# Phobos/Deimos Missions

E. Brian Pritchard* and Edwin F. Harrison†
*NASA Langley Research Center, Hampton, Va.*

## Theme

LANDER, Lander/Orbiter, and sample return missions to the Satellites of Mars, Phobos and Deimos are analyzed for launch opportunities from 1977 to 1981 with both Titan-Centaur and Space Shuttle-Centaur launch systems. In addition, both single satellite and dual satellite missions are examined for the case of lander-only missions.

## Content

The science payload which would be desirable for Phobos/Deimos missions is quite similar to a geologically oriented payload for a Mars surface mission, as one might expect. Table 1 presents a representative science package for a Phobos or Deimos lander mission.

Missions to Phobos or Deimos are similar in many respects to more conventional Mars missions. The launch opportunities and mission constraints are the same, but there are significant differences. These involve the operations at Mars illustrated schematically in Fig 1. In general, Mars arrival conditions are such that the plane of the capture orbit is greatly inclined to the equatorial plane (20° or more), wherein lie the orbital planes of Phobos and Deimos. Because plane changes are most efficiently performed at low orbital speeds, the capture ellipse must have a high eccentricity and the plane changes must be performed as near to the apoapsis as possible.

Figure 1 shows the sequence of events for lander-only missions. For lander/orbiter missions, the two spacecraft would be separated after a station-keeping mode had been established at the satellite. A small $\Delta V$ would be given to the orbiter to establish a circular or elliptic orbit relative to the satellite. For sample return missions, the departure sequence of events would be the reverse of the arrival sequence.

The major $\Delta V$ items for missions to Phobos and/or Deimos are: a) the Earth departure $\Delta V$ (injection energies are 13 to 14 $km^2/sec^2$ for missions in the 1977 to 1981 time period), b) the $\Delta V$ to inject into the capture orbit, c) the $\Delta V$ to perform the necessary plane change, d) the $\Delta V$ to transfer to the intermediate phasing orbit, e) the $\Delta V$ to circularize the orbit preparatory to rendezvous, and f) the rendezvous $\Delta V$. Since b, d and e all nominally occur at periapsis of the approach hyperbola, it is convenient to sum the three into an impulsive $\Delta V$ to transfer from the approach hyperbola to a circular orbit at the satellite altitude. The 1977, 1979 and 1981 $\Delta V$ requirements, summarized in Table 2, are separated into primary velocities defined impulsively for the major maneuvers and secondary finite velocities. The primary velocities are applicable for either Phobos or Deimos missions if capture orbit eccentricities of 0.8 and 0.6 are selected, respectively, to equalize the plane change $\Delta V$. The secondary finite velocities were developed from comparison with the Viking mission. For instance, the 100 m/sec estimated for gravity losses during MOI is based on the use of Viking orbiter engines; but, three main burns are performed instead of the single major burn of the Viking system. It should also be noted that the arrival $\Delta V$ is even higher for the 1984 opportunity. Thus, the total $\Delta V$ requirements increase for every launch opportunity from 1979 through 1984.

For satellite lander missions a total $\Delta V$ budget of 2330 to 2850 m/sec is required, depending on the mission opportunity. This may be compared to the Viking mission where the $\Delta V$ budget is about 1500 m/sec. A sample return mission would

**Table 1 Science payload for Mars satellite lander**

| Instrument | Purpose | Mass, kg | Power, w |
|---|---|---|---|
| Television camera systems (2) | Provide high resolution images of entire surface of satellite | 30 | 45 |
| Alpha backscattering experiment | Provide information on elemental composition of soil | 1 | 3 |
| X-ray fluorescence spectrometer | | 4 | 2 |
| X-ray diffractometer | Identify types and relative abundance of crystalline phases present in soil | 5 | 4 |
| Seismometer | Determine tectonic activities and interior structure | 3 | 10 |
| Landing dynamics-soil mechanics experiment | Investigate surface physical properties | 1 | 1 |
| Drill ($\simeq$ .3 m deep) | Acquire samples | 8 | 100 |
| | | 52 | |

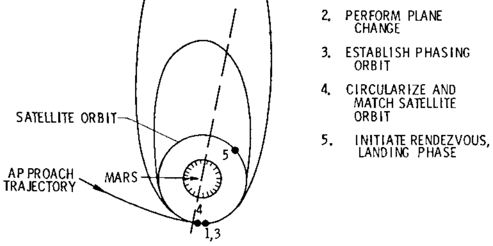


**Fig. 1 Mission schematic.**

Presented as Paper 71-830 at the AIAA Space Systems Meeting, Denver, Colo., July 19–20, 1971; submitted July 26, 1972; synoptic received November 10, 1971; revision received April 3, 1972. Full paper is available from AIAA. Price: AIAA Members, $1.50; nonmembers, $2.00. Microfiche, $1.00. **Order must be accompanied by remittance.**

Index categories: Unmanned Lunar and Interplanetary Systems; Lunar and Planetary Trajectories; Spacecraft Mission Studies and Economics.

*Head, Mission Analysis and Applications Branch, Space Technology Division. Member AIAA.

†Head, Mission Analysis Section, Space Technology Division. Member AIAA.

