods which may be employed.

F. DESCRIPTION OF SEVERAL DESIGN CONCEPTS

A number of studies have been and are now being conducted to determine the most suitable design of capsules, landers, and spacecraft. The proper evolution from Mariner IV, the first spacecraft for exploration of Mars, depends on booster capability, funds available, and the degree to which each vehicle design should lead logically to the next. The approaches to be described include: (1) a small, non-survivable entry capsule to sample the atmosphere; (2) a fly-by lander combination; and (3) an orbiter lander combination.

1. Mars Atmosphere Probe

The purpose of this small entry capsule proposed by Seiff and others (Reference 22) is to obtain a reliable definition of the properties of the Mars atmosphere before attempting the design and landing of larger, more complex vehicles on the surface. The measurement of the atmosphere will not only aid in the biological investigations but will also serve an important scientific objective in itself.

The proposed technique involves measurement of the response of an entry vehicle to the atmosphere. The NASA Ames Research Center has shown that by measuring the deceleration, radiation, temperature, and pressure experienced by such a probe it is possible to deduce the properties of the Mars atmosphere with considerable accuracy. The drag deceleration is sensed by accelerometers and is integrated to provide density as a function of altitude, knowing the initial velocity at the time of entry from deep-space tracking of the spacecraft during interplanetary flight. The accuracy of the density measurement can be improved by low altitude subsonic measurements of pressure and temperature.

Measurement of atmospheric composition would be made by taking advantage of the radiative emission from the gases in the shock layer which is present during entry. Measurement of radiation intensity of properly selected frequency bands permits identification of the constituents of the atmosphere from the known radiative characteristics of various gas mixtures.

The vehicle design selected for this experiment is a sphere having a W/C_DA of 40 kg/m^2 (See Figure 9). The blunt shape is needed to provide a shock-layer for radiation measurement, and the low W/C_DA is required to provide sufficient time for communication during entry after blackout and before impact. The spherical shape also simplifies the deduction of density from drag deceleration, since drag is the only steady force acting on a sphere and is independent of the angle of attack. Preliminary estimates of such a probe have indicated high confidence in achieving a design having a diameter of 0.6 to 1.0 m, and a total weight of 11-16 kg. Payload weight would be under 7 kg. The probe would be carried to the vicinity of Mars on a spacecraft such as Mariner.

The probe is equipped with a telemetry transmitter which would be used to transmit measured data to the flyby bus, from which it would be relayed to Earth. The instrumentation and telemetry would be battery operated, and the device would not be designed to survive impact; its mission would consist only of making and transmitting measurements during descent through the atmosphere.

2. Flyby Lander

Next to be described is Avco's concept for an Advanced Mariner spacecraft which would use the Atlas Centaur launch vehicle, and would consist of a flyby bus and a lander.

The payload capability of the Atlas Centaur for a typical Mars mission is about 650 kg. The primary purpose of this flyby lander (Ref. 23 and 24) is to demonstrate the capability of successful landing and survival on the planetary surface for several hours. In order to achieve this objective and also to obtain the broadest spectrum of scientific information, the choise of a combined lander and fly-by bus was selected. The spacecraft, shown in Figure 10, will separate from the Centaur booster after it has been placed on its interplanetary trajectory. Solar panels will be deployed after separation and the spacecraft will be oriented with a cold gas reaction control system to keep the solar cells pointed toward the Sun. These cells will convert the Sun's rays into electrical energy which will be used to charge the spacecraft batteries. Control of the spacecraft attitude in roll (that is, its rotation about the sunline) will be obtained by use of a star tracker which senses the direction to the star Canopus. An antenna on the spacecraft will receive commands from Earth and transmit scientific data accumulated in transit. Approximately one day after launch enough tracking information will have been obtained to allow a midcourse correction to be made. The spacecraft will be oriented by command from the Earth to the proper attitude and its rocket fired to make the necessary velocity change. A second midcourse correction can be made if necessary one or two weeks later. Approximately 9 months will be required for the interplanetary flight, and during that time no additional maneuvers will be performed.

