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Mars Sample Return: A Direct and Minimum-Risk Design

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Current NASA strategy for Mars exploration is to seek simpler and more reliable missions with focused science objectives. This requirement has made untenable most previously proposed Mars sample return missions. The mission proposed here achieves a simplified mission profile by leveraging interplanetary trajectory design, limiting surface science operations, and using advanced propulsion and thermal protection systems. The fast mini direct Mars sample return mission presented could bring back a 0.5-kg sample from the surface of Mars with a total mission duration of less than 1.5 Earth years using a single, medium-lift launch vehicle. The mission constraints require an aggressive design that dictates the use of advanced storable liquid propulsion systems, high-performance thermal protection system materials, and limited spacecraft design mass margins. The key elements are detailed for a mission that eliminates the some of the high-risk operations that were a part of previously proposed Mars sample return missions that used propulsive insertion into orbit at Mars, complex rover operations on the surface, orbit rendezvous, and in-orbit sample transfers.

Nomenclature

C_3	= launch injection energy, km^2/s^2
I_{sp}	= specific impulse, s
$m/C_D A$	= ballistic coefficient, kg/m^2
V_∞	= hyperbolic excess velocity, km/s

Introduction

SINCE the successful Viking landers sent back many intriguing pictures of Mars and much scientific information beginning in the late 1970s, scientists and mission planners have constantly sought a mission that would return samples of the Martian surface to Earth. This mission is viewed as a key step towards the exploration of Mars.

Mars sample return (MSR) missions were proposed even before the U.S. Viking spacecraft landed.[1,2] To send a spacecraft to Mars, enter the atmosphere, land on the surface, and return a sample to the Earth requires complex mission planning and spacecraft design. Critical mission elements for MSR missions include interplanetary trajectories for both Earth-to-Mars and Mars-to-Earth transfer legs, atmospheric entry, descent and landing, sample acquisition, orbital rendezvous, and sample transfer, and such missions have become larger in scope and objectives. In the 1980s, MSR became Mars rover sample return (MRSR) as rover development and technology became more tangible.[3–5] However, these missions typically involved multiple launchers, long duration interplanetary transfer legs and surface stays, complex rover operations, and rendezvous in Mars orbit for sample transfer. Furthermore, the missions generally cost from \$2 billion to \$10 billion (using dollar values from the late 1980s). Current NASA strategy for planetary exploration is to seek simpler, cheaper, and more reliable missions. This requirement has left virtually all previously proposed MRSR missions economically untenable.

In recent years, MSR mission planning has sought to limit operational complexity and science objectives to reduce mission cost.[6,7] The MSR mission proposed in this paper represents an economical, back-to-basics approach to mission design. By optimized interplanetary trajectory design, short-duration surface operations, and state-of-the-art propulsion and thermal protection systems, this mission achieves significant mass reduction and simplified mission operations. The proposed concept, called the fast mini direct Mars sample return (FMD-MSR) mission, represents the cheapest and fastest class of missions that could return a 0.5-kg sample from the surface of Mars to Earth with a total mission duration of less than 1.5 Earth years using a single launch vehicle.

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Mission Overview

The FMD-MSR would use a single Atlas II-AS launch vehicle to place a 1750-kg payload on a fast, type I trajectory to Mars with a launch injection energy C_3 of 15 km^2/s^2. Launches from Earth have been examined for opportunities from 2003 to 2009. The 2003 launch opportunity has been chosen as the baseline mission scenario. (A discussion of the effect of 2005, 2007, and 2009 opportunities on the mission design will be presented later in the paper.) The launch payload consists of the Mars entry vehicle (MEV) and the trans-Mars cruise stage (see Fig. 1). The cruise stage supports only those subsystem functions that are necessary en route to Mars, including midcourse correction maneuvers, communications, power, guidance, and navigation. After a 5-month journey to Mars, the MEV separates from the cruise stage and directly enters the Martian atmosphere from a hyperbolic approach trajectory. No spacecraft components are inserted into Mars orbit on arrival at the planet.