Table 2 Phobos/Deimos arrival and departure velocity requirements

a) Primary velocities

| Earth launch date | 1977 | 1979 | 1981 |
|---|---|---|---|
| Impulsive circularization $\Delta V$ at arrival, m/sec | 1850 | 2020 | 2300 |
| Orbit plane change $\Delta V$, m/sec | 79 | 121 | 195 |
| Impulsive $\Delta V$ at departure, m/sec | 1950 | 1950 | 1950 |
| Orbit plane change $\Delta V$, m/sec (at departure) | 79 | 121 | 195 |

b) Secondary (finite) velocities

| | $\Delta V$, m/sec |
|---|---|
| Midcourse | 25 |
| Gravity losses during moi | 100 |
| Navigation uncertainties | 175 |
| Orbit trims | 50 |
| Rendezvous and landing on satellite | 50 |
| | 400 |

require a $\Delta V$ budget of 5500 m/sec plus a $\Delta V$ allotment for retrobaking at Earth if that mode were selected rather than aerobraking. This would include 400 m/sec of finite velocities on the outbound leg of the mission and about 150 m/sec on the inbound leg for midcourse maneuvers and gravity loss compensation.

Using standard information to provide launch vehicle capabilities, spacecraft mass allowances may be defined for the missions considered here. The baseline propulsion system used in this analysis is the Viking '75 propulsion system. In order to accommodate the science package of Table 1, it appears that a landed mass roughly equivalent to the Viking landed mass (about 500 kg) would be required for Phobos/Deimos landers, particularly if maximum use is to be made of Viking technology and hardware. On this basis, Table 3 indicates that the Titan IIIC is probably adequate for lander missions in 1977 and 1979. Use of the Titan IIID-Centaur provides the capability to place Viking sized landers on both Phobos and Deimos or to deliver a lander-orbiter combination to one satellite.

The landed mass for a 1977 mission (Table 4) is defined by the Titan IIID-Centaur capability—1270 kg which is divided as shown for the Lander/Orbiter and Dual Satellite Lander missions. In both cases the allowable lander mass is of the same order as the Viking landed weight and hence should be adequate for the missions. The sample return mission requires an ascent stage of 860 kg to return 11 kg samples and film to Earth. This includes a 27 kg system designed to retrobrake to a loose Earth orbit and cruise support module with a mass of 250 kg. These systems requirements, when subtracted from the landed weight allowance of 1270 kg, leave an allowance of 410 kg for the lander system which remains at the satellite and includes the surface science and sample collection systems and support structure and equipment. This should be adequate since it is of the same order as the Viking landed weight.

Table 3 Effect of launch vehicle and launch year on systems mass for lander missions

| Launch date | Launch vehicle | Injected mass, kg | Usable systems mass, kg Lander | Lander | Orbiter |
|---|---|---|---|---|---|
| 1977 | Titan IIIC | 1770 | 530 | | |
| 1979 | ↓ | 1660 | 440 | | |
| 1981 | | 1730 | 370 | | |
| 1977 | Titan IIID- | 4110 | 1270[a] | 820 | 450 |
| 1979 | Centaur | 4000 | 1100[a] | 650 | 450 |
| 1981 | ↓ | 4040 | 910[a] | 460 | 450 |

[a] About one-half of this mass could be delivered to each satellite during a lander mission to both Phobos and Deimos (two spacecraft on the same launch vehicle). Note: Viking Landed mass is about 500 kg.

Table 4 Comparison of system mass of various mission modes (1977 launch with Titan IIID-Centaur)

| Mission mode | System | Mass, kg |
|---|---|---|
| Lander | Landed | 1270 |
| Lander/Orbiter | Lander | 820 |
| | Orbiter | 450 |
| Sample return | Landed | 1270 |
| | Left on Satellite | 410 |
| | Ascent | 860 |
| | Return to Earth orbit | 27 |
| | Cruise support module | 250 |
| | Samples and Film | 11 |
| Duel satellite Landers | Landed on Phobos | 590 |
| | Landed on Deimos | 590 |

Looking at the spectrum of launch opportunities (Table 5), it is clear that the Titan IIID-Centaur is inadequate for sample return missions in 1979 and 1981 if we assume that the discarded systems at the satellite weigh on the order of 400 to 500 kg. The Space Shuttle-Centaur operating in a mode which delivers a fully loaded Centaur plus spacecraft to low Earth orbit can easily accommodate the sample return mission.

Other options which were considered in the study of sample return missions in the order of decreasing effectiveness were: a) leave the cruise module in the initial capture orbit and perform a rendezvous on the return leg of the mission; b) develop a new propulsion system (an $LiAlH_4/H_2O_2$) hybrid system was used in the study $-I_{SP} = 350$ sec; and c) use a spin stabilized cruise support module based on the Planetary Explorer system and weighing 140 kg.

Table 5 Effect of launch vehicle on Mars satellite sample return missions

| Launch date | Launch vehicle | Support module mass, kg | Propulsion at Mars $I_{sp}$, sec | Mass left on satellite, kg Retro to Earth orbit | Direct entry at Earth |
|---|---|---|---|---|---|
| 1977 | TITAN IIID- | 250[a] | 286[b] | 410 | 300 |
| 1979 | CENTAUR | | | 210 | 100 |
| 1981 | ↓ | | | — | — |
| 1977 | TITAN IIIF- | | | 850 | 740 |
| 1979 | CENTAUR | | | 590 | 480 |
| 1981 | ↓ | | | 290 | 200 |
| 1977 | SPACE SHUTTLE- | | | 1680 | 1570 |
| 1979 | CENTAUR | | | 1300 | 1210 |
| 1981 | ↓ | ↓ | ↓ | 930 | 840 |

[a] Based on the 3 axis-stacilized spacecraft proposed for the Viking '75 alternate mission mode.
[b] Assumed Viking orbiter type engines.