At a distance of about a million kilometers from the planet, the lander will be separated from the bus, spun up, and the solid propellant rocket will be ignited to place it on an impact trajectory. Meanwhile the bus will continue on its flyby course, and will obtain 100 T.V. pictures with a resolution of one kilometer. Its point of closest approach to the planet will be 6500 km. Ten days will be required to play back the television data to earth, as the bus continues on into orbit about the Sun.

The lander will be decelerated in the Mars atmosphere. After it slows sufficiently, a drogue chute is deployed, followed after further deceleration by a main chute, as shown in Figure 11.

During its descent on the parachute, atmospheric measurements will be transmitted to the flyby bus from which they are relayed to earth. The heat shield will be jettisoned when the main parachute is deployed. After impact, the aluminum honeycomb "crush-up" material which protects the payload will peel away to expose scientific instruments and an antenna which will transmit collected data to earth directly and also via bus relay. The weight which will be available for the scientific payload depends strongly on the assumptions made regarding the atmosphere of Mars. This information is expected to be considerably improved in the months ahead, but indications are now that the surface pressure on Mars may be as low as one percent of the atmospheric pressure on Earth. Since this condition may be accompanied by high surface winds, the lander design will be conservative; that is, it will be designed to function under the most pessimistic conditions. It will also be simple in concept,to achieve maximum reliability; hence the selection of a passive crush-up technique to absorb the loads of impact. Since the scientific payload weight is limited, its lifetime after arrival at Mars will be only a few hours, operating from battery power. Nevertheless, this will be sufficient to accommodate a life detection experiment. Figure 12 shows a cut-away drawing of the lander.

3. <u>Orbiter Lander</u>

A more ambitious orbiter-lander type of spacecraft, such as might be used in the Voyager program, will be described next. The Saturn launch vehicle has a wide range of payload capabilities depending on the combination of upper stages which are used. The Avco study (Reference 13) was based on an S-VI upper stage which would be capable of placing a spacecraft weighing

2700 to 3200 kg on a typical Mars trajectory. In order to achieve all of the objectives and obtain the broadest spectrum of scientific information, the choice of a combined lander and orbiter is particularly attractive.

As in the case of the Advanced Mariner, the Avco concept of the Voyager spacecraft will be separated from the launch vehicle after it is placed on its course to Mars, and oriented so that its solar panels face the Sun. The significantly larger spacecraft weight introduces several modifications to the mission capability. Whereas the lifetime of the Advanced Mariner lander previously described is relatively short, the Voyager lander will continue in operation for six months after arrival at Mars. Furthermore, the Voyager spacecraft will be placed in orbit around Mars after separation of the lander, and will continue for six months in its most important scientific mission of taking television pictures of the planet surface. Figure 13 shows the spacecraft on its interplanetary trajectory.

The in-transit and lander separation phases of the mission will be similar to those already described. When the orbiter bus reaches its point of closest approach to Mars, the on-board guidance and control system on the orbiter can be used to make final adjustments to insure that it enters the desired orbit. The main propulsion system of the orbiter will be ignited and will burn for about 15 minutes to place the spacecraft in a Mars orbit. A large 2.5 m diameter antenna will be deployed to allow a greater data transmission capacity to earth. The scientific platform, which will be pointed towards the surface of the planet, will record television pictures and other scientific measurements. The platform will also be equipped with an antenna which will be used to communicate with the lander.

Returning now to the lander, after separation from the spacecraft it will be spun up by rockets and then its own propulsion system will be fired to give it the necessary velocity change. The lander will decelerate in the atmosphere of Mars and when it reaches a fairly low velocity, a drogue chute will be deployed which will pull away the heat shield and release the landed payload. The final descent to the surface will be made either with the use of a parachute or with the aid of a propulsion system to provide a soft landing. The lander is acorn shaped and except under the most adverse terrain conditions will assume a more or less upright attitude after impact, as shown in Figure 14. When the sides of the lander are deployed, they will force the lander into an erect position. This in turn will allow the antenna to be raised and pointed towards Earth. The lander contains a Sun tracker as well as a device to measure the direction of the local vertical. This information together with knowledge of time and ephemeris data, is sufficient to determine the pointing direction to Earth. The lander is equipped with a variety of instrumentation to carry out its scientific mission. The data which it accumulates is transmitted directly to Earth by the 1.5 m parabolic antenna. A VHF self-leveling antenna is also provided so that data can be transmitted from lander to orbiter and then relayed to Earth as an alternate communication link in case of failure of the direct transmission.

