Mars Sample Return,

A Concept Point Design by Team-X

(JPL's Advanced Project Design Team)[1]

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Abstract — A Mars sample return mission will answer critical questions concerning the possibility of life on Mars. Shallow drilling from the Lander provides pristine samples while the rover provides samples from a variety of sites. Included are discussions of a Mars Sample Return mission point design where requirements development, mission design, flight system design and identification of technology developments needed to enable the mission are presented. After launch on a Delta IV Heavy the Lander and Orbiter are separated and individually guided to Mars. The Lander is targeted for direct atmospheric entry where it will use active aerodynamic control to arrive within a few kilometers of the landing site. After surface operations the Mars Ascent Vehicle is launched into low Mars orbit where the Orbiter, after Mars orbit insertion and aero-braking, has arrived to rendezvous with the sample. Once the sample is placed in the Earth Entry Vehicle (EEV), the Orbiter returns the EEV for direct Earth entry. Many technology advances are needed to enable a Mars sample return mission. The spacecraft must be able to perform an autonomous landing on Mars while avoiding hazards. An autonomous Mars launch system will need to be developed along with an autonomous system for rendezvous and capture of the sample.

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INTRODUCTION

Purpose and scope: The Jet Propulsion Laboratory (JPL) and the Mars Program Office have conducted a series of studies to assess the feasibility of a Mars sample return (MSR) mission and to identify the technology development steps needed to better optimize this exploration program. This paper describes the results of a single mission point design, dealing with a concept for a MSR mission. Included are discussions of the resulting mission concept, requirements development, mission design, flight system design and identification of technology developments needed to enable the mission. This mission concept represents one point design or one approach for accomplishing the MSR mission. It is intended to be considered with other MSR concept options to better understand the design trade space and technology development requirements. By exploring a range of options the design trade space can be well understood and a better or more optimized MSR mission will be achieved. Mission performance or science return needs to be weighed against the cost and risk issues of the proposed options. This paper focuses on technical feasibility including requirement development, design issues and technology needs. While cost and risk were assessed as part of the Team-X study, these results are not provided due to the scope constraints of this paper and the proprietary nature of the cost information.

Study Approach — The MSR concept study presented in this paper was sponsored by the Mars Program Office in November 2001 and conducted by JPL's Advanced Project Design Team, Team-X. Team-X uses a concurrent engineering design process to assess cost, risk and performance trade issues for future mission concepts. The Team is composed of science, mission design, spacecraft system and subsystem engineers, ground-system, programmatics and cost analyst specialists.

The Mars Program Office established the mission objectives, science requirements as well as the Mars infrastructure assumptions for the study. Team-X was provided with the proposed Smart Lander mission concept and was instructed to use this concept as the baseline approach for the entry, descent and landing (EDL) systems.

The Smart Lander Mission will be a MSR precursor to qualify needed technology. The Smart Lander design will be scaled up in size for the lager MSR mission. Reuse of Smart Lander algorithms and software are assumed for aero-maneuvering during EDL, landing site targeting while attached to the subsonic parachute, guidance and navigation during powered descent, hazard avoidance and robust landing capability. Accordingly the MSR EDL System and Lander are based on the proposed Smart Lander design. The Smart Lander is expected to achieve a safe precision landing

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near a targeted landing site with a radial error of a few kilometers. In order to meet MSR schedule constraints the Smart Lander project will need to have a successful technology demonstration with a launch no later than 2009.

The rendezvous and capture payload on the Orbiter was not assessed during this study. Team-X used design assumptions based on the CNES Orbiter system proposed for previous MSR studies. The rendezvous payload will provide attitude and delta-V commands to the Orbiter bus where they will be executed.

Other provided elements for this study were the earth Entry vehicle and rover. The JPL Mars Program Office provided the EEV specifications and the MSR rover was assumed to be a Mars Exploration Rover (MER) mission class rover in terms of mass volume and accommodation requirements. The approach for Mars/Earth planetary protection was also provided and while not described in detail is accounted for in the system design and mass lists.

Disclaimer — The MSR concept point design presented in this paper does not represent the baseline or favored approach being considered by the Mars Program Office or JPL. It is intended as one point design in a series of option studies to explore the MSR mission trade space.

2. Science Objectives

The objective of this mission is to return a scientifically selected, documented and interesting sample from the surface of Mars. The sampling site must be well documented to link the sample and surface science to orbital remote sensing and existing Mars knowledge. Fresh rock samples are a key sampling objective. Shallow drilling from the lander is included as is a MER class rover and instruments to both help in the sample collection and in the documentation of the sample site.

Baseline Objectives:

- The objective of the mission is to return samples of Mars material to Earth for analysis.
- The total mass of samples returned by a first mission shall be greater than 500g.
- Returned samples shall include rock, regolith and atmosphere and shall be selected using a payload of scientific instruments and sub-surface sampling tools.
- Sample diversity shall be assured by providing mobility for the sample selection and collection payload.
- Payload mobility shall be no less than 1 km, measured as a radial-distance from the landing site, accomplished in a few months.
- A sample from a single hole of a depth of at least 2m shall be returned.
- Landing sites within 15 degrees of the equator and at altitudes below +1.5 km (with respect to the MGS/MOLA-based mean reference) shall be accessible.

Instruments — The instruments included in the point design are from existing or proposed missions. They are intended to be representative of instruments which will be selected at a future date in terms of mass, power, volume, data rate and other accommodation requirements. Based on this representative instrument set, data storage and return requirements are 1.0 gigabit per day from the surface. Mass estimates for the Lander payload are provided in Table 4.

Orbiter:

1. Optical navigation camera
2. Electra communication package

These sensors have science applicability but are not classified as payload instruments.

Lander:

1. Meteorology package (air temperature, water content, wind speed, pressure)
2. HEDS package (e.g. environment observation, in-situ resource acquisition, 25 kilogram (kg) allocation).
3. Descent Imager
4. Drill/sample arm measurements, magnets and color targets

Rover:

1. Pancam (multi-spectral and stereo imaging)
2. Mini-TES thermal emission spectrometer
3. Microscopic imager
4. Raman spectrometer
5. Mossbauer spectrometer
6. APXS
7. Mini-corer/sample arm

The rover element was not designed in detail and was assumed to be comparable to the MER rover in mass volume and other accommodation requirements.

3. Mission Design

This MSR mission point design will be carried out as follows. In November 2011 a Delta IV Heavy launch vehicle injects two flight elements a Lander and an Orbiter on a type II trajectory to Mars. The Lander and Orbiter are separated at launch and are individually navigated to Mars. The Lander is supported by a Carrier Stage during cruise to Mars where it is separated from the Lander just prior to atmospheric entry. The Orbiter performs a Mars orbit insertion (MOI) maneuver followed by an aero-braking campaign to position the Orbiter in low Mars orbit where small maneuvers will allow it to rendezvous on and capture the Orbiting Sample Canister (OS). Meanwhile, after direct entry into the Martian atmosphere, the Lander uses active aerodynamic control to arrive within a few kilometers of the designated landing site. The MSR EDL approach will be demonstrated by the Mars Smart Lander mission. During the three month surface stay a rover will be deployed to gather samples that will be transferred to the OS carried by the Mars Ascent Vehicle (MAV). The MAV is then launched into low Mars orbit where the OS remains attached to the MAV upper stage during the Orbiter rendezvous effort. The OS is released from the upper stage by command from the Orbiter. Once separated from the upper stage the OS is captured by the Orbiter and is placed in the EEV. A type II trajectory returns the Orbiter and EEV for direct entry into the Earth's atmosphere. After EEV release the Orbiter executes an Earth avoidance maneuver.

Mission timeline:

Earth launch	11/2011
Mars arrival	09/2012
Aero-brake end	11/2012
Landed mission end	02/2013
Capture phase end	03/2013
Return phasing end	09/2013
Return trajectory start	09/2013
Earth arrival	08/2014

An Earth launch C_3 of $10.1 km^2/sec^2$ will provide a 20 deg. Earth elevation angle for up to one hour after Mars landing allowing direct communication with the Earth. It should be noted that a direct Earth link for EDL is not required for MSR as this link is provided by the Mars communications relay network but this trajectory provides for one at the cost of this higher C_3. Future studies consider lower C_3 options. At Mars a 3.0 km/s V-infinity results in a Lander entry velocity (at 3522.2 km radius) of 5771 m/s and an orbiter MOI ΔV of 1106 m/s including an estimated 3% gravity loss. The backup launch opportunity in 2013 can be achieved with this system if direct Earth link is not required during EDL.

Carrier Stage — The Lander with attached Carrier Stage is targeted for direct entry at Mars. The Carrier Stage provides 50 m/s ΔV (including margin) for trajectory correction maneuvers (TCMs). The Carrier Stage will be separated from the Lander just prior to atmospheric entry.

