Mars Sample Return Spacecraft Systems Architecture[1]

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Abstract—The Mars Sample Return mission plans to collect sets of samples from two different sites on Mars and return them to Earth in 2008. The mission consists of 15 different vehicles and spacecraft plus two launch vehicles, with elements being provided by the U.S, France, and Italy.

These vehicles include two U.S. provided Landers, each with a sample collection Rover, Mars Ascent Vehicle, and an Orbiting Sample satellite. France is providing the sample return Orbiter which carries a U.S payload for sample detection and capture plus two Earth Entry Vehicles for landing the samples on Earth. The Orbiter also delivers four NetLanders to Mars for performing unique surface science.

Significant in-situ science is included. New technologies are being developed to aerocapture into Mars orbit, to collect and safeguard the samples, to launch the samples into Mars orbit, and to enable autonomous Mars orbit rendezvous and capture for return to Earth.

TABLE OF CONTENTS

1. INTRODUCTION

The Mars Sample Return (MSR) mission, scheduled for launches in 2003 and 2005, is an ambitious plan to collect sets of scientifically valuable samples from two different sites on Mars. The individual samples will be well documented with in-situ measurements, so that the context of the samples is well understood, and they will be isolated from each other in a sealed canister. The samples will be launched into Mars orbit and retrieved by an Orbiter for return to Earth in 2008.

In addition to returning samples for analysis on Earth, significant in-situ science will be conducted. The Lander will include a suite of experiments to be performed on the surface of Mars, including imaging. The Rover will feature the Athena science payload to perform analyses of Martian material plus its own imaging system. The Orbiter will deliver the NetLanders, which are 4 independent small landers with science payloads that will function together as a network on the surface of Mars.

2. MISSION OVERVIEW

The initial MSR campaign described in this paper is comprised of two launches, one in 2003 and one in 2005. Each of these flights will result in the placement of a single sample canister in Mars orbit, and each of these canisters will contain about 500 grams of Martian rock and soil. The 2005 launch will also include a French Orbiter which will rendezvous with, and capture, the two canisters. Each of the Orbiting Samples (OS) will be placed into its own Earth Entry Vehicle (EEV) for return to Earth. The EEV's will be delivered to their Earth re-entry trajectories by the Orbiter and released.

2003 Mission

The '03 launch is on a US provided intermediate launch vehicle and sends one Lander System to Mars. The Lander will be targeted to a selected site between -5 deg and +15 deg latitude. This allowable latitude range is driven by the seasonal Mars solar conditions and the power generation capabilities of the Lander. The Lander and Rover will conduct a 90 day surface mission, collecting samples with both the Rover and with a Lander Based Sampler (LBS) system provided by Italy.

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The nominal surface mission will conclude with the launch of the Mars Ascent Vehicle (MAV), placing the OS into Mars orbit with its precious cargo of samples, awaiting retrieval by the French Orbiter. The Rover may conduct an extended mission with a UHF radio link to Mars data relay assets such as Mars '01 and Mars Express.

Key '03 mission dates are provided by Lee, et al [1]:

- Launch – May '03
- Lander Mars entry and landing – Dec. '03
- MAV launch – Mar. '04

2005 Mission

The '05 launch is on a French provided Ariane 5 which will deliver both a Lander System, an Orbiter, and 4 NetLanders to Mars. The Lander will be targeted to a selected site between +5 deg and +25 deg latitude, driven by the seasonal Mars solar conditions and the power generation capabilities of the Lander. Like the '03 mission, the Lander and Rover will conduct a 90 day surface mission, collecting samples with both the Rover and with the Italian LBS.

The nominal surface mission will conclude with the launch of the MAV, placing the OS into Mars orbit, awaiting retrieval by the French Orbiter. The Rover may conduct an extended mission.

The Orbiter will search for the two OS by tracking their radio beacons, and the ground will determine the orbits so that the Orbiter can be commanded to the proper orbits to retrieve them. The terminal rendezvous and capture phases are autonomous.

With the two OS captured and placed in their respective EEV's, the Orbiter will propulsively return to Earth. The Orbiter will target the EEV's to the proper entry corridor and release them shortly before Earth entry. Then the Orbiter will perform a deflection maneuver to miss the Earth.

Key '05 mission dates are provided by Lee, et al [1]:

- Launch – Aug. '05
- Lander Mars entry and landing – Jul. '06
- Orbiter Mars orbit aerocapture and insertion – Jul. '06
- NetLander Mars entry and landing – Jul. '06
- MAV launch – Oct. '06
- Orbiter departure for Earth – Jun. '07
- EEV entry – Nov. '08

3. Mission System Description

The '03 mission contains the following major elements:

1. Lander – Derived from the Lockheed Martin Mars '01 design.
2. Rover – A 6 wheeled vehicle which is significantly larger than the Sojourner rover.
3. Mars Ascent Vehicle (MAV) – A two stage launch vehicle which can place a 3.7 kg payload into Mars orbit. The MAV solid rocket motor booster system is a completely new development managed by Kennedy Space Center.
4. Orbiting Sample (OS) – A 16 cm diameter spherical Martian satellite which contains a sealed sample set and has a radio beacon for location by the retrieval Orbiter in '06.