The concepts which have been described for Voyager represent the results of studies performed by Avco. Studies are still underway, and other design approaches which are being considered are described in Reference 25.

G. ACKNOWLEDGEMENT

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Figure 1 MARINER II SPACECRAFT

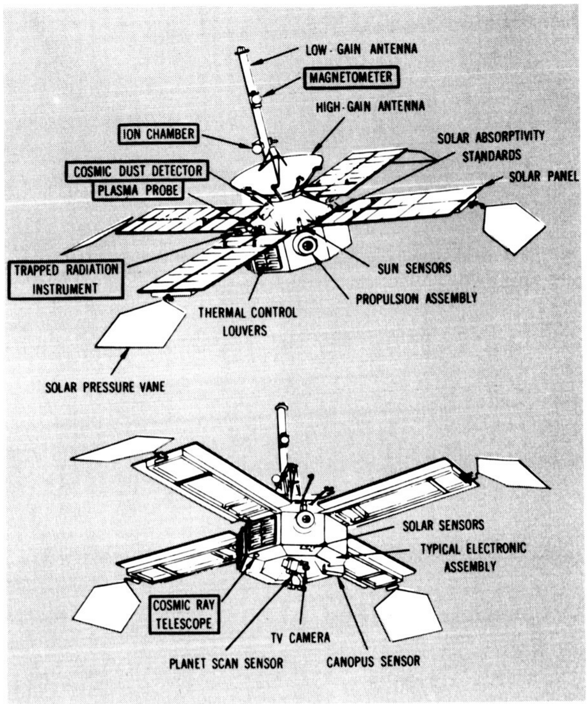


Figure 2 MARINER IV SPACECRAFT

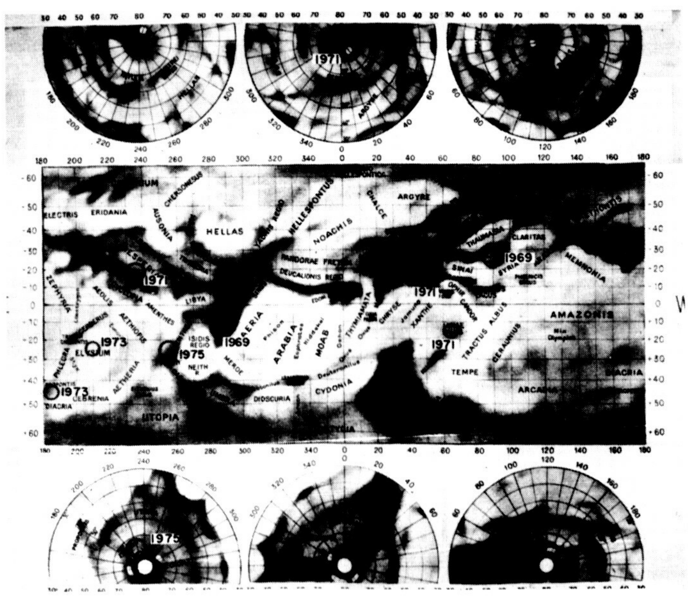