MAV — The MAV is a two-stage hybrid launch vehicle with a solid rocket motor (SRM) first stage and a liquid rocket second stage. The mass of the first stage at ignition is 265 kg, including 138.3 kg of propellant to push 126.7 kg, which includes the first stage hardware, 3.8 kg fairing and the 84.1 kg second stage. The second stage includes 49.1 kg of propellant and 35 kg of dry mass including second stage hardware, 4.6 kg of payload and 1 kg of propellant for control.

MAV orbit parameters to LMO are: Assumes a Hohmann launch ellipse, 0 km altitude periapsis and 500 km altitude apoapsis gives a = 3647 km and e = 0.068549 (i = 15°, □ = 0°, □ = 180°, M = 0°). MSR LMO: Low-Mars Orbit, 500 km altitude circular orbit (i = 15°, □ = 0°, □ = 0°, M = 0°).

Orbiter — At the end of the launch period the arrival V-infinity has a declination at Mars of –28 deg. In order to get into a 15 deg inclination orbit a 200 m/s burn is needed at apoapse of the 20-hour orbit, which occurs at latitude of just over 15 deg. This approach has not been optimized and it may be more cost effective to design the MAV to achieve a 45 deg orbit rather than the 15 deg. orbit to allow flexibility at the cost of propellant mass on the MAV.

After MOI the Orbiter begins an orbit phasing/aero-braking effort to create an orbit where natural precession or drift will align the Orbiter with the OS orbit plane with minimal ΔV.

The Orbiter pre-rendezvous phasing orbit parameters: 250 km altitude periapsis and 1400 km altitude apoapsis gives a = 4222 km and e = 0.136191 (i = 15°, □ = 0°, □ = 180°, M = 0°).

The Orbiter (250x1400km orbit) and the MAV ascent orbit are nearly coplanar, and the Orbiter is just ahead of local vertical at MAV launch maximizing link time. The torroidal MAV antenna pattern allows coverage by the MSR Orbiter for about 5 minutes prior to launch and 25 minutes after. The mean anomaly of the ASI telecom asset is also adjusted to put it in view starting 5 minutes prior to launch and lasting 10 minutes after.

The return trajectory will leave Mars on 2013-09-23 and arrive at Earth on 2014-08-23. This Type II transfer has a departure C3 = 5.83 km2/s2 with a Mars departure ΔV of 1959 m/s. This will be accomplished in two parts to minimize gravity losses and to allow for departure geometry phasing. A gravity loss of 3% is assumed. The arrival V-infinity at Earth is 4.656 km/s, which results in an entry velocity of 12.01 km/s at 125 km altitude. The Orbiter ΔV budget is shown in Table 1:

Table 1: Rendezvous & Return Orbiter ΔV

Mission event	**ΔV (m/s)**
TCMs	30
MOI	1106
MOI gravity loss	33
Inclination change to 15 deg	200
Aero-braking to 1400 km x 250 km	125
Rendezvous	300
Departure	1959
Departure gravity loss	60
TCMs	30
Deflection maneuver	30
Margin	60
Total before departure	1794
Total from departure through arrival	2139

4. SPACECRAFT SYSTEMS

The MSR mission is composed of several flight system elements. The Lander includes the Lander bus, MER class rover, MAV, drill, sample arm, sample transfer system, and science instruments. The Lander is supported by the Carrier Stage during the Mars cruise phase. The lander bus and Entry System are scaled-up versions of the Mars Smart Lander design. The Rendezvous and Return Orbiter supports the Entry System and Lander mass during launch. The Carrier Stage will be attached to the base of the Entry System (Figure 1). After launch the Entry System containing the Lander with attached Carrier Stage will be separated from the top of the Orbiter. The Orbiter will then be separated from the launch vehicle adapter. Ultra-flex solar arrays with single axis articulation will then be deployed. The Orbiter configuration has passive dynamic stability during aero-braking by providing a center-of-pressure / center-of-mass offset. The 2-axis articulated HGA) will be deployed after aero-braking. Following OS capture and transfer to the EEV, the OSCAR and other elements not required for Earth return (112.8 kg drop mass) will be separated. A final separation event occurs when the EEV is targeted and released for Earth entry.

Planetary protection posses a significant design driver for the mission. This is a category V (C) mission according to the "NPG 8020.12B Planetary Protection Provisions for Robotic Extraterrestrial Missions". Category V (C) missions requirements include: stringent measures to prohibit unplanned impact with the Earth, the need for sterilization of returned hardware which directly contacted the target planet, and the need for containment of any unsterilized sample collected there and returned to Earth. These requirements will affect all phases of the mission, namely the outbound leg; sample acquisition, transfer and storage; sealing of the sample container; monitoring of the sample; return phase of the mission; Earth entry phase; and sample receiving laboratory.

An important factor when assessing concept level technical feasibility of multi-stage mission architectures, such as MSR, is expected dry mass and power growth. This dry mass and power growth occurs during the development phase and must be considered in all aspects of the design. The structure, propulsion and power systems must be sized to accommodate this expected growth. The Team-X approach is to apply a 30% contingency (expected growth) above the engineers current-best-estimate (CBE). This is first applied to the CBE power and then an additional 30% mass growth is applied to the CBE mass. Each element of the MSR architecture has been assigned a minimum of 30% mass and power growth. The resulting mass is then added as a payload allocation for the next stage to carry: e.g.: MAV Stage-2 has a 30% mass contingency applied. This Stage-2 with mass plus contingency is used to size MAV Stage-1 where 30% mass growth is then assigned to the CBE Stage-1 elements.

Flight Systems — The flight system is composed of the following elements shown in Figure 1:

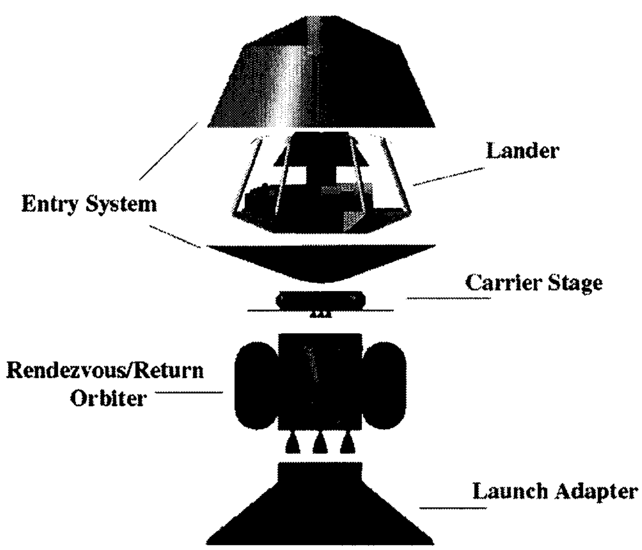


Figure 1: Launch stack configuration. Rendezvous and Return Orbiter supports the Lander; the Carrier Stage is attached to the Lander but is outside the load path between the Orbiter and the Lander.

Orbiting Sample Canister (OS) — The OS provides compartmentalized containment for a minimum of 500 grams of Martian material (Table 2). For this point design the OS is completely passive. It is a 16 cm sphere with no active attitude control or thermal requirements (Figure 2). It remains attached to the MAV upper stage for radio beacon support during the rendezvous. A command from the Orbiter initiates separation from the upper stage to allow the OS to be captured by the Orbiter. A timer on the upper stage will separate the OS after approximately 30 days if a command is not received. A low aldedo surface allows the OS to be tracked optically by the orbiter. A key design driver for the OS is assured containment and confirmation that the seals are intact prior to Earth entry.

Table 2: Orbiting Sample Canister Mass List

Orbiting Sample Canister (OS)	Mass (kg)
Sample	.5
Payload Total	*.5*
Structure	3.1
Thermal	.02
OS Total	*3.1*
OS Total (CBE)	**3.6**
Contingency (30%)	1.0
OS Total Mass	**4.6**

Mars Ascent Vehicle (MAV)

The MAV must provide a ΔV of 2000 m/s for the first stage and 2468 m/s for the second stage, including gravity losses, to place the OS in a 500 km circular orbit. This point design is a two-stage hybrid launch vehicle with a solid rocket motor first stage and a liquid bi-propellant second stage. Control is provided for both stages by the second stage mono-prop head-end steering system (Figure 2).