The '05 mission contains the following major elements:

1. Lander – A build to print of the '03 Lander with some minor science payload changes.
2. Rover – A build to print of the '03 Rover
3. MAV – A build to print of the '03 MAV
4. OS – A build to print of the '03 OS
5. Orbiter – A French supplied Orbiter which delivers 4 NetLanders to Mars, aerocaptures into Mars orbit, and retrieves the 2 OS. The Orbiter carries a US-provided payload (OSCAR) with the necessary equipment to detect and capture the OS.
6. NetLander – These 4 Landers are provided by France, are delivered to Mars by the French Orbiter, and operate on the Martian surface as a network.
7. Earth Entry Vehicle (EEV) – These 2 entry vehicles are provided by NASA Langley and each delivers one OS to the surface of the Earth for recovery.

The masses of the major mission elements are provided in the table below, for the '05 mission.

MSR Element	Launch Mass (kg)
Lander	**1800**
Cruise Stage	100
Backshell	329
Heatshield	150
Lander Bus	661
Lander payload	**390**
MAV	**160**
Stage 1	119
Stage 2	21
Launcher	20
Rover Systems	**138**
Rover	90
Support Equipment	48
Sample Transfer Chain	**47**
Orbiting Sample (OS)	3
Lander STC Equip.	29
Lander-Based Sampler	15
Lander-Based Imaging	**6**
Additional Payloads	**40**
Lander Propellant	170
Orbiter	**2700**
Orbiter	**505**
Orbiter Cruise Stage	**155**

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Heatshield	**250**
Orbiter Propellant	**1400**
Net Landers	**260**
OSCAR	**130**
Base Structure	21
MORS	35
SCATS	24
Earth Entry Vehicle	50
SYLDA (to support Lander)	**450**
Launch Adapters	**250**
Total Launch Mass	**5200**

4. DRIVING REQUIREMENTS AND SCIENCE GOALS

Planetary Protection

Mars must be protected from forward contamination by Earth organisms, to the degree practical using current standards and processes. More importantly, Earth must be protected from the uncontrolled release of any unsterilized Martian material.

Samples

The requirement is to return 500 grams of Martian rock fragments and soil to Earth with sufficient diversity and context to characterize the geology of the landing site area. Samples should be isolated and kept to below 50 deg C. No Earth biological contamination can be returned in the Martian samples, in order to avoid ambiguity in the analysis of the material.

Additional Science

The MSR mission has additional science requirements for the NetLander network and for other science payloads to be supported by the Rover and by the Lander.

5. KEY TRADE STUDIES

One of the key trade studies for MSR was selecting the Mars Orbit Rendezvous (MOR) architecture versus direct landing and return. Just as Apollo decided on a rendezvous architecture, we have chosen MOR, because it results in a significantly lower launch mass. Also driven by launch mass was the decision to implement aerocapture for Mars Orbit Insertion (MOI) versus a propulsive MOI, and the decision to utilize direct Earth entry for the EEV versus placing the sample in Earth orbit for retrieval.

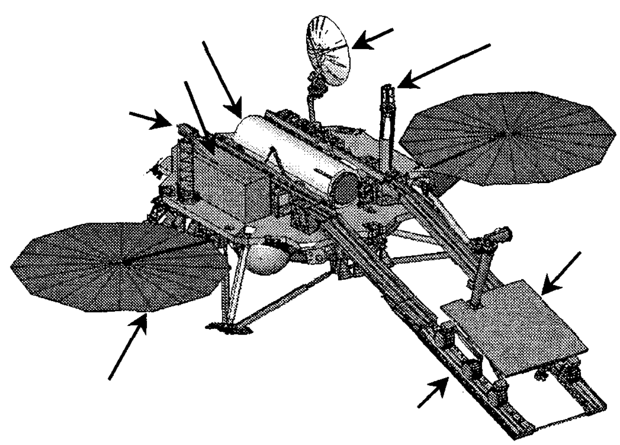
A soft landing system was chosen over Mars Pathfinder (MPF) style airbags because of the amount of mass required to be landed on Mars. The MPF system was not readily scalable to landing a 400 kg payload, whereas the Mars '01 soft landing system was scalable without requiring extensive new technology developments. the soft landing system was also estimated to be a lower mass design and capable of providing more volume to the payload.