Figure 3 MARTIAN LANDING SITES

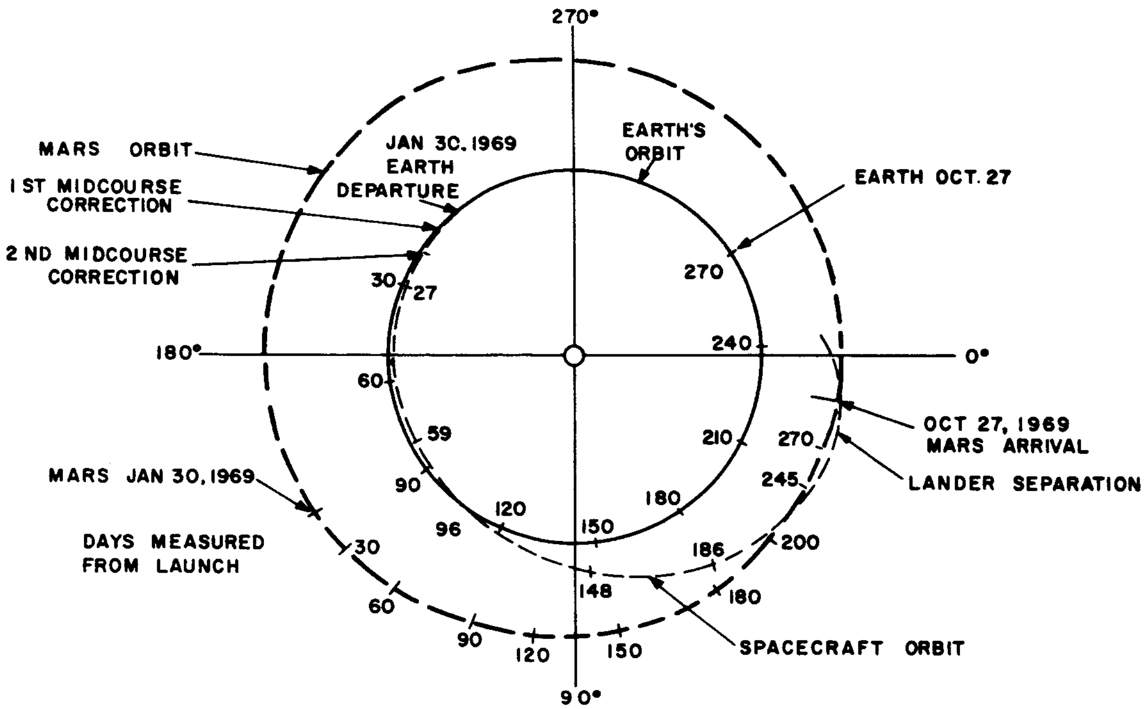


Figure 4 TYPICAL INTERPLANETARY EARTH TO MARS TRANSFER

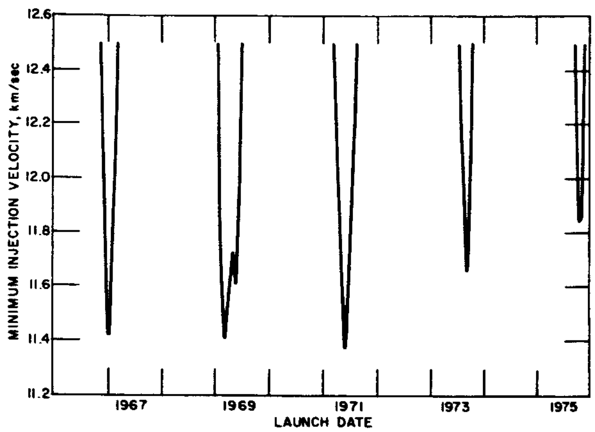


Figure 5 MINIMUM INJECTION VELOCITY VERSUS
LAUNCH DATA MARS TYPE I

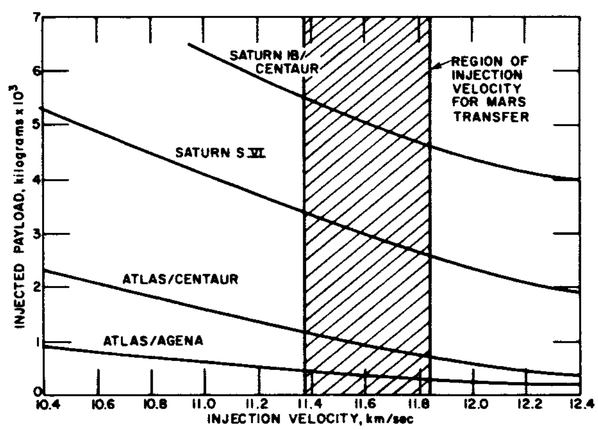


Figure 6 INJECTED PAYLOAD CAPABILITY FOR
VARIOUS LAUNCH VEHICLES

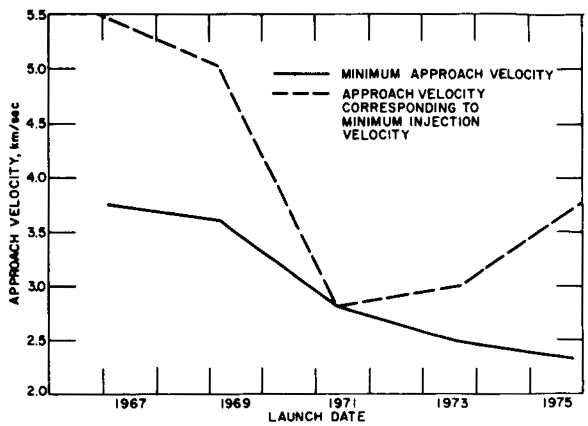


Figure 7 APPROACH VELOCITY VERSUS LAUNCH DATA MARS TYPE I