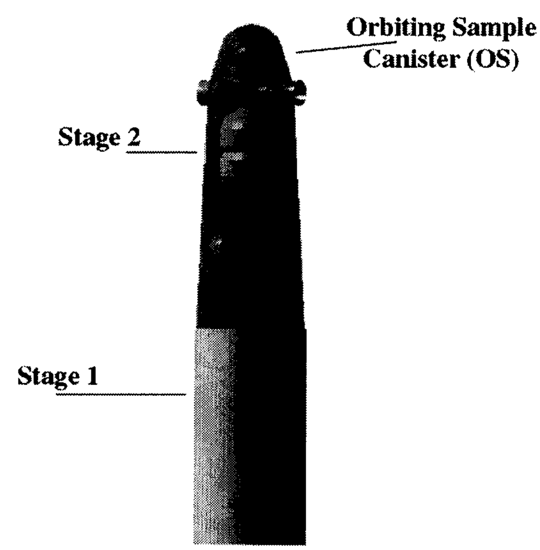


Figure 2: Mars Ascent Vehicle (MAV)

The mass of the first stage at ignition is 265 kg, including 138.3 kg propellant to push 126.7 kg, which includes the first stage hardware, 3.8 kg fairing and the 84.1 kg second stage. The second stage includes 49.1 kg of propellant and

35 kg of dry mass including second stage hardware, 4.6 kg payload and 1 kg of propellant for the pitch and yaw thrusters.

Stage 1 — This stage consists of an enlarged Star17 solid rocket motor, connecting structure, MAV fairing and fairing separation hardware. It has a specific impulse of 287 s, and a mass fraction of 0.89. Thrust is approximately 5000 N. In addition to the requalification required for the augmented size the Star17, it will also be qualified to operate at a lower grain temperature. Motor ignition will be provided by an interface from the Lander. The MAV will be kept warm in a thermal enclosure provided on the Lander. The MAV launch adapter will erect the MAV to a vertical position relative to the Lander payload pallet just prior to launch.

Table 3: Mars Ascent Vehicle Mass List

Mars Ascent Vehicle (MAV)	Mass (kg)
OS	4.6
Payload Total	*4.6*
MAV Stage 2	
Attitude Control	1.0
Command & Data	1.0
Power	1.4
Propulsion	9.0
Structure	7.5
S/C Adapter	.7
Cabling	1.5
Telecomm	.7
Thermal	.3
MAV Stage 2 Dry Total (CBE)	*23.2*
MAV Stage 2 + OS Total	**27.7**
Contingency (30%)	7.0
MAV Stage 2 w/Contingency	*34.7*
Stage 2 Propellant & Pressurant	45.9
MAV Stage 2 Total (Wet)	**80.6**
MAV Stage 1	
Propulsion 1	16.9
Structure	13.1
MAV Adapter	2.8
MAV Stage 1 Dry Total (CBE)	*32.8*
MAV Stage 1+ Stage 2 Total	**113.4**
System Contingency (30%)	9.8
MAV w/Contingency	*123.2*
Stage 1 Propellant & Pressurant	138.5
MAV Launch Total (Wet)	**261.8**

Stage 2 — The MAV upper stage contains the OS, avionics, communications system, solar power system and connecting structure. It is a chemical bipropellant (bi-prop), 3-axis controlled stage. The telecommunications subsystem will provide health and status tones during ascent and will provide a beacon on orbit to allow the orbiter to determine the MAV-OS position. It will have solar cells to power the telecom beacon while in the sun and will operate in this mode for 30 days. If a command has not been received the OS will be released.

Stage 2 has one main engine with approximately 200 N thrust and a specific impulse of 320s. Located at the nose of the MAV are four 22 N bipropellant attitude control thrusters configured with each pair forming a "V" for reaction control. The proposed propellant combination is MON25-MMH. This propellant combination enables the bipropellant propulsion system to operate with cold propellants minimizing thermal heating requirements. Advanced lightweight tanks, pressurant system components, and propellant control components are assumed.

ACS — Key Attitude Control System (ACS) requirements for the MAV include:

Pitch and yaw pointing control for the thrust vector to within ±1 degree, 3 sigma. This will limit the peak cosine loss to 0.03% (RSS of pitch and yaw).

Pointing knowledge of the thrust vector to within 0.2 degrees, 3 sigma. The requirement supports achieving the desired orbit to within a reasonable tolerance. It corresponds to pitch and yaw pointing knowledge to within ±0.14 degrees, 3 sigma.

An underlying assumption is that pointing control errors not related to knowledge will average out so that knowledge errors are the primary contributor to heading errors that will affect the final orbit.

Pointing stability to within ±0.1 deg/sec, 3 sigma. This is a loose requirement for the average rate. Peak jitter may be higher during the SRM burn. The requirement corresponds to crossing a ±1-degree deadband in 20 seconds.

Autonomous on-board guidance for 6-degree-of-freedom (DOF) trajectory following is enabled by inertial navigation using precision accelerometers and gyros. Stage 2 includes micro-accelerometers; JPL MESO gyros and a computer to run ACS software, along with propulsion control and propulsion drive electronics.

Micro-accelerometers available today can provide 3-sigma per-axis bias accuracy to within tens of μg's at low bandwidths (e.g., up to 1 Hz). Accuracy goes down proportional to the square root of the bandwidth. An uncompensated bias of 30 μg's acting over 10 minutes will correspond to a velocity error of about 0.2 m/sec and a position error of about 53 meters. Compare the velocity error with the total delta-V of 4500 m/sec. The 0.2 m/sec error is less than 0.005% of the total delta-V.

The baseline includes redundant sets of JPL micro-gyros. These gyros are expected to achieve 3-sigma per-axis bias stability to within ±0.1 deg/hour. Attitude errors due to bias acting over 10 minutes would be within ±0.017 degrees, 3 sigma. RSS of errors in 2 axes will be less than 0.024 degrees, 3 sigma.

The Lander using knowledge estimated from Lander sensor measurements will initialize MAV attitude. It is assumed that Lander position will be well known and can be used to initialize the MAV state vector prior to launch.

Control Authority relative to Stage 1 SRM disturbance torque is an issue for this MAV configuration. A preliminary analysis was done to determine the likely disturbance torque due to the SRM. The key assumptions for the analysis were: A 1.75 meter offset between the exit cone and the rocket center of mass at the end of the burn. A 0.75-meter offset between the plane of the control thrusters and the center of mass. A 1-mm misalignment between the nominal thrust direction and the center of mass. A worst case 0.1 degree 3-sigma thrust vector misalignment due to internal burn characteristics and a 5000-N peak thrust from the SRM. Based on these assumptions, the worst-case disturbance torque would be about 20 Nm. Assume a set of four 22-N thrusters configured as two V's. This would provide a maximum thrust of 31 N in pitch and yaw. The maximum control torque would be 23 Nm (0.75-m moment arm). The baseline design carries only about 15% margin for control authority. This implies a moderate to high risk, since the assumed 0.1-degree thrust vector misalignment is better than typically assumed by a factor of 5. The underlying assumption for this study is that SRM manufacturing processes will improve by 2007 such that a 0.1-degree thrust vector misalignment is relatively easy to achieve. One fallback would be to lower the thrust of the SRM. Since it will be a custom-designed version of the Star 17, there is no apparent reason that it couldn't be made to provide 4000-N peak thrust, for instance. The baseline design for the liquid upper stage carries a factor of 10 margin for control authority and is not an issue.

Telecom — During MAV launch the UHF system on the second stage will provide beacon tones to the Orbiter for health, status and reconstruction. The Orbiter will extract Doppler measurements when the MAV is in beacon mode to determine rough positioning of the MAV/OS. After the Stage 2 burn is complete, the vehicle maximum axis of inertia will be spun up about a preferred direction at 2 to 3 RPM. From this point on, the MAV/OS will remain in this spin orientation. The second stage will include a UHF toroidal antenna that has its beam pattern wrapped around the long axis. Consequently, when the vehicle is spinning about the axis of maximum inertia (transverse axis of the rocket), beam coverage will be centered about the spin line. The toroidal antenna will provide coverage tens of degrees off the spin line. The telecom system is designed with 5 dB link margin at maximum range between MAV and ASI/MSR of approx. 6500 km/1500km respectively.

Stage 2 will have redundant UHF transceivers (200 mW transmit power) with a transmit/receive switch and a single UHF ring antenna. The transceiver will operate in a half-duplex mode. The UHF peak gain is ≤2.5dBi. There will be a ring of RF transparent material on the fairing to allow the antenna to radiate during MAV launch. The UHF system has the following specifications:

Transceiver: MCAS2 type
Internal USO (must be integrated into the transceiver hardware)
Power consumption: Transmit/Receive = 5W, DC
UHF transmit power = 200 mW
UHF peak gain is ≤2.5dBi
Telecom Subsystem Mass Total = 0.7kg

CDS — For the Mars ascent vehicle (Stage 2), the CDS design is illustrated below. The design is dual-string with an 80C51-based Controller Module, which was also used on DS-2.

The 80C51 microprocssesor has modest data processing capability (8-bit). It includes 128K SRAM, 128K EEPROM, and 32 D/A and A/D converters. The amount of memory required should be small, 10 Mbits from the accelerometer and Inertial Measurement Unit (IMU) and A/D processing over an 8 hour period. It has low power consumption and mass. It is capable of surviving high shock environments such as the MAV launch. It includes interfaces to the ACS sensors (accelerometers and IMU). A second chip is used to interface with the telecommunications system. A timer is required with the controller module. The mass of the command and data system (CDS) for the stage 2 MAV is 1.0 kg (dual-string) and power consumption is 2.3 watts.