The MEV contains the entry and descent system (aeroshell, aftcover, parachute system, and terminal descent propulsion system) and the Mars ascent vehicle (MAV) attached to the lander support structure. The lander contains the surface science and sample acquisition instruments. Figure 2 shows the MEV–MAV vehicle, including the first- and second-stage propulsion systems, and the Earth return vehicle (ERV). Surface operations last only one week after initial landing, thus minimizing power and other subsystem requirements. By this means, the total mission duration (about 1.5 years) is minimized, in contrast to other sample return scenarios that involve operations of long Mars surface duration or launches from Earth over several launch opportunities.

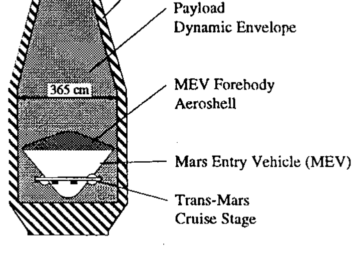

Fig. 1 FMD-MSR launch configuration in Atlas II-AS launch vehicle shroud (launch vehicle payload adapter not shown).

Table 1 Mars sample return: mass estimate

System	Mass, kg
ERV	
SRV	18
Power and comm.	19
Structure	17
Support systems	6
Total	60
MAV	
ERV	60
1st-stage propulsion (wet/dry)	940/90
2nd-stage propulsion (wet/dry)	200/20
Total	1200
MEV	
Aeroshell structure and TPS	190
Parachute subsystem	60
Descent propulsion	90
Lander support structure	40
Science instruments	40
Support systems	70
MAV	1200
Total	1690
Launch payload	
Cruise stage	60
MEV	1690
Total	1750

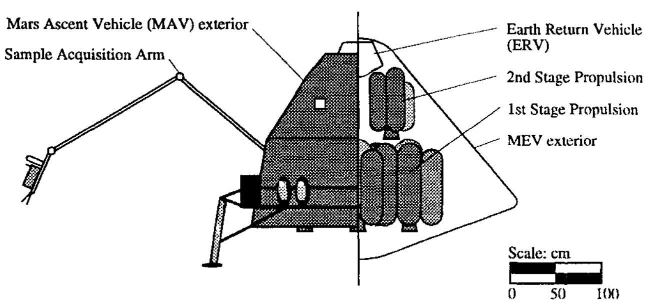

Fig. 2 MEV–MAV configuration for Mars surface operations.

The MAV lifts off with 0.5 kg of surface samples taken from the lander's immediate vicinity and stored in the sample return vehicle (SRV) located on the forward end of the ERV. The first stage of the MAV propulsion system inserts the vehicle into a 250-km Mars parking orbit. After achieving orbit, the MAV orbit and timing sequencing are precisely determined, using communication links to the Deep Space Network (DSN), in preparation for the trans-Earth injection maneuver. No orbital rendezvous at Mars or Earth or sample transfer is required in the present scenario, thus reducing the risk and complexity of the mission.

After one or two days in Mars orbit, the second stage of the MAV inserts the ERV on a high-energy Earth-return trajectory. A high-energy return trajectory is required because of the nonoptimum relative positions of Earth and Mars. The return transfer to Earth takes about 9 months. About 1 day prior to arrival at Earth, the SRV separates from the ERV. The 20-kg SRV directly enters the Earth's atmosphere at a velocity of about 14.5 km/s. The SRV is a spartan vehicle containing the Mars surface sample and uses only battery power and passive transponders. After deceleration in the Earth's atmosphere to subsonic speeds, a parachute is deployed and the SRV is retrieved using an air-snatch. A landing and recovery on the Earth's surface is also possible and would have to be considered in a more detailed study. The baseline mass estimate of the spacecraft components is given in Table 1.