A two-stage solid rocket MAV was chosen over a two-stage liquid fueled MAV, because the lower mass fraction was more important than the higher specific impulse (Isp) of a liquid rocket system. There were also cost and technology issues factoring into this trade. Direct capture of the OS was chosen over a docking and transfer operation because the former approach resulted in lower mass, lower complexity, and lower cost.

6. LANDER

The design of the MSR lander is focused on satisfying several key functional requirements. First, the lander must deliver a payload that supports sample collection operations to the surface of Mars. Second, the lander must support at least 90 Martian days (sols) of surface activities geared toward the collection of at least 500 grams of samples. Finally, the lander must enable the launch of the collected samples into low-Mars orbit. Satisfaction of these requirements led to a design resulting in the largest spacecraft that will ever to touch down on the surface of the Red Planet. With a launch mass of 1850 kg and a landing mass of nearly 1065 kg, the lander's weight on the surface of Mars will exceed its Viking predecessor by 85%.

In part, the large lander mass is required to enable the delivery of the 400 kg of payload needed to support mission operations, including the large Athena class rover that will extract rock core samples, an Italian-provided drill to collect sub-surface samples, a two-stage Mars Ascent Vehicle to launch the collected samples into low-Mars orbit, and several scientific payloads that will advance technology critical to future human Mars exploration efforts. The capability to land such a large payload is the most driving requirement on the design. Every kilogram of payload mass decreases the amount of mass available to solve challenging problems such as making the spacecraft rugged enough to support a safe touchdown at rocky sites. Satisfaction of the mass requirement is further complicated by the volume requirement. In order to launch on an intermediate-class



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launch vehicle, the maximum diameter was limited to 3.65 meters.

While en route to Mars, the lander will ride inside the protective confines of a conical-shaped, 3.65-meter-diameter aeroshell with an ablative heatshield at the bottom. A cruise stage consisting primarily of solar panels attaches to the top of the aeroshell and will provide power during transit to the Red Planet. This stage will be jettisoned approximately five minutes prior to entry into the Martian atmosphere. As the lander plunges toward the surface, a 15 cm center of gravity offset will enable the spacecraft to achieve a 0.18 lift to drag ratio. The lifting profile will be needed for energy dissipation because the lander is too heavy to utilize a pure ballistic descent trajectory.

An exploded view of the Lander System cruise configuration is shown in Figure 1.

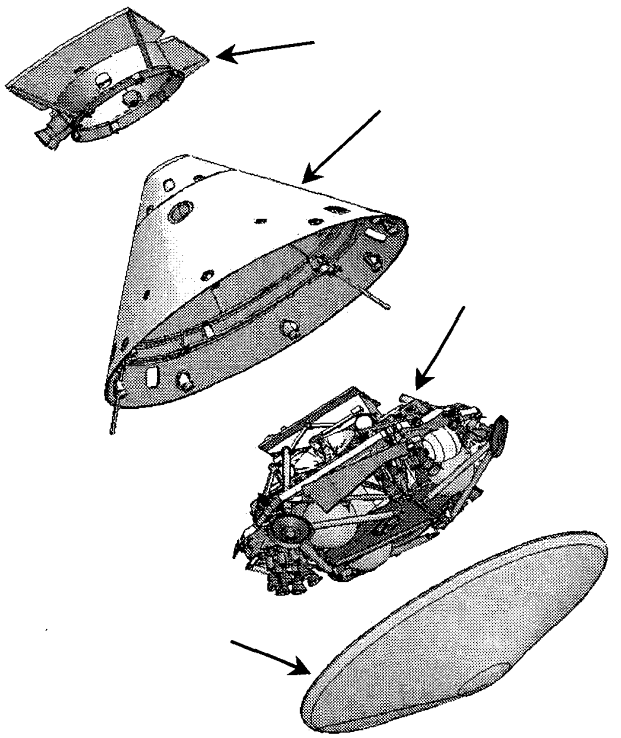

Figure 1 - Lander Cruise Configuration

A soft touchdown will be achieved using a combination of a 20-meter-diameter, Viking-derived parachute and three clusters of retro-rockets for the terminal descent. Several of these rockets are canted by 20 degrees from the vertical and will be used during the final 20 meters of descent to minimize landing site alteration due to plume effects. Pathfinder-style airbags will not be employed because calculations show that the system mass needed to support the heavy MSR lander would have weighed more than the 115 kilograms of propellant needed to achieve a propulsive landing.

The morphological design of the lander's surface configuration is based on a hexagonal-shaped deck supported by three deployable legs. When fully extended, these legs will elevate the deck nearly one meter above the Martian surface. All of the payload rides on top of this deck, while the bottom houses the propulsion system and a physical enclosure to provide a warm environment for the spacecraft's avionics. Surface power will be provided by two deployable solar panels. The deployed Lander surface configuration is shown in Figure 2.