Power — The MAV power subsystem provides 24 W during the 100 second Stage 1 burn, 17 W during the 150 minute coast period and 52 W during the 150 second Stage 2 burn. The solar arrays provide power to the communications system during the 30-day rendezvous and retrial orbit while in sun. The solar array can provide power in any orientation and is a combination of body mounted and rigid deployable arrays. Triple junction solar cells are 29% efficient yielding a 90 W/Kg power density with an overall array mass of 0.8Kg. A thermal battery supports high discharge rates during the MAV ascent for a total of 2.7 W-Hours. The 0.166Kg battery mass estimated by specific power and energy is doubled to account for small size scaling effects. The battery uses a multi-tap design to provide: 5 Vdc, 15 Vdc, and 28 Vdc as required. Switching is provided for pyro functions. Power electronics mass is estimated to be 0.28Kg.

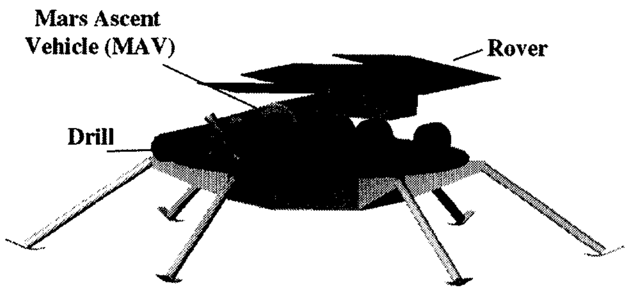
Structure — Stage 1 is a stretched Star-17 with modified mounting flanges. The Star-17 structure will carry MAV loads and attach to the adapter, which stays on the Lander platform. The Stage 2 interface structure stays attached to Stage 1 and must reach past the 200N bi-prop engine on Stage 2. The fairing is separated from Stage 1 but encloses Stage 2: size ~0.5 m diameter x 0.8m long. A Carbon-Carbon shell with 0.015in thickness tolerates the high heat loads during ascent. The mass of the fairing system is 3.8kg total including joints and flanges (1.5kg) and split/separation ordnance (2.5kg).

The Stage 2 bi-prop system is composed of 30 cm diameter load carrying fuel and oxidizer tanks, small pressurant tank, 200N bi-prop engine, four 20N bi-prop reaction control thrusters, avionics and power subsystem. Solar array area is $0.2m^2$, but 2/3 of that area will be deployed on two flip-out panels. The solar arrays will be deployed using simple spring hinges with pin-pullers actuated from fairing jettisoning.

Lander

The MSR Lander is derived from the Smart Lander precursor mission with increased propellant capacity and thrust to accommodate a larger landed mass. The Lander payload consists of the MAV, the MAV launch support system, the Sample Transfer Chain, MER class rover (Figure 3) and 50kg allocation of science instruments (including drill). Lander system mass is detailed in Table 4.

ACS — The Lander will provide attitude control after the EDL system separates from the Carrier until touchdown on the surface of Mars. The EDL sequence includes aero-maneuvering during entry, supersonic and subsonic parachutes, and terminal powered descent with hazard avoidance. During actively controlled phases, Lander attitude will be 3-axis stabilized using thrusters. During powered descent the Lander will be 6-DOF stabilized using six 2670-N throttled main engines. Pitch and yaw control is achieved by varying the thrust of the individual engines. Eight 89-N hydrazine thrusters are used for roll control and for horizontal translation during powered descent and for 3-axis control before entry and during aero-maneuvering. Cutouts in the back shell are provided for the thrusters during this phase.



(UltraFlex solar arrays not deployed)

Figure 3: MSR Lander

Table 4: Lander Mass List

Lander	Mass (kg)
Payload	
Instruments	38.5
Drill	15.4
Rover	159.6
Sample Arm	7.7
Sample Transfer Chain	26.9
MAV (all stages + OS)	261.8
Payload Total	*509.9*
Lander Bus	
Attitude Control	72.6
Command & Data	19.9
Power	111.6
Propulsion Descent stage	84.9
Structure	348.3
S/C Adapter	13.2
Cabling	50.1
Telecomm	24.2
Thermal	38.3
Bus Total (CBE)	*763.1*
Lander Total Dry (CBE)	**1273.0**
System Contingency (30%)	303.4
Lander w/Contingency	*1576.4*
Lander Propellant & Pressurant	213.9
Lander Total (Wet)	**1790.3**

ACS Requirements for the Lander are as follows:

Autonomous on-board EDL guidance, navigation, and control will include the following functional capabilities:

- Attitude re-orientation capability for aero-maneuvering during entry
- Thrust vector control algorithms for maneuvering during descent and landing
- Descent guidance laser radar technology, with auto-focus and auto-stabilization over a >1000:1 dynamic range in descent from 1.5-km altitude down to the surface
- Terrain-relative navigation capability using scanning laser generated elevation maps
- Hazard identification through on-board interpretation of elevation maps
- Target-relative closed-loop terminal-path guidance laws
- Soft-landing approach lateral and/or hovering maneuver capability for hazard avoidance

During aero-maneuvering:

- Pointing control to within ±1 degree, 3 sigma, per axis.
- Pointing knowledge to within ±0.5 degrees, 3 sigma, per axis.
- Pointing stability to within ±0.1 deg/sec, 3 sigma, per axis. This is a loose requirement for the average rate. It corresponds to 20 seconds to cross a ±1 degree deadband.
- Slew about the velocity vector through 180 degrees in less than 10 seconds. This would support an aero-maneuvering strategy that calls for rapid roll reversals.

During powered descent:

- Pointing control for the thrust vector to within 1 degree, 3 sigma. This corresponds to pitch and yaw control to within ±0.7071 degrees, 3 sigma.
- Pointing knowledge of the thrust vector to within 0.5 degrees, 3 sigma. This corresponds to pitch and yaw knowledge to within ±0.35 degrees, 3 sigma.
- Pointing stability to within ±0.1 deg/sec, 3 sigma, per axis. This is a loose requirement for the average rate. It corresponds to 14 seconds to cross a ±0.7 degree deadband.

Autonomous 6-DOF trajectory path guidance and control is required to enable a safe landing that meets the following requirements:

- Terminal velocity relative to the local vertical < 3 m/sec, 3 sigma.
- Terminal velocity relative to the local horizontal < 1 m/sec, 3 sigma.
- Touchdown within several kilometers, 3 sigma, radial error relative to the target site.

There is a requirement for pointing knowledge to support MAV attitude initialization.

- Per-axis knowledge of the MAV support platform needs to be within ±0.1 degrees, 3 sigma. This supports a total allocation of 0.2 degrees for knowledge of the MAV thrust direction.

The Lander includes the following ACS sensors to meet the stated requirements:

An inertial measurement unit (IMU) containing gyros and accelerometers. Accelerometers are used to help determine when to deploy the supersonic parachute. The IMU is used for attitude and position determination after separation from the Carrier all the way down to the surface.

The Scanning LIDAR, also referred to as the scanning laser mapper (LAMP), generates a 3-D map of the local terrain. This terrain map is used for hazard avoidance during terminal descent. Scanning LIDAR generates local elevation maps of the surface starting at 1 to 1.5 km altitude and also provides altitude measurements starting at 2 km above the surface.

The phased array radar provides altitude measurements from 4 km above the surface to determine when to release the subsonic parachute and start powered descent. It may also provide information about local terrain as a functional backup to the LIDAR. Surface-relative altitude and velocity are measured using radar starting at about 3.7 km.

On-board navigation system and hazard identification logic designates a safe, reachable local target-landing site. Terrain-relative navigation is based on local elevation maps. The on-board guidance system computes time to separate from subsonic parachute and begin powered descent. It also establishes a trajectory that will reach the designated target while maximizing performance margin. Radar and LIDAR, along with hazard detection and re-targeting logic, continue to operate during powered descent.

The IMU on the Lander will be used to determine Lander attitude relative to Mars. Accelerometers are used to determine the local gravity vector in the Landed Payload frame. Gyros are used to determine the Mars rotation vector in the Landed Payload frame. With well-known Lander position, the directions of the gravity and rotation vectors can be used to determine the orientation of the Landed Payload frame relative to a Mars's local frame. The orientation of the Landed Payload in the Mars local frame will be used to initialize MAV attitude prior to launch.