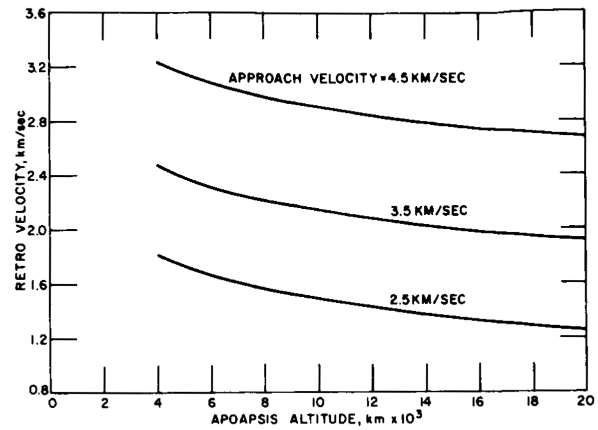
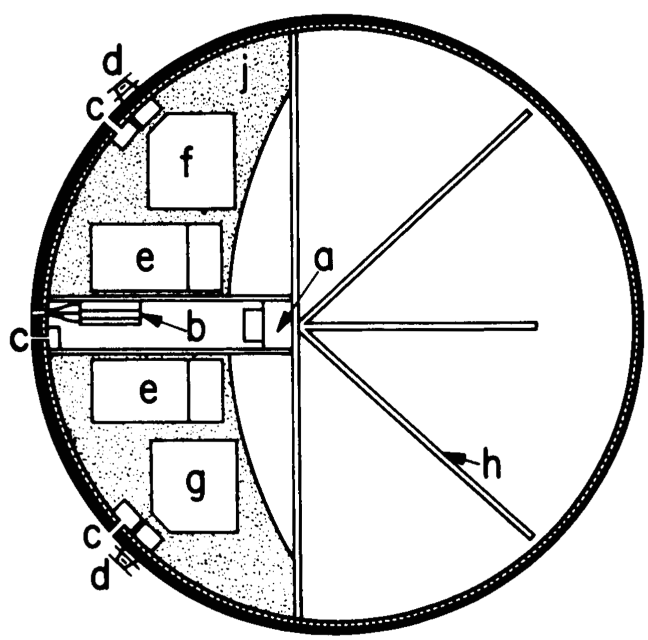


Figure 8 RETRO VELOCITY REQUIRED TO ORBIT VERSUS APOAPSIS ALTITUDE PERIAPSIS ALTITUDE = 4000 KM



A. THREE-AXIS ACCELEROMETER ON CG. B. FOUR-CHANNEL RADIOMETER. C. PRESSURE SENSORS. D. STATIC-TEMPERATURE SENSORS (DEPLOYED AT LOW SPEEDS). E. TELEMETRY TRANSMITTER. F. DATA-STORAGE UNIT. G. BATTERIES AND POWER CONDITIONING. H. TRANSMITTING ANTENNA. J. PLASTIC FOAM.