Another key requirement on the lander involves the ability to support surface operations leading to the collection of 500 grams of samples. This requirement drives the need to provide over 2000 W-hrs of energy per sol for drilling functions and the daily transmission of 70 Mbits of data. Previous design landers (Mars 1998 and Mars 2001) relied on low-power UHF communications to a relay orbiter circling Mars as the prime method for sending surface telemetry back to Earth. However, analysis indicates that overflights from relay orbiters will not occur at intervals regular enough to support the twice-per-day communications opportunities needed for Earth-based planning of drill and rover activities. Consequently, the MSR lander design was forced to utilize the higher-power, direct-to-Earth

Figure 2 - Lander Deployed Configuration

communication mode using X-band.

Lockheed Martin Astronautics is JPL's industrial partner in this challenging endeavor and will build the MSR lander at their Denver facility. In order to minimize development costs and risk, a key design consideration was to maximize heritage from previous designs. Much of the avionics and flight software was derived from Lockheed's design for the Mars 2001 lander, and the entry, descent, and landing systems claim significant heritage from Viking. By utilizing this paradigm, the MSR lander team has been able to eliminate new technology developments from a program heavily challenged by design-to-cost constraints.

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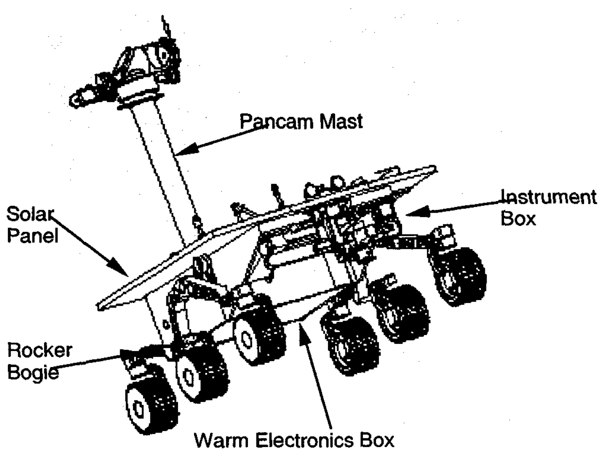
7. ROVER

The Athena Rover for the Mars Sample Return (MSR) mission must meet the following requirements:
document and collect a set of sample consisting of 45 rock cores and 2 soil samples; ensure that a minimum of 5 grams of sample mass is collected from each distinct geologic site; transfer samples from the Athena Rover to the MSR lander; accommodate the Athena science payload which consists of the Panoramic Cameras (Pancam), miniature Thermal Emissions Spectrometer (miniTES), miniature corer (minicorer), alpha proton x-ray spectrometer (APXS), Moessbauer spectrometer, Raman spectrometer, and a microimager. In addition, the Athena Rover is required to document and explore the sample site (i.e., establish the context of the collected samples); perform the primary mission for 90 sols after Mars landing at a mid-latitude landing site, with a goal of operating 180 sols total; traverse in terrain comparable to Viking 1 and Pathfinder landing sites; during traversing achieve (as a goal) an integrated distance traveled of 1 km ('03 mission), 5 km ('05 mission); and have the capability, as a goal, to operate beyond line-of-sight and over the lander's visual horizon.



The concept design for the Athena Rover is depicted below and is as described in Figure 3.

Figure 3 - Athena Rover

The Athena Rover is a 6-wheeled vehicle, 75kg in mass (including payload), and 131cm long, 110cm wide and 150cm tall in its deployed configuration (Pancam assembly deployed). It is stowed on the Lander deck in configuration with its Lander mounted rover equipment (LMRE) for launch and during the cruise-to-Mars phase of the MSR mission. At deployment, the Lander fires cable-cutting pyros, releasing tie-downs which restrain the rover in this position. The Lander is also commanded to release, then deploy a set of ramps which provide for rover egress to the surface of Mars. Once on the Mars surface, the rover is commanded to release, then deploy, the Pancam mast so that the deployed configuration is achieved.

The rover has a ground clearance of 25cm. The distribution of mass on the vehicle has been arranged so that the center of mass is near the +X face of the Warm Electronics Box (WEB) and at a height to the center of the WEB. As a consequence, the vehicle could withstand a tilt of 45deg in any direction without over-turning, although fault-protection limits prevent the vehicle from exceeding tilts of 35deg during traverses. This configuration of the center of mass is also suited for drilling, as the minicorer is located inside the instrument box and mounted to the +X face of the WEB.

A rocker bogie design is used which allows the traversing of obstacles of more than a wheel diameter (20cm) in size. Each wheel has cleats and is independently actuated and geared, providing for climbing in soft sand and scrambling over rocks. The front and rear wheels are independently steered, allowing the vehicle to turn in place. The vehicle has a top speed of on flat ground of 6cm/sec. Under control for hazard avoidance, the vehicle achieves a top speed of 1m/min.