The achievable knowledge of orientation based on IMU accuracy is based on the following factors. Mars's gravity is approximately 0.4 g's. The accelerometers in the LN-100S will have a per-axis resolution to within ±30 micro-g's, 3 sigma. Bias should be within ±0.25 milli-g, 3 sigma, after warm up just prior to MAV launch. The per-axis angle errors due to accelerometer bias will be within ±0.036 degrees, 3 sigma. The Mars rotation rate is roughly 15 degrees/hour. The targeted landing site will be relatively close to the Mars equator, so a large component of the rotation rate will be perpendicular to the local gravity vector. The rotation rate can therefore be used to accurately determine the clock angle of the Lander frame about the local gravity vector. The gyros in the IMU will have a per-axis bias stability to within ±0.003 deg/hour, 3 sigma. The clock angle error due to gyro bias should be within ±0.015 degrees, 3 sigma. It appears feasible to achieve per-axis knowledge of the IMU orientation relative to a Mars's local frame to within ±0.05 degrees, 3 sigma. This should be adequate for achieving per-axis knowledge of the MAV support structure orientation to within ±0.1 degrees, 3 sigma. However, there will be misalignments that need to be considered.

Telecom — The telecom system needs to support 1.0 kbps DTE rate during surface operations for health check/status of the Lander. The UHF relay needs to provide 1 Gbit/day to orbiting assets such as the ASI. The Lander telecom system will support transmission during cruise and EDL by the Carrier Stage and the Entry System antennas respectively. The X-band DTE telecom subsystem consists of a 40W RF transmitter with an arrayed-element HGA. The LGA is used for EDL and emergency surface operations. Normal DTE communication from the surface is accomplished using 0.3 m HGA. At 2.35 AU a 1.0 kbps return link and 1.0 kbps forward link must be provided using the 34 m DSN subnet. Emergency DTE communication from the surface is accomplished using the LGA. At 2.35 AU an 8.0 bps return link and 31.25 bps forward link must be provided using the 70-m DSN subnet.

The UHF telecom subsystem uses the Electra radio with an LGA to communicate with the ASI Orbiter. A 235 kbps average return including both day and night passes must be provided. This is achievable with a 3 dB margin but additional margin may be desirable to offset antenna degradation on the rover and topographical effects. If the average ASI coverage per Sol is 4 hours, then a 120 kbps rate would return the needed 1.0 Gbit of volume with 6 dB margin. (CNES may also be available in this timeframe; it would provide about 20% of the data return capability of ASI)

Hardware specifications:

- HGA: X-band, peak gain = 24dBi with 2-axis articulation
- LGA: X-band, peak gain = ~5dBi
- X-band transponder: SDST (x2), Transmit/Receive = 14W, DC
- X-band TWTA (x2). 40W RF, 67W DC
- Additional hardware includes diplexers (x2), isolators (x2), -3dB hybrid coupler, polarizer, rotary joints (x2), WGTS, CXS (x2), loads, coax, waveguide
- UHF LGA: peak gain = ~6dBi
- Transceiver: Electra payload
- Internal receiver redundancy
- Internal USO
- UHF receive only = 13W, DC
- UHF transmit/receive = 68W, DC

CDS — The Smart Lander CDS was also used for the MSR Lander to maintain hardware and software heritage. System requirements for the Lander are the same as stated previously for the Orbiter. The flash memory used will meet the 10 krad radiation requirement. The storage requirement is 2 Gbits (one NVM card). This is derived from the EDL data received from the radar, IMU, LIDAR, etc. Given the small storage requirement, it is assumed that the science data is not compressed (although a 2:1 lossless compression is feasible to provide extra margin).

The equipment list for the Lander includes a hybrid of cPCI and VME cards. This design includes an X-2000 (cPCI) and RAD-750 system flight computer. Non-Volatile Memory is 2 Gbits.

The VME 6U is composed of the following elements:

- I2C Bridge Assembly is used to provide an interface for power switching.
- High Speed serial I/O assembly is used to interface with instruments and ACS.
- Uplink/Downlink Telecom interface
- Tzero and Miscellaneous I/F (for interfacing to launch vehicle, etc.)

- 1553 Remote Engineering Unit will provide interfacing between lander and carrier
- Power Conditioning Assembly (PCA)

The CDS is dual-string with each string providing a PCI/1553 backplane. The power required for this system is 55.8 watts average and 17.18 watts standby. However, the Lander requires reduced power during night sleep mode operations. Running the flight computer and other elements at a slower clock speed reduces power. Other elements will use a timer to wake-up for planned events such as a telecom pass. The remote engineering unit is powered off after landing, as the ACS elements are no longer required.

Propulsion — The propulsion system provides 250 m/s to an initial drop mass of 1850 kg. The 250 m/s accounts for maneuvering propellant consumption plus the final descent velocity change propellant including 1.0 kg of propellant for roll control. All pitch and yaw attitude control is accounted for in the 250 m/s value.

The lander propulsion system is patterned after the Viking system. It has six throttleable hydrazine thrusters, eight 89 N thrusters that fire through holes in the back-shell, five hydrazine tanks, and one pressurant tank. The 89 N thrusters are used for yaw and roll attitude control before the heat shield is jettisoned and for roll control only after it is jettisoned as the main engines provide pitch and yaw control. A technology program will be necessary to develop and qualify the throttleable lander engines.

Thermal — The Lander thermal protection system will use flight proven elements. The lightweight CO_2 insulation assumed for the MAV hanger and warm electronics box is expected to be available for this mission but other options are available with a small mass impact. A range of thermal surfaces, thermal conduction and isolation elements, temperature sensors, and electric heaters/thermostats are incorporated in the thermal design. The baseline plan for this study is to use RHU's (up to 15) to reduce electric power demands. A comprehensive study of solar options will be investigated before a decision is made to use RHUs.

Power — Lander power requirements for the descent phase were not reviewed in detail a 13.6 Kg Li-SOCL2 battery provides power during the descent. A traditional electrical pyro initiation system is used but Laser or explosive systems could reduce system mass.

Lander power requirements are met using an Ultra-flex Solar Array and Li-ion battery for the 90-day landed mission. A fixed Ultra-flex Array is deployed in the horizontal position. An optical depth of 0.5 was assumed with a worst case Ls = 225 at end of mission. A conversion efficiency of 20% was assumed after all layout and performance loses were considered.

Lander solar array specification:

1. Single Wing Ultra-flex
2. 29% Efficient Cell Technology
3. 80% Packaging Factor
4. Total Area 7.67 m^2
5. Diameter is 3.12 meters
6. Total Power 523 Watts
7. EOL at 1.67 AU and 0 degrees
8. Specific Power 28.4 Watt/Kg
9. Total Mass is 18.39 Kg

The battery sizing case is Mars' night requiring a total of 1770 W-hours with 5% distribution losses. One battery failure will result in a DOD of 60%. With no failures DOD is 40%. The Lander secondary battery specification:

- 3 batteries
- 8 Cell string length
- 54 Ampere-Hour Design (new design)
- 18.3 Kg mass per battery
- 4665 Total System W-hours
- Total mass is 46.75 Kg

Lander power electronics mass estimate for the Lander is 19.3 Kg power avionics estimate

Structure — The MSR Lander will be a scaled version of the Smart Lander for the main structural elements. The Lander pallet is a new design including interface hardware for rover, MAV and instruments. A power subsystem including deployable, fixed Ultra-flex solar arrays has also been added to the MSR Lander.

Entry System

The Entry System is a scaled-up version of the Mars Smart Lander consisting of aero-shell, parachutes, and hardware to support and protect the Lander and payload elements during the EDL phase (Figure 4). The heat shield and back-shell system size was increased to accommodate a larger payload mass. An uncertainty factor of 50% has been applied for the MSR heat shield mass to account for uncertainty and potential mass growth. The larger Delta IV fairing size allows an increased heat shield diameter of 4.5m. Given a total Entry System mass of 2,467.9 Kg (Table 5) a ballistic coefficient of 100.4 kg/m^2 is achieved. During cruise a pumped loop heat pipe system will be used to reject thermal energy and will be similar in design to that used on the Mars Pathfinder mission.

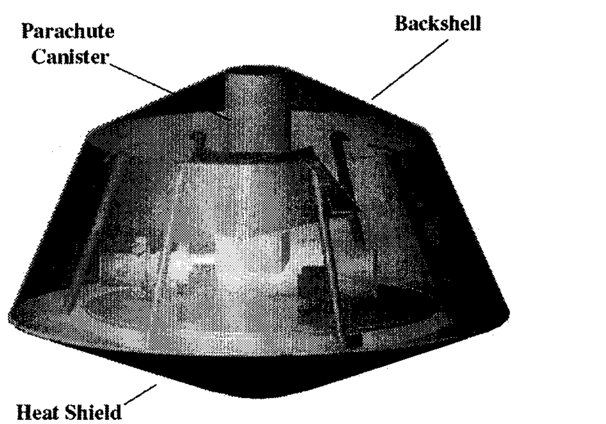


Figure 4: MSR Entry System

ACS — The EDL sequence includes aero-maneuvering during entry, supersonic and subsonic parachutes, and terminal powered descent with hazard avoidance. All active ACS elements are located on the Lander.