Figure 9

1. SOLAR CELLS
2. LIMIT CYCLE SUN SENSOR
3. ATTITUDE CONTROL JETS
4. COLD GAS TANKS
5. LANDER PROPULSION
6. CANOPUS TRACKER
7. AQUISITION SUN SENSORS
8. HEMI-OMNI ANTENNA
9. HIGH GAIN ANTENNA
10. LANDER SPIN UP ROCKETS
11. GIMBALED PAYLOAD PLATFORM

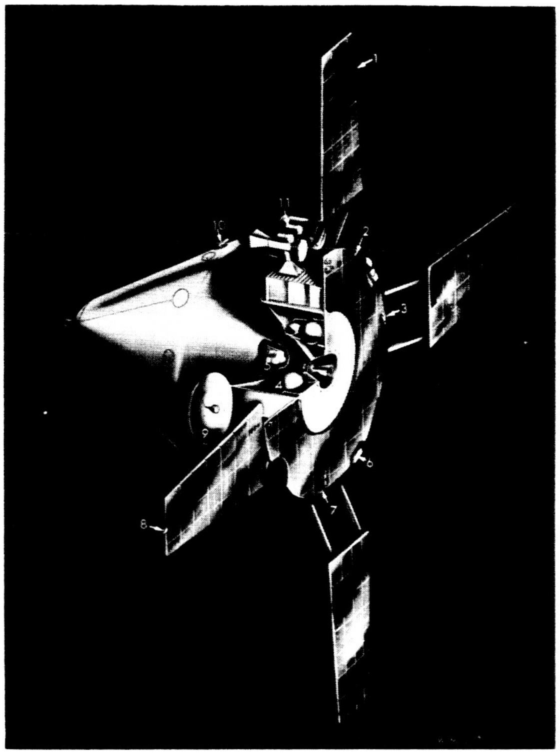


Figure 10 AVCO'S CONCEPT OF THE ADVANCED MARINER SPACECRAFT

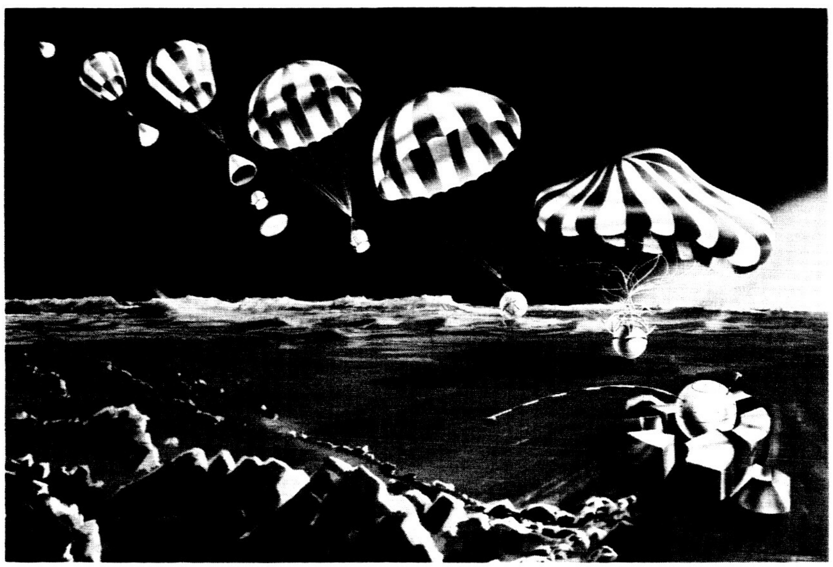


Figure 11 ENTRY AND LANDING SEQUENCE

1. DROGUE ATTACHMENT
2. BERYLLIUM AFTERBODY
3. PAYLOAD FLOTATION SYSTEM
4. DROGUE CHUTE
5. PRE-ENTRY COMMUNICATION SYSTEM
6. PAYLOAD SUPPORT RING
7. DESCENT COMMUNICATION SYSTEM
8. HORN ANTENNA (S-BAND)
9. FOREBODY HEAT SHIELD SYSTEM
10. MAIN CHUTE
11. IMPACT ATTENUATION SYSTEM
12. LANDED PAYLOAD COMMUNICATION SYSTEM