The rover is powered by a 1.2sqm solar panel comprised of 55 strings of 20, 5.5mil GaAs cells. The solar panel is backed up by 3, 5amp-hr lithium-ion rechargeable batteries, providing (at nominally 16V) up to 225W-hr of energy. The combined panel and battery system allows the rover to draw over 80W of peak power while the peak panel production is more than 50W for 3hr each sol on Mars, the power needed for drilling. The power requirement for driving is 21W.

Rover components not designed to survive ambient Mars temperatures (-90degC during a Martian night) are contained in the warm electronics box (WEB). The WEB is insulated by a vacuum/CO2 air gap, coated with low-emissivity paints, and heated under a combination of waste heat from electronics, radioisotope heating units (RHUs) and resistive heaters. The thermal design also utilizes a miniature looped-heat-pipe system, which is intended to transfer heat from batteries to a radiator mounted on the –X WEB wall. This design maintains the batteries at a temperature above –20degC for discharge, 0degC for recharge and storage, between -30degC and +30degC for survival during all mission phases. All other electronic components within the WEB must be maintained between -40degC and +40degC during all mission phases including operations on Mars. Initial analysis of this thermal design suggests this will be true.

Computer control of the rover is provided by an integrated set of computing and power distribution electronics. The computer is a 32bit R3000 Synova Mongoose processor with a 12MHz clock rated at 10Mips utilizing a VxWorks operating system. There are 4 types of memory supporting the processor: 32Mbytes of DRAM, 4Mbytes of EEPROM for code storage, 64Mbytes of Flash memory for nonvolatile

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data storage and 128Kbytes of PROM for boot code. The power distribution system conditions the nominal 16V power to users within the computing system: 5V for logic, cameras and A/D, ±15V for the gyro and accelerometers, 3.3V for FPGAs and encoders. The power distribution system also provides for currrent limited power to be supplied to the actuators preventing computer brownouts during rover operation. I/O is supplied by a high speed 1Mbps serial port and a low speed 9.6kbps port which are shared among the subsystems and payload elements. A special serial port is provided to readout and buffer data from the twelve rover cameras. These cameras are: 2 color-capable stereo-capable cameras comprising the Pancam instrument, a pair of stereo-capable cameras to be used for rover navigation, 2 pairs of 2 stereo-capable cameras mounted on the +X and –X faces of the rover to be used in hazard avoidance, 2 stereo-capable cameras mounted in the instrument box to be used to image drilling and instrument placement operations, a camera with a set of optics suited for closeup images of a sample site (called the microimager), and a camera with optics suited for use as a sun sensor.

Battery charge is regulated by a separate battery charger board (BCB) which monitors bus voltage, battery temperature and battery state of charge. This board contains a separate microcontroller and logic to manage the battery throughout all mission phases. This board also contains the mission clock and a separate primary battery pack to maintain time and to serve as a wakeup timer for initiating rover operations conducted by the main computer.

The software in the main computer of the rover, once initiated, executes a control loop which monitors status of the vehicle, checks for the presence of commands to execute, maintains a buffer of telemetry and performs health checks. Activities such as imaging, driving, drilling or instrument operations are performed under commands transmitted in a sequence to the rover from a ground control station. Each command execution results in the generation of telemetry, which is stored for eventual transmission.

Control of the rover is performed in a three-tiered manner at each degree of freedom on the vehicle. At the lowest level, a pulse-width modulated control loop services an individual motor. Feedback from the individual motor to a position control loop coordinates several degrees of freedom. The third tier is a monitoring control loop, which assesses vehicle safety from measurements from sets of sensors on the rover.

As an example, in driving with hazard avoidance, the lowest control level is driving the motors in each of 6 wheels with the objective of achieving (at the second control level) an average of motor encoder counts which correspond to the commanded distance traveled by the rover. While the motors are actuated and the vehicle is moving, the third level of control, monitoring of accelerometers and the rate gyro, is performed to detect anomalous tilts and driving off-course. When the motors are powered off and the vehicle is stopped, the computer conducts a proximity and hazard detection function, using its stereo camera system to determine the presence of obstacles in its path. The vehicle path objective is modified to steer autonomously to avoid obstacles. Then, after the obstacles are no longer detected by the vehicle, the rover continues to drive to achieve the original commanded goal location. While stopped, the computer also updates its measurement of the distance traveled (integrated encoder counts) and heading, using the rate gyro and sun sensor. This calculation provides an estimate of progress toward the goal location.