Interplanetary Trajectory Design and Optimization

The FMD-MSR mission timeline and interplanetary trajectories are shown in Fig. 3 for the 2003 launch opportunity. Launch trajectories from the Earth to Mars are characterized by C_3. The value of C_3 determines the allowable mass that can be injected on a planetary escape trajectory for a given launch vehicle. In addition, C_3 constrains the choices for launch and arrival dates (dictated by the ever-changing relative positions of Earth and Mars in their orbits) for the interplanetary trajectories between Earth and Mars. The Atlas II-AS launch vehicle was chosen for this study because of its payload launch mass capability, cost, and availability. A maximum C_3 of 15 km^2/s^2 (injected mass 1750 kg) was chosen to ensure some mass margin at launch and to provide a sufficiently large launch window. Figure 4 is a contour plot of the Earth departure C_3 for the 2003 opportunity. Launch energies and arrival conditions were calculated using patched conic interplanetary transfers as described in Ref. 8. The two regions of minimum C_3 represent the short-trip-time type I trajectories and the longer-trip-time type II trajectories. Subsequently, it will be shown that only short-transfer-time trajectories to Mars are feasible for the current mission scenario.

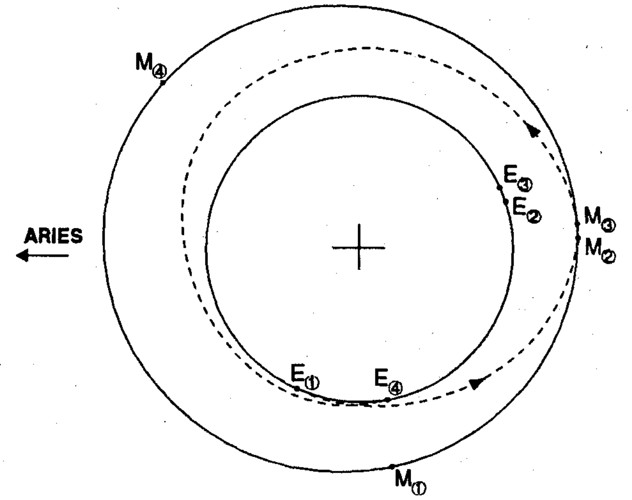

Fig. 3 Ecliptic projection of FMD-MSR interplanetary trajectories: 1, Earth launch, May 30, 2003, C_3 = 15 km^2/s; 2, Mars arrival, Oct. 13, 2003, V_e = 7.1 km/s; 3, Mars departure, Oct. 19, 2003, C_3 = 32.7 km^2/s^2; and 4, Earth arrival, July 13, 2004, V_e = 14.5 km/s.

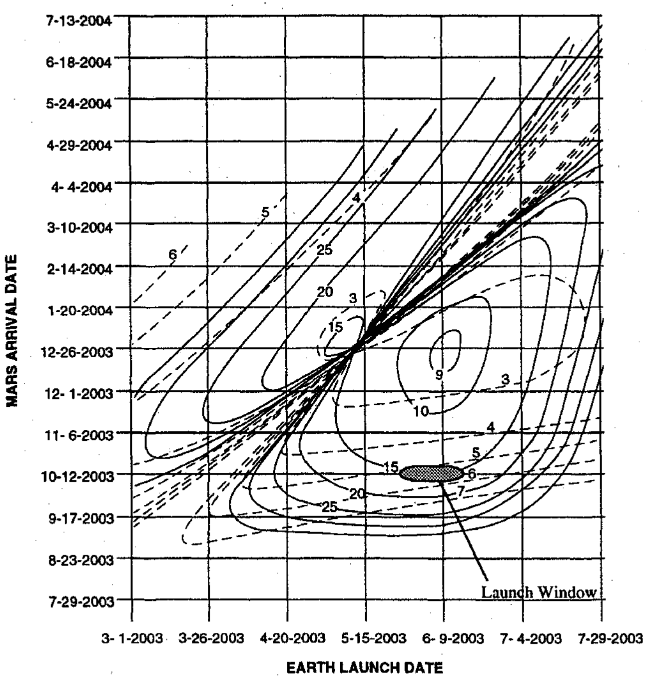

Fig. 4 Earth–Mars trajectory injection (C_3) and hyperbolic arrival velocity (V_∞) contours for 2003 launch opportunity: ——, C_3 (km^2/s^2) and – – –, V_∞ (km/s).