ACS Requirements for the EDL System include:

Pointing control to within ±1.0 degree, 3 sigma, per axis. This applies prior to entry and during aero-maneuvering.

Pointing knowledge to within ±0.5 degrees, 3 sigma, per axis. This applies prior to entry and during aero-maneuvering.

Pointing stability to within ±0.1 deg/sec, 3 sigma, per axis. This is a placeholder requirement for the average drift rate. It will allow 20 seconds to cross a ±1.0 degree thruster deadband.

Slew about the velocity vector through 180 degrees in less than 10 seconds. This will support an aero-maneuvering strategy that calls for rapid roll reversals.

Algorithms and software for aero-maneuvering will be inherited from the Smart Lander program. These represent new technology currently at TRL 4 or 5 and not yet flight proven.

Table 5: Entry System Mass List

Entry System	Mass (kg)
Payload	
Parachutes (2)	82.1
Lander w/Payload (Total Wet Mass)	1790.3
Payload Total	*1872.4*
Entry System Elements	
Power	6.1
Structure	208.3
Cabling	5.0
Telecomm	2.8
Thermal	197.1
Entry System Element Total (CBE)	*419.3*
Entry System Mass	**2291.7**
System Contingency (35%)	176.2
Entry System w/Contingency	*2467.9*
Entry System Mass Total	**2467.9**
Entry System Dia. 4.5m	
Drag Coefficient 1.57	
Ballistic Coefficient **100.4 kg/m^2**	
Entry System Element Total	**677.9**

Telecom — Telecom requirements for the Entry System are semaphore tones during EDL for anomaly resolution purposes. The active telecom hardware utilized by the Entry System and Carrier Stage is carried on the Lander. The Entry System and Carrier Stage use passive components only. The UHF antenna does have a coax cable leading up from the transmitter/receiver. Depending on the location of the antenna, a cable cutter may be required. If the antenna is attached to the Lander, a cable cutter is not needed. The UHF LGA peak gain is ~6dBi.

Structures — The Smart Lander entry system is sized for 1610 kg landed mass and the MSR is sized for 1771 kg. The Smart Lander aero-shell is approximately 4.05 m. The maximum aero-shell size that will fit into the Delta IV fairing is 4.5 m. This diameter will allow MSR to maintain an acceptable ballistic coefficient for entry. A scale factor for the structure mass of the Entry System was based on area, 20.25 m^2 /16.4 m^2 = 1.235, so the structures-related mass was scaled up by 1.235.

Carrier Stage

The Carrier Stage provides electrical and propulsive resources for the Lander during Earth to Mars cruise (Figure 5). Unlike the Mars Smart Lander, the MSR Carrier Stage will not be capable of independent flight. The MSR Carrier Stage is similar to MER style architecture where all the avionics are on the lander. This leaves primarily solar arrays and propulsion elements on the Carrier Stage saving mass and cost. The drawback is that the Carrier Stage cannot provide EDL relay for the Lander. However, Mars Program communication relay assets will be in place to provide the necessary relay. The Carrier Stage shall provide 30 m/s to an initial launch mass of 2737.9 kg and drop the 2467.9 kg Entry System such that it enters Mars' atmosphere (Table 6). The carrier will not perform a deflection maneuver for planetary protection and will require appropriate sanitization.

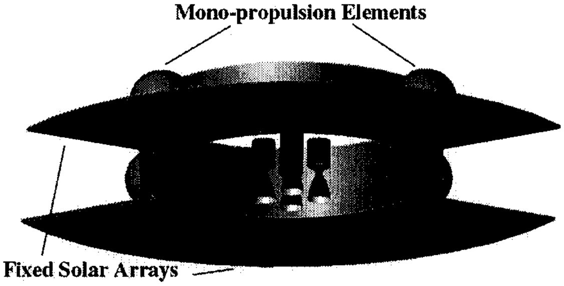


Figure 5: MSR Carrier Stage

ACS — Key ACS requirements include:

Pointing control for the boresight of the optical navigation camera to within 0.15 degrees, 3 sigma. This is about one-tenth of the camera FOV and corresponds to pitch and yaw pointing control to within ±0.106 degrees, 3 sigma.

Pointing knowledge of the opt-nav camera boresight to within 0.075 degrees, 3 sigma. This is half of the pointing control requirement and corresponds to pitch and yaw pointing knowledge to within ±0.053 degrees, 3 sigma. Pointing stability for the opt-nav camera to within ±6 arcsec/sec, 3 sigma, per axis. This is for the average drift rate, which will allow 2 minutes to cross a ±0.1 degree deadband.

Pointing control for the HGA boresight to within 0.5 degrees, 3 sigma. This will limit the loss due to pointing errors to less than 0.2 dB for the 0.3-m X-band HGA.

The Carrier Spacecraft velocity vector on approach needs to be targeted to within 0.1 degree, 3 sigma, prior to separation of the Entry System. Thrust vector pointing control during delta-V maneuvers will need to be within 1.0 degree, 3 sigma, to support Carrier Stage targeting.

Attitude control approach uses eight 0.7-N hydrazine MIT thrusters for RCS control and small delta-V maneuvers. Stellar inertial attitude determination is accomplished using precision star tracker and gyros. Coarse wide FOV sun sensors and gyros are used during initial deployment and for safe mode.

The optical navigation camera on the Carrier Stage will be used for accurate determination of spacecraft relative position on approach to Mars. An alternate strategy would be to use orbiting assets at Mars for position determination based on Doppler. The optical navigation camera is single string. The Doppler approach provides functional redundancy.

This blowdown hydrazine propulsion system has eight 0.7 N minimum impulse thrusters (MIT) for Delta V maneuvers and attitude control.

Table 6: Carrier Stage Mass List

Carrier Stage	Mass (kg)
Payload	
EDL System	677.6
Lander with Payload	1790.3
Entry System Total Wet	*2467.9*
Carrier Stage Bus	
Attitude Control	22.9
Command & Data	1.8
Power	32.8
Propulsion	12.7
Structure	65.4
S/C Adapter	2.7
Cabling	15.6
Telecomm	3.3
Thermal	13.3
Carrier Stage Bus Total (CBE)	*170.6*
Carrier Stage + Entry System	2638.5
System Contingency (30%)	51.2
Carrier Spacecraft w/Contingency	***2689.7***
Propellant & Pressurant	48.2
Carrier Spacecraft Total (Wet)	**2737.9**
Rendezvous/Relay Orbiter (Wet)	4245.9
Launch Mass Total	**6983.8**
Launch Vehicle Capability	7392.9
Spacecraft Mass Margin	**409.1**

Telecom — Telecom requirements for the Carrier Stage are to provide direct to Earth (DTE) communications and support emergency communications during cruise. The Entry System and Carrier Stage use passive telecom components only. The Lander provides the active telecom hardware utilized by the Entry System and Carrier Stage. The X-band DTE telecom subsystem uses the 40 W RF transmitter from the Lander with an arrayed-element HGA mounted on the Carrier Stage. A single LGA provides initial acquisition and emergency coverage.

The Carrier Stage antennas are coupled to the transmitter/receiver via a circular wave-guide stacked antenna configuration. This has heritage from Mars Pathfinder and MER. The RF signals for either antenna are differentiated by opposite circular polarizations. As a result no active control (i.e. antenna switching, cable cutting) is required.

During early cruise the spacecraft will use the LGA (~8dBi) out to 0.3 AU at 750 bps down-link and 250 bps up-link with Doppler and ranging using the 34-m DSN subnet.

During latter cruise a 0.3m HGA (24dBi) will be used out to 2.35 AU. This will support a 1.0 kbps down-link and 125 bps up-link with Doppler and ranging to the 34 m DSN subnet. At 2.35 AU the LGA will provide 20 bps for emergency down-link and 62.5 bps up-link using the 70 m DSN subnet. This assumes an off-boresight on the LGA of no more than a 45°.

Power — Solar Arrays are configured with two-fixed panel deployable wings that use 29% triple junction solar cells. Avionics include control and distribution using array string switching and distribution switching. Propulsion valve driver boards and pyro driver boards are also mounted on the Carrier Stage. All other elements are carried on the Lander. Only 28 Vdc power is supplied from Lander to the Carrier. The solar array provides 522 Watts at EOL with 4.3 m.2 The mass is 8.0 Kg for both solar arrays without substrate.

Structure — The Carrier Stage is attached to the Entry System heat shield. It supports two star trackers, deployable solar arrays, an LGA and fixed HGA and propulsion system components. The mono-prop hydrazine system carries 50 kg of fuel in two tanks.