Command and telemetry functions on the rover are provided by S-Band radios located on the rover and the lander. These radios are capable of a rate of 256 kbps in telemetry transmission (rover to lander) and a rate of 8 kbps in command transmission (lander to rover). Data and command sequences at the lander are transmitted to/from earth by an X-band transmission system. Given that the rover has a goal of over-the-horizon operations independent of the lander and a goal of an extended mission beyond the expected lifetime of the lander, the rover also carries a UHF radio for rover to orbiter relay communications. This radio is capable of a rate of 256 kbps in telemetry transmission (rover to orbiter) and a rate of 8 kbps in command transmission (orbiter to rover).

In operation on Mars, the rover is the key vehicle for collection of a diverse set of samples during the MSR mission. The rover obtains a panoramic image using cameras on the Pancam and a spectral measurement from the miniTES to send to ground operations. An assessment of these images leads to a selection of likely rock or soil targets for sampling. The rover is commanded to drive to a location and carefully position so that a sample can be acquired. A core sample is collected, imaged by the microimager and the sample minerology measured by the Raman spectrometer. Additional measurements are obtained using the APXS and Moessbauer spectrometers at this sampling site. These measurements lead to an assessment of the diversity of the sample with respect to other samples collected during the rover mission. Additional cores may be collected at this rock or soil from this site before the rover moves to the next location for sampling. This activity flow (panoramic imaging, driving to a location, sampling and measurement) continues until sufficient samples are collected to warrant a trip back to the lander for sample transfer. The rover drives back to the ramps, positions itself at the base of the ramps, and then regresses to a fixtured location on the lander for sample transfer. In this location, the sample cache manipulator positions over an aperture in the MAV third stage leading to the sample canister. The manipulator then releases filled cache segments into the canister completing the sample transfer. The rover returns to the Martian surface to resume its sample collection mission.

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During sample collection, it is estimated that the rover will gather approximately 5Gbits of data in support of documentation and planning for operations. The challenge will be to maintain the pace of rover activities leading to the collection of the required sample set and to transmit this volume of data in support of these activities. The sampling will be conducted with the first-of-its-kind miniature drill and soil scooping system.

8. MARS ASCENT VEHICLE

The Mars Ascent Vehicle that will be a part of the '03/'05 Mars Sample Return Mission is a two stage solid propellant rocket that will insert an Orbiting Sample (OS) into orbit around Mars.

The MAV is divided for implementation into three components, a JPL provided Payload Assembly, a Kennedy Space Center managed Booster System which is currently in the process of being contracted to industry and a JPL provided system thermal enclosure.

Functional Requirements

The functional requirements levied on the MAV may be summed up by the following statement: From its stowed position on the Lander, the MAV shall put the OS into Mars orbit.

The most important derived requirements are those which specify the orbit.

The MAV shall insert the OS into a Mars Orbit with a semi-major axis of 3987 km ±100 km (99.7 %). The magnitude of the semi-major axis directly affects MAV mass and Orbiter rendezvous delta V. The error component of the requirement has driven the need for various forms of control on the MAV.

The MAV shall meet a prescribed inclination (between 40 and 46 degrees) to within 1 degree (99.7%). This requirement has also driven the need for a controlled ascent.

Key MAV Trade Studies

The original Mars Ascent Vehicle for the '03/'05 Mars Sample Return Mission was a fully controlled liquid propellant vehicle where the third stage contained the samples and actively participated in rendezvous with the Orbiter up to three years after ascent. While more robust, the Lander was not capable of landing the mass of this version of the MAV.

In order to solve mass and other problems, the MAV baseline was changed to a solid propellant vehicle that was entirely gyroscopically stabilized. Unfortunately, detailed trajectory analysis demonstrated that the injection accuracy requirements were not being satisfied.

The current control strategy uses thrust vector control, a cold gas system and spin stabilization during different mission phases. As the MAV and other elements mature, thrust termination or nutation control may need to be added.

The initial liquid and solid MAVs were three stage vehicles. Mass, cost, complexity and risk trades all favored a two stage design.

Baseline System Design

Figure 4 shows the MAV separated into its three major ascent components, Stage 1, Stage 2 and an aerodynamic fairing which covers Stage 2.

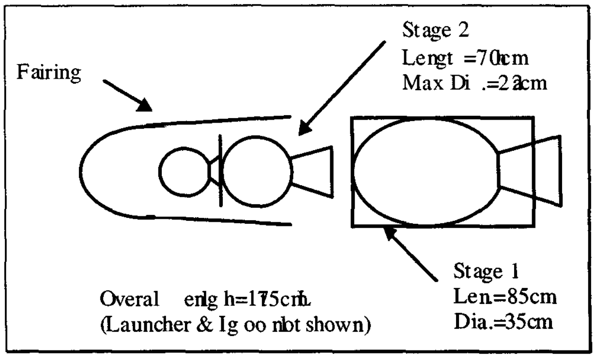

Figure 4 - Basic MAV Configuration.

Also included as part of the flight system are an Igloo (thermal control system) and launcher (not shown). All MAV components are stowed horizontally on the Lander deck until ascent.