The MEV arrives at Mars with a V_∞ that is somewhat greater than 5 km/s (see Fig. 4). The tradeoff for the mission designer involves minimizing V_∞, which is directly related to the atmosphere entry velocity, while remaining within the constraints dictated by other mission parameters. Some of these parameters include: the choice of Mars arrival date, C_3 restrictions, latitude of landing, and Mars

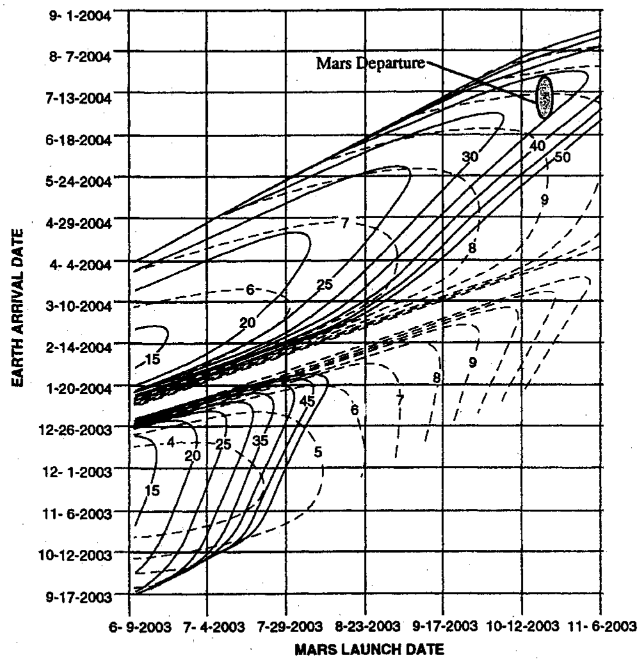

Fig. 5 Mars–Earth trajectory injection (C_3) and hyperbolic arrival velocity (V_∞) contours for 2003 launch opportunity: ——, C_3 (km²/s²) and - - -, V_∞ (km/s).

parking orbit geometry. The FMD-MSR arrives at Mars in October 2003, with an entry velocity of approximately 7.1 km/s relative to the rotating atmosphere. The hyperbolic velocity vector's declination to the Mars equator is 21–22 deg at arrival, which yields favorable atmospheric entries in a posigrade direction at low-latitude equatorial sites.

Figure 5 shows the C_3 contours for the Mars departure trajectories. To minimize the C_3 for Earth return, the MEV needs to arrive at Mars as early as possible (see Fig. 5), i.e., a short Earth–Mars trajectory leg is important. The Mars arrival date in Fig. 4 needs to be as early as possible because the arrival date and the surface duration directly affect the Mars departure date. The short surface stay time of about 1 week was chosen as a feasible minimum to perform the necessary operations. As the Mars surface stay time increases, Earth return trajectories require excessively large C_3 penalties that result from delaying the Mars date of departure (for the mission scenario presented). Only when surface stays approach 200–400 days does the next opportunity for Earth return becomes possible with relatively low C_3. This delay greatly extends the overall mission duration and increases mission cost and complexity. To reduce complexity and landed mass, rover operations are excluded in this mission. Surface samples are obtained through a robotic arm or appropriate sample acquisition device. Previously proposed MRSR missions that included rover operations could not support brief surface stay times because rover operations required long operation periods.

The later launch opportunities following 2003 result in more difficult missions due to the less favorable interplanetary trajectory energetics. The Earth–Mars trajectory requires a higher C_3 than in the 2003 opportunity, which reduces the available injected mass. For these opportunities, $C_3 = 20\,\mathrm{km}^2/\mathrm{s}^2$ is needed, which reduces the injected launch mass by about 150 kg. Either a scaled-down MEV spacecraft or a more energetic Atlas II-AS (possibly with a higher-performance third stage) would be required because of the higher C_3 requirements. Table 2 gives a summary of the launch and arrival dates for the later mission opportunities. The Earth launch dates shown are centered on a 15-day launch window. Furthermore, the Earth return velocities are significantly higher for these later opportunities and thus require a more capable heat shield.

Mars Entry and Descent

The MEV enters the Martian atmosphere on a hyperbolic trajectory with an entry velocity, relative to the rotating atmosphere, of 7.15 km/s and a −15-deg flight-path angle at an altitude of 125 km.