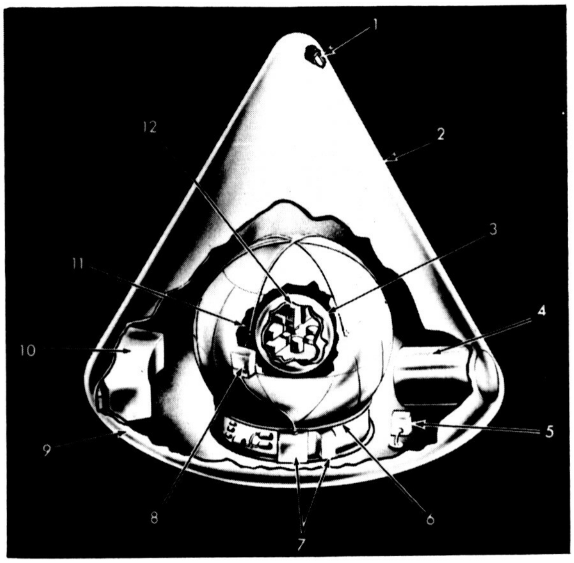


Figure 12 CUT-AWAY DRAWING OF THE ADVANCED MARINER LANDER

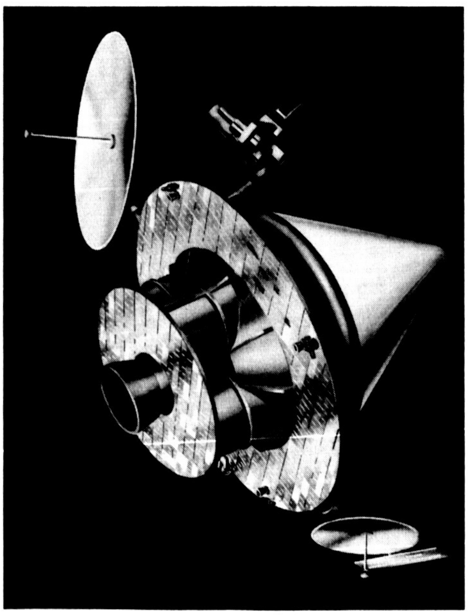


Figure 13 AVCO's CONCEPT OF A MARS VOYAGER SPACECRAFT

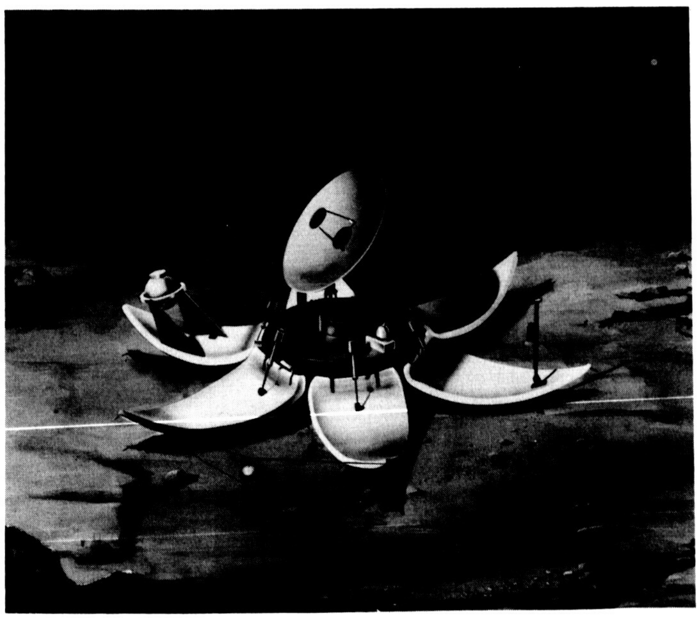


Figure 14 AVCO'S CONCEPT OF A MARS VOYAGER LANDER