Rendezvous and Return Orbiter

The Rendezvous and Return Orbiter provides support during all flight phases from launch, Mars Orbit insertion, aero-braking, OS rendezvous and capture (The Orbiter will be the active element during the rendezvous phase), trans-Earth injection (TEI), EEV targeting and release, and the Earth avoidance maneuver. It performs chemical orbit insertion at Mars (MOI) to a highly elliptical orbit and aero-brakes down to a final orbit of 250 km×1400 km during the landed mission. It provides a telecom asset for the landed mission during this time. The orbiter carries the Orbital Sample Capture and Retrieval (OSCAR) device, shown in Figure 6. The mass allocated for this system is 86 kg. Once the sample capture is completed 112.8 Kg including most of the OSCAR and other elements will be jettisoned. These elements include the Electra package carried by telecom, the primary launch battery, associated thermal, and a portion of cabling. An Orbiter subsystem mass summary and drop mass summary are included in Table 7.

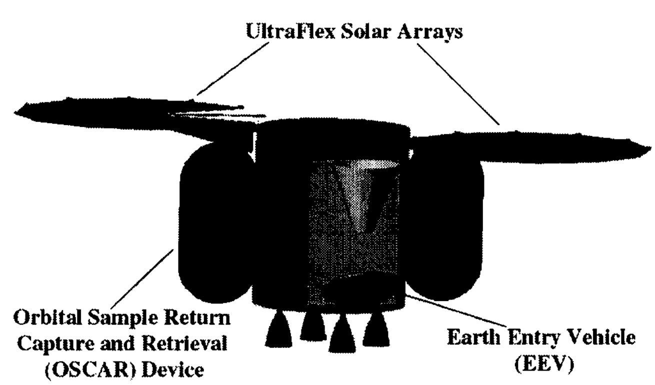


Figure 6: Rendezvous and Return Orbiter

Table 7: Rendezvous and Return Orbiter Mass List

Orbiter	Drop Mass	Mass (kg)
Payload		
EEV	-4.6	32.2
Sample Capture	61.0	61.0
Sample Capture Scar mass	0	25.0
Payload Total	*56.4*	*118.2*
Orbiter Bus		
Attitude Control	0	17.6
Command & Data	0	6.6
Power	8.4	89.6
Propulsion 1	0	130.3
Structure	0	396.8
S/C Adapter	0	47.6
Cabling	3.9	39.6
Telecomm	14.0	25.2
Thermal	4.0	35.5
Orbiter Bus Total (CBE)	*30.3*	*788.9*
Spacecraft Total (Dry)	**86.7**	**907.1**
System Contingency (30%)	26.0	272.1
Spacecraft w/Contingency	*112.8*	*1179.9*
Propellant & Pressurant	0	3066.8
Orbiter Total (Wet)	**112.8**	**4245.9**
Carrier+Entry System Mass		2737.9
L/V Adapter		0
Total Launch Mass		**6983.8**
Launch Capability		**7392.9**
Mass Margin		**409.1**

ACS — The Rendezvous and Return Orbiter is 3-axis stabilized using reaction control thrusters. Sixteen 0.7-N hydrazine MIT thrusters are provided for 6-DOF control (attitude and translation) during OS rendezvous and capture and for roll control during main engine firings. Top level ACS requirements for the Orbiter are as follows:

Pointing control for the boresight of the opt-nav camera to within 0.14 degrees, 3 sigma. This is about one-tenth of the camera FOV. It corresponds to pitch and yaw pointing control to within ±0.1 degrees, 3 sigma.

Pointing control of the HGA boresight to within 0.4 degrees, 3 sigma. This will limit the loss due to pointing to within 0.4 dB for the 1-m X-band antenna. This corresponds to pitch and yaw pointing control to within ±0.28 degrees, 3 sigma.

Per-axis pointing control of the Orbiter during rendezvous and OS capture to within ±0.28 degrees, 3 sigma. This is, comparable to the pointing control required for the HGA.

Pointing knowledge of the opt-nav camera boresight is 0.07 degrees, 3 sigma. This is half of the pointing control requirement and corresponds to pitch and yaw pointing knowledge to within ±0.05 degrees, 3 sigma.

Pointing knowledge of the HGA boresight to within 0.2 degrees, 3 sigma. This is half of the HGA pointing control requirement. It corresponds to pitch and yaw pointing knowledge to within ±0.14 degrees, 3 sigma.

Per-axis pointing knowledge of the Orbiter during rendezvous and OS capture to within ±0.14 degrees, 3 sigma.

Pointing stability for the opt-nav camera to within ±6.4 arcsec/sec, 3 sigma, per axis. This corresponds to crossing a thruster deadband of ±0.1 degrees in about 2 minutes. This is relatively frequent for precision imaging, but jitter is expected to remain low because the Orbiter is using MIT thrusters with small minimum impulse. The ACS baseline is capable of tighter pointing stability if needed to insure less frequent thruster firings.

Pointing stability for the scanning LIDAR needs to be within ±0.14 deg/sec (±2.5 mrad/sec), 3 sigma, per axis.

Spacecraft translation control during terminal rendezvous and OS capture is as follows:

- Position control in transverse axes (perpendicular to velocity) to within ±14 cm, 3 sigma, relative to the OS. It corresponds to a 3-sigma radial error of about 20 cm.
- Velocity control in transverse axes to within ±3 mm/sec, 3 sigma, relative to the OS. This corresponds to translation across a ±14 cm distance in a little over 90 seconds.
- Velocity control along the velocity vector to within ±3 cm/sec, 3 sigma, relative to the OS. This assumes that the forward velocity relative to the OS will be significantly > 3 cm/sec during the capture phase.

The ACS subsystem includes propulsion control and propulsion drive electronics. Stellar inertial attitude determination using precision star tracker and gyros:

- LN-200S IMU contains 3 gyros and 3 accelerometers. Two IMUs included for redundancy. Gyros provide bias stability to within ±0.3 deg/hour, per axis. TRL of 9.
- Denmark Technical University (DTU) micro-Advanced Stellar Compass (μASC): a fully internally redundant star tracker representative of technology that should be available by the technology cutoff date of 2007.
- DTU μASC star tracker contains 4 heads, 2 sets of electronics, and 2 power supplies that are cross-strapped for full internal redundancy. It provides 3-sigma accuracy to within ±3 arcsec in 3 axes. With a mass of 2.2 kg and power of 4 W it has a current TRL of 6.
- Other star tracker vendors are also developing comparable devices that should be available at TRL 6 by 2007.

Coarse analog sun sensors and precision gyros are used for safe mode and initial deployment. For communications with Earth, there will be a 1.0 m X-band HGA attached to the bus using gimbals with 2 degree-of-freedom (DOF). Two solar arrays will each be attached to the bus with gimbals with 1 DOF. The ACS is block redundant.

Telecom on the Orbiter will be capable of determining the direction of the OS from a distance of hundreds of kilometers but the Orbiter also carries a payload containing components needed to find and retrieve the OS. This payload includes:

- Optical navigation camera capable of imaging the OS from several thousand kilometers.

- Scanning LIDAR capable of determining OS range, direction, and relative velocity within a few thousand kilometers.
- Orbital Sample Capture and Retrieval (OSCAR) device.
- Mechanism for transferring the OS to an Earth Entry Vehicle (EEV).

Telecom — The Rendezvous and Relay Orbiter needs to support a DTE return link rate of 40 kbps to a 70m station at X-Band at a range of 2.35 AU during rendezvous and 2.47 AU after rendezvous. It will also have an UHF transmit/receive capability for telemetry and beacon tracking. Course RF tracking of the MAV will use Doppler measurements of a stable UHF signal. The Electra payload is used on the orbiter to track the RF signal. The UHF package will also support communications for MSR surface elements.

Telecom hardware that is not necessary for the Earth return trip will be jettisoned from the spacecraft. This is done to decrease mass of the return vehicle. Unnecessary hardware includes the HGA assembly (including the antenna, rotary joints, gimbals, HGA boom, and other HGA support structure), and the UHF proximity relay hardware.

The communications system was designed to support the following links:

1. Early outgoing cruise will be on the LGA out to 0.3 AU at 200 bps down-link and 1.0 kbps up-link with Doppler and ranging using the 34 m DSN subnet.
2. Latter outgoing cruise will be on the 1.0 m HGA out to 2.47 AU at 3.0 kbps down-link and 2.0 kbps up-link with Doppler and ranging using the 34 m DSN subnet.
3. Rendezvous with MAV will be on the 1.0 m HGA out to 2.35 AU at 40 kbps science down-link using the 34 m DSN subnet.
4. Incoming cruise will use the 9 dBic MGA starting at 1.70 AU at 100 bps down-link and 250 bps up-link with Doppler and ranging using the 34 m DSN subnet.
5. Emergency communication support will be provided by the 19 dBic MGA starting at 2.47 AU at 40 bps down-link and 31.25 bps up-link using the 70 m DSN subnet.