Table 2 2005, 2007, and 2009 launch opportunities

Launch opportunity	Earth launch	Mars arrival	C_3	V_∞
2005	8/7/05	1/11/06	20	5.0
	Mars launch	Earth arrival	C_3	V_∞
	1/8/06	10/14/06	42	15.3
Launch opportunity	Earth launch	Mars arrival	C_3	V_∞
2007	9/20/07	3/2/08	20	6.5
	Mars launch	Earth arrival	C_3	V_∞
	3/9/08	11/21/08	42	16.8
Launch opportunity	Earth launch	Mars arrival	C_3	V_∞
2009	10/25/09	4/25/10	20	6.5
	Mars launch	Earth arrival	C_3	V_∞
	5/2/10	11/4/10	43	16.7

The entry vehicle is protected from the atmospheric heating by an aeroshell composed of a thermal protection system (TPS) consisting of ablation material that is bonded to a support structure. The aeroshell base diameter is limited to 3.6 m by the Atlas II-AS shroud diameter. The aeroshell geometry is based on the 70-deg-half-angle, spherically blunted cone (nose radius of 0.9 m). The design, based on the Viking landers,[9] was originally proposed for the MESUR mission[10,11] and is being used on Mars Pathfinder.[12] The MEV enters on a ballistic (no lift) trajectory with aerodynamic properties that have been previously examined.[13] Entry trajectories were simulated using an Adams–Moulton predictor–corrector integration algorithm. The design entry mass of the MEV is 1690 kg, which yields a ballistic coefficient ($m/C_D A$) of 98 kg/m². The MEV entry trajectory (see Fig. 6) results in a peak stagnation point pressure of 0.30 atm and a peak deceleration load of 17 Earth g. Furthermore, the peak nonablating stagnation point heating rate is 108 W/cm², and the corresponding integrated stagnation point heat load is about 3600 J/cm², which is only about 10% greater than the heating expected during the Mars Pathfinder entry.

The relatively high stagnation pressure (about 21% higher than for the Mars Pathfinder) is of concern in the MEV TPS design, especially when considering SLA-561, which was the ablator that was used on the Viking missions and is being used on the Mars Pathfinder mission. As a result of extensive arc jet testing of SLA-561 at NASA Ames,[14] a pressure limit of 0.25 atm was established, which will require a TPS material other than SLA-561 for the MEV. Furthermore, the extreme need to minimize spacecraft mass, because a large fraction of the mass landed on Mars must be devoted to the ascent propulsion system, demands the use of a capable, yet low density, ablator for the MEV aeroshell. One material, currently being developed and tested at NASA Ames, is SIRCA[15] (silicone-impregnated reusable ceramic ablator). Initial tests of this material, which is slightly lighter than SLA-561, indicate the ability to survive higher pressures and heating rates than SLA-561. Using past experience for supporting Mars entry mission heat shield design and estimating aerothermodynamic heating,[16–18] the MEV is designed to have a 6% TPS mass fraction (fraction of the entry mass) by using the SIRCA advanced ablator. For comparison, the forebody TPS on Mars Pathfinder is approximately 5–6% of the entry vehicle's mass.

After surviving the heating pulse, the MEV descends to an altitude between 4 and 5 km, where the lander vehicle containing the MAV is pulled from the aeroshell by a parachute. The high ballistic coefficient of the MEV forces the parachute deployment to be supersonic and to occur at a low altitude. This restriction should not greatly constrain the choice of a landing site, since the equatorial region tends to have low elevations. A disk-gap band parachute deploys at a speed of Mach 1.8 and slows the descent to a velocity of 50 m/s. At an altitude of approximately 500 m above the surface, a hydrazine propulsion system, based on the Viking lander concept, controls the terminal descent to the surface, resulting in low landing loads. Hydrazine is used to avoid contamination of the landing site, since its combustion product is water vapor. The lander touches down with a velocity of 2 m/s and begins surface operations.