Figure 5 depicts a simplified MAV block diagram. The blocks shown exclude structures and mechanisms for clarity, but otherwise constitute the minimum functionality necessary to meet MAV requirements.

Most notably, the Avionics, Power and Pyro functions are only resident on the first stage, which saves a great deal of mass. The S1 and S2 motors each provide half of the total MAV delta V. The cold gas system provides control and spin-up functionality. The Ascent Status Radio provides a telemetry link to a Mars orbiting asset.

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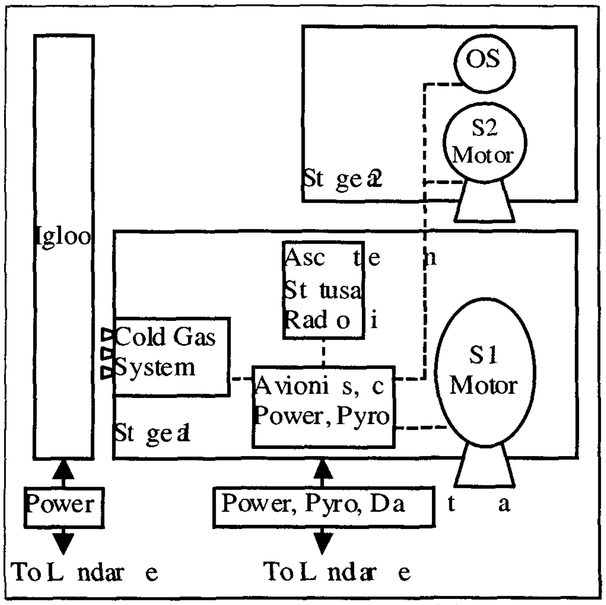

Figure 5 - Simplified MAV Block Diagram

Two mass lists are provided, dividing the MAV by deliverable mass and by on-ascent mass. Table 1 lists the deliverable pieces of MAV. The three different major deliverables are the Payload Assembly (PLA), the Booster System and the Igloo. Please note that at the time of writing, a Booster System Request For Proposals has been released. The values reported here are a composite of various industry designs. Until contract award, these values should be considered preliminary.

Table 1 - MAV System Deliverables.

Sub-element	Hardware	Subtotal [kg]	Totals [kg]
Payload Assembly			8.99
	OS	3.60	
	Payload Assm. Stage 2 Inerts	2.04	
	Payload Assm. Fairing	3.36	
Booster System			133.78
	Stage 2 Inerts and Propellant	17.85	
	Stage 1 Inerts and Propellant	110.40	
	Launcher Hardware	5.53	
Igloo (System Thermal Control)			11.70
			154.47
	Subtract OS and OS Support	4.60	
	MAV SYSTEM TOTAL →		**149.87**

The OS and OS support mass are subtracted from the total mass because the hardware is delivered by the Sample Transfer Chain.

Table 2 breaks the system into ascent phases. The left side gives the total mass for each mission phase while the right side gives the mass contribution of the relevant system component.

Table 2 - Mass of System States and Components.

System State	Stack mass [kg]	Comp. mass [kg]	System Component
OS Released	3.60	3.60	OS
Stage 2 Burnout	7.89	4.29	Stage 2 Inerts
Stage 2 Ignition Mass	19.89	12.00	Stage 2 Propellant
Stage 1 Burnout	59.64	39.76	Fairing & Stage 1 Inerts
Gross Lift-off Mass	133.64	74.00	Stage 1 Propellant
Pre-Ascent	150.87	17.23	Igloo & Launcher

Ascent Sequence

To prepare the MAV for launch, the Launcher releases MAV tie-downs and elevates it to 45 degrees with respect to the Lander deck. When ready, the MAV jettisons its thermal control Igloo and ignites its first stage.

The First stage uses a gimbaled nozzle to control its thrust vector during powered flight. After burnout, a cold gas system maintains attitude control until fairing separation. After fairing separation, the cold gas system reorients the vehicle to the direction that Stage 2 will fire and spins it up to 300 RPM.

Once the vehicle is spinning, Stage 1 performs its last functions by firing two time delay pyros on the second stage and separating itself. Seconds later, the first delay pyro ignites the second stage motor. After motor burnout, the OS is released by the second delay pyro.

Driving Interface Requirements

The MAV second stage has a fixed total impulse. Therefore, any mass knowledge error translates directly into a delta V error and hence a semi-major axis error. Currently, the Sample Transfer Chain is required to supply sample mass knowledge to within 50 grams. Providing a reliable measurement system or relaxing the mass knowledge requirement drives both sides of the interface.

The MAV Solid Rocket Motors must be stored above –40 C. To maintain the motors at or above –40 C with ambient temperatures as low as –135 C, an Igloo (thermal control system) and heaters are necessary which drive MAV mass. Heater energy requirements drive Lander solar array sizing, battery sizing and Landed scenarios.