The X-band DTE telecom subsystem consists of a 17 W RF (55W, DC) transmitter with a 1.0 m HGA (36dBi). A MGA (19 dBi) is used for the return to Earth after the HGA is jettisoned. Two LGAs (6.5 dBi) provide emergency coverage at short spacecraft-Earth ranges. The transponders require 14W, DC in transmit/receive mode. Additional hardware includes diplexer (x2), -3dB hybrid coupler, rotary joint (x2), WGTS, CXS (x2), coax and wave-guide. The hardware design utilizes cross-strapped, dual string redundancy. If a transponder and transmitting amplifier fails, or the wave-guide transfer switch (WGTS) fails, the telecom subsystem remains fully functional.

The UHF telecom subsystem is the Electra RF package, consisting of a transponder with a dual channel receiver and two UHF LGA's (~6dBi). Two antennas are used to accommodate increased telecom coverage from various spacecraft orientations. The Electra package contains an USO to permit Doppler measurements. There is enough margin in the UHF link budget to lower the peak gain of the UHF antenna. This will broaden the beam pattern of the antenna to increase the coverage area. The system requires 13W, DC while in receives mode and 68W, DC while in transmit and receive mode.

CDS — Standard top level Orbiter requirements are:

- Telemetry processing (payload data rate is 12 kbps)
- Up-link command processing and distribution
- Maintenance and distribution of spacecraft time
- Storage of science and engineering data
- Spacecraft fault protection
- Subsystem control and services
- Thermal control

The Rendezvous and Return Orbiter design essentially consists of two boards, a processor board and a memory/payload interface board. The processor performs all navigation functions except those needed for capture of the Orbital System. For this case, the OSCAR computes the navigation and provides ΔV inputs to the CDS processor for navigation commands.

The processor uses a GD-603r and consists of 12 Mbytes of memory (vendor plans to upgrade this amount). This processor board is modified to remove the 1553 bus controller saving power, and add an interface to telecom for command inputs. There are also several high-speed serial interfaces available on this board for ACS if needed. The processor board includes a timer for spacecraft time. The design is block redundant.

The instrument interface board is the MER NVM/Camera board. The memory on this board is 2 Gbits flash and 4 Mbytes EEPROM. It presently supports two high-speed serial interfaces (10 Mbps). The total mass is 6.6 kg and power is 13.3 watts (during sample capture).

The OSCAR will calculate the required delta-V for rendezvous and capture and provide this information to the spacecraft, which will use its actuators to deliver the required delta-V.

Propulsion — The propulsion system must provide a 1794 m/s velocity increment for TCMs and MOI, support aero-braking and OS rendezvous/capture. After accommodating a drop/jettison mass of 112.8 kg, provide 2139 m/s for the Earth return.

Dual mode chemical propulsion meets the high delta-V requirement. The dual mode propulsion system has a bang-bang pressurization and four Leros 1-c, 467N, NTO-Hydrazine engines with a specific impulse of 325s. Four engines are used to reduce the Mars orbit insertion burn duration and consequent gravity losses. These will also be off modulated for pitch and yaw control. Sixteen 0.7 N MIT thrusters for provided for redundancy and agile maneuvering in Mars orbit to acquire the OS. Mechanical failure of a MIT thruster is considered highly unlikely based on previous mission experience with hydrazine thrusters. However, there is the possibility of using less than the full set of 16 thrusters as a backup for rendezvous and capture.

Power — Solar Arrays have dual wings to minimize solar torque. Single axis articulation with roll-steering to maintain Sun pointing removes cosine losses. The Solar arrays are small Ultra-flex arrays that use quad junction 3.3V, 33% efficient cells. Aero-braking with Ultra-flex arrays has been considered and is expected to be feasible in this time frame. Specification for the solar array is as follows:

- 1167 Watts at 1.67 AU at EOL
- 3412 Watts at 1 AU / AM0 / 28 C
- Total area 10 square meters
- Diameter 2.5 meters
- Total mass 18.6 Kg

The secondary battery is a single string IPV NiH2 Battery providing a dual fault tolerant design. It is sized by OS acquisition mode in Mars' orbit with an eclipse time of 47 minutes. A worst case 40% DOD was assumed. The 57 A-hour IPV NiH2 battery has a mass of 35 Kg. The secondary battery is supplemented by primary battery during the launch phase. Approximately 1570 W-hours can be provide by the primary battery. It mass is 7.0 Kg. Power electronics were sized parametrically and are estimated at 27.9 Kg.

Structure — The Rendezvous and Return Orbiter supports the Entry System and Lander mass during launch (Table 7). The Carrier Stage mass will also be supported by the Orbiter but will be attached to the base of the Entry System (Figure 1). After launch the Entry System containing the Lander with attached Carrier Stage will be separated from the top of the Orbiter. The Orbiter will then be separated from the launch vehicle adapter. Ultra-flex solar arrays with single axis articulation will then be deployed. The Orbiter configuration has passive dynamic stability during aero-braking by providing a center-of-pressure / center-of-mass offset. The 2-axis articulated HGA will be deployed after aero-braking. Following OS capture and transfer to the EEV, the OSCAR and other elements not required for Earth return (112.8 kg drop mass) will be separated. A final separation event occurs when the EEV is targeted and released for Earth entry. The Rendezvous and Return Orbiter mass was calculated using the Team-X parametric sizing tool that considers total launch mass, articulation elements, separation hardware, propellant mass and secondary structure.

5. TECHNOLOGY

Many technology advances are needed to enable a Mars sample return mission. The spacecraft will require novel, lightweight, and components and must be able to perform an autonomous precision landing on Mars's surface while avoiding hazards. An autonomous Mars launch system will need to be developed along with an autonomous system for rendezvous and capture of the OS.

Planetary protection posses a significant design driver for the mission. Stringent measures are required to prohibit unplanned impact with the Earth. Sterilization of return hardware that directly contacted Mars will be required. This along with the requirement for assured containment of samples affect all phases of the mission, namely the outbound leg; sample acquisition, transfer and storage; sealing of the sample container; monitoring of the sample; return phase of the mission; Earth entry phase; and sample receiving laboratory.

MAV - The second stage is a new bi-prop system, which requires the development of lightweight engines, valves, and tanks. It also requires the development of a main engine and RCS thrusters that use the MMH-MON25 propellant combination.

The first stage solid motor is a new development While not a major development, it must be able to operate at a low grain temperature. Also, ACS requirements for the head-end steering system require significant improvements in SRM thrust alignment.

Entry System - Technology development for the heat shield material could reduce the entry system mass, although current technology material, inherited from Pathfinder/MER, will meet current flight requirements.

Algorithms and software for aero-maneuvering are being developed for the Smart Lander program, which are currently at a TRL level of 4 or 5. If this technology is not carried to completion, it must be developed for this mission. The Smart Lander is not expected to fly before this missions technology cutoff date of 2007.

Lander - The Lander platform is expected to have heritage from Smart Lander but the following new technology is required for this mission: A throttleable lander engine must be developed to support this mission. The current status of the scanning laser mapper (LIDAR) used for generating local elevation maps during descent is a new development and currently at a TRL of 4. Algorithms and software for hazard identification and avoidance during terminal descent is new and the current TRL is estimated at 2 to 3. The MAV hanger and WEB can use conventional technology, although the use of lightweight CO_2 insulation for both the WEB and MAV hanger would reduce thermal subsystem mass.

Rendezvous and Return Orbiter - Several elements need development: The OSCAR for search and acquisition of the OS, sample transfer mechanism to the OS while in the MAV on the Lander, transfer to the EEV, and the EEV ejection mechanism. Technology development is required for the algorithms and software required to for supporting rendezvous. The power subsystem is utilizing quad-junction solar cells, which may not be available within the development schedule; the fall back will be triple-junction cells. Ultra-flex solar arrays need to be qualified for aero-braking. The propulsion system utilizes new technology, but the technologies which include light weight tanks, high performance NTO-hydrazine engines, pressurization and MIT thrusters are expected to be developed within the development schedule for this mission.

ACS Technology - Relies on the availability of JPL MESO gyros to keep the ACS mass and power low. Gyros must be low mass and provide bias stability to within ±0.1 deg/hour, 3 sigma, per axis. The JPL MESO gyros are currently at TRL 4. The fallback is to use much more massive gyros, which would mean a large increase in ACS mass.

Assumes that manufacturing capability for solid rocket motors will improve significantly by the technology cutoff date. The SRM on Stage 1 will need to have a worst case thrust vector misalignment 5 times better than currently achievable. This represents a significant technological advance. The fallback is to lower the peak thrust from the SRM or increase thruster size. Neither option would require new technology.

6. CONCLUSIONS

A Mars Sample Return mission satisfying the objectives of the Mars Program Plan could be feasible for launch in the proposed time frame with aggressive investment in the technology areas described in the previous section. With the proposed launch date for this mission, it is reasonable to assume that Smart Lander will fly first. However, Smart Lander is not expected to fly before the technology cutoff date of 2007 so significant schedule risk exists for the proposed launch date. Also, the Delta IV Heavy launch vehicle is also problematic but other launch options should exist.

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