Mars Ascent and Return to Earth

After a week of surface science operations, primarily involving imaging the landing site and storing samples in the SRV, the MAV

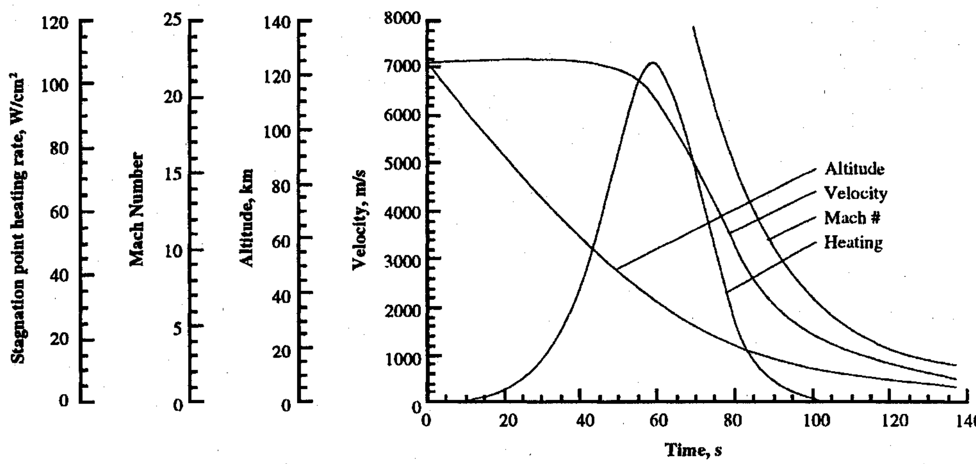

Fig. 6 MEV entry trajectory. Note: COSPAR NS mean atm and $Rn = 0.9$ m. Entry conditions: flightpath angle = −15 deg, entry velocity = 7.15 km/s, entry altitude = 125 km, and $m/C_DA = 97.7$ kg/m^2.

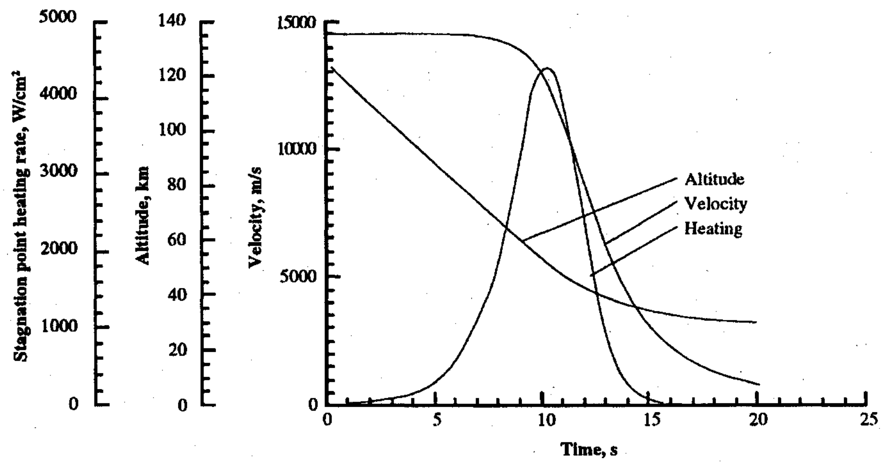

Fig. 7 SRV Earth entry trajectory. Note: $Rn = 0.125$ m. Entry conditions: flightpath angle = −30 deg, entry velocity = 14.5 km/s, entry altitude = 122 km, and $m/C_DA = 42$ kg/m^2.

initiates the launch ascent to Mars orbit. The first- and second-stage MAV propulsion systems are based on the advanced liquid axial stage (ALAS).[19] ALAS is a SDIO-derived system that uses advanced materials and propellants (chlorine pentafloride and hydrazine), which are storable for long durations, and that has been flight-tested. This lightweight system can achieve specific impulses[19] (I_{sp}) of 355 s in the low-pressure Martian environment. The estimated liftoff mass of the MAV is 1200 kg, with about 850 kg dedicated to propellant for the first stage. (Ascent trajectories were modeled using the POST code[20] to determine the propulsion system ΔV requirements.) Simulation parameters are described in Ref. 21 for a single-stage-to-orbit trajectory for the first stage of the MAV.

The MAV is inserted into a 250-km parking orbit after a 5-min ascent burn. The first stage is subsequently jettisoned after achieving orbit. Taking advantage of an eastward launch direction, the net ΔV to achieve orbit is 4330 m/s with a 2.5% loss from aerodynamic drag. The parking orbit is circular with a near-equatorial inclination. This geometry avoids out-of-plane velocity losses during the trans-Earth injection because the Mars departure asymptote declination is about −5 deg.

Prior to beginning the Mars escape maneuver, precise orbit determination and clock sequencing are obtained through telemetry with the Earth-based DSN. Accurate knowledge of the state conditions is critical for controlling the spacecraft's attitude and timing of the propulsion burn, using the second-stage propulsion system, for the Earth return trajectory. The ΔV for the Earth return injection into Mars orbit is about 4050 m/s for a corresponding C_3 of 32.7 km^2/s^2. Approximately 180 kg of propellant is used to send the 61-kg ERV on a Mars escape trajectory. The journey from Mars back to the Earth with the surface samples requires about 9 months. The SRV enters the Earth's atmosphere at a velocity of about 14.5 km/s and encounters a much more severe heating environment than for the Mars entry. Therefore, a more robust thermal protection system is required for the much higher heating rates and heat loads experienced during Earth entry. The ERV performs only the bare minimum of spacecraft functions during the return to Earth to safekeep the Mars surface samples that are located in the SRV.

Earth Entry of the Sample Return Vehicle

About one day prior to entry into the Earth's atmosphere, the SRV separates from the ERV. The SRV is a blunt, 60-deg half-angle, conical entry vehicle with a base diameter of 60 cm and a corresponding $m/C_DA = 42\,\mathrm{kg/m^2}$. The vehicle enters the Earth's atmosphere with a relative velocity of 14.5 km/s and a flight-path angle of −30 deg at 122 km. The entry trajectory is shown in Fig. 7. Because of the higher entry speed, the heating is much more severe than during the Mars entry. The peak (nonablating) stagnation point heating rate for the SRV is 4.4 kW/cm^2 and the stagnation point heat load is 19.2 kJ/cm^2. The TPS material for the SRV is assumed to be a carbon phenolic. Current testing of more advanced TPS materials at NASA Ames potentially could replace carbon phenolic with lighter-weight materials and thus improve the spacecraft mass margins. The estimated mass of the SRV heat shield TPS (based on comparative TPS mass estimates for the Rosetta vehicle[22]) is 3 kg, which is about 17% of the entry mass. For comparison, the TPS mass fraction for the Rosetta vehicle is about 22%.

After surviving the heating pulse during entry, the SRV descends to an altitude between 15 and 20 km, where a parachute deploys to separate the SRV from the hot aeroshell. When the SRV decelerates to a terminal velocity of 5–10 m/s, it is recovered via an air snatch or controlled descent to touchdown. Several methods of accomplishing this are described in Ref. 23.

Conclusion

This paper describes the mission and a conceptual spacecraft design for a FMD-MSR mission. The mission design is constrained by the following guidelines: 1) mission duration less than 1.5 Earth years; 2) single launch vehicle, no rendezvous in Earth or Mars orbit, direct atmospheric entry at Mars and Earth, and minimal surface science operations; and 3) return of 0.5 kg of samples from the Mars surface.

Estimates from this study show that 11% of the MEV mass is dedicated to the aeroshell TPS and structure and 67% is used by the first- and second-stage propulsion systems. The significant mass contribution of these two systems requires an aggressive mission design that dictates the use of advanced storable liquid propulsion systems and high-performance TPS materials to minimize spacecraft mass. Mass margins used in the spacecraft design are tightly constrained and for many systems are only 15%. The short mission duration and limited Mars surface payload preclude the use of a rover vehicle on the Martian surface and limit the scientific investigations. Surface samples are obtained through a robotic arm or a similar device. Analysis shows that the 2003 launch opportunity yields more favorable launch vehicle mass margins than later opportunities.

The sample return mission described in this paper does not contain the potentially high risk operations of other proposed MSR missions such as propulsive insertion at Mars, orbit rendezvous at Mars, rover operations on the surface, and sample transfer in orbit. In a tightly constrained mission cost environment, the FMD-MSR mission is a highly desirable candidate for performing Mars sample return in spite of its reduced surface science operations. It has been shown that the key elements for an aggressive mission for returning surface samples from Mars are feasible.

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