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9. Sample Transfer Chain

The Sample Transfer Chain (STC) provides the path for sample return across the entire Mars Sample Return Mission (MSR). STC consists of the hardware that is necessary for the collection, storage, and safe return of the Martian samples to Earth. This includes the Lander Based Sampler (LBS), the Sample Tubes, the Orbiting Sample (OS), the Sample Capture and Transfer System (SCATS), and the Containment Vessel (CV). STC is also responsible for ensuring that planetary protection is not violated when the samples are returned to Earth.

The Lander Based Sampler is a deep driller (Deedri) provided by the Italian Space Agency (ASI). It is a four-degree of freedom arm with a drill box on the end of it that has the ability to collect samples about .5 meters below the surface. The LBS is a highly constrained system. It must provide the capability to collect a minimum of 325 grams of sample, store the individual sample cores in a series of sample tubes, cap them, and possibly even weigh the samples (under study). The LBS resides on the lander deck beside the Mars Ascent Vehicle (MAV) and in worst case conditions, it could have a 1.7 meter reach to the ground. Once it reaches the ground, the drill rod extends allowing collection of sample cores. The drill is designed to collect rock or soil samples, however, due to landing site restrictions as well as power restrictions, it will most likely collect samples that are a combination of soil and rock pebbles. This regolith is placed in the sample tubes, the tubes are capped, and then they are transferred into the OS inside the nose of the MAV (See Figure 6). Deedri also collects sample cores that are placed into the Additional Payloads for in situ science.

Like the LBS, the Rover uses the STC sample tubes to contain the rock and soil samples that it collects. The individual tubes as seen in Figure 7 are used to prevent bulk transfer of sample. They preserve the integrity of the individual samples by preventing them from mixing.

Once 500 grams of Martian sample have been acquired and placed inside the OS, the OS is sealed and sterilized to "break-the-chain-of-contact" with the Mars environment. The lid on the OS will have been opened a variety of times while on the surface of Mars to allow sample transfer. Once it is opened, the seals and the inside will be contaminated with dust from the Martian surface. However, the rest of the OS is sealed inside the MAV and never encounters the Martian atmosphere. In order to prevent transfer of the Martian contaminants on the OS into the Earth's atmosphere, the OS is sealed and the contaminated areas are sterilized. There are a few methods that can be utilized for this including pyrotechnic welding and chemical sterilization. Once the OS is sealed and sterilized, the MAV is launched. The MAV places the OS in orbit around Mars, where it waits for the '05 Orbiter to capture it.

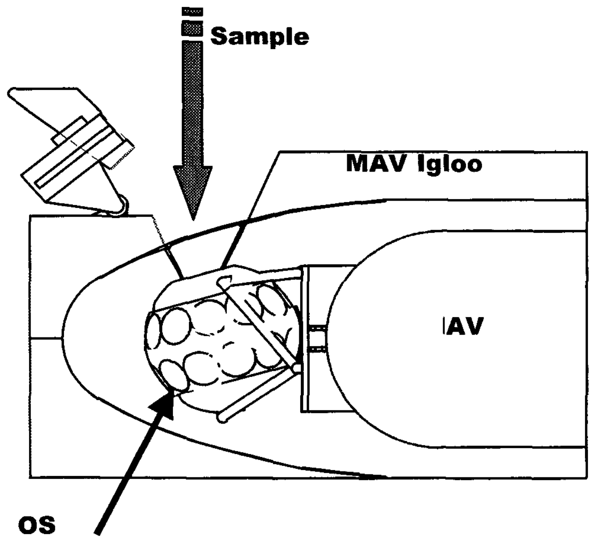

Figure 6 - OS inside the MAV

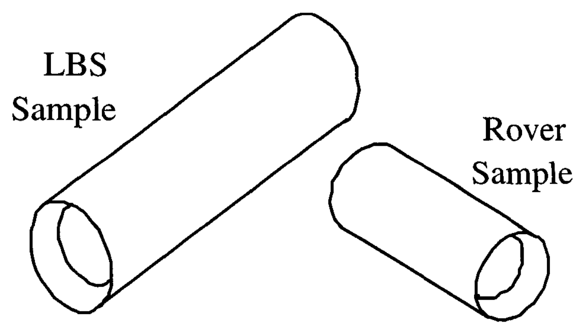

Figure 7 - Rover and LBS Sample Tubes

The OS is another very highly constrained system. As shown in Figure 8, it is made up of two basic pieces, the sample canister (SaC) and the power structure. This 3.6-kg sphere must survive in orbit around Mars for 6 years. The SaC contains the sample and is volumetrically constrained by the need to collect 500 grams. The power structure contains solar cells that provide power to a beacon that allows it to be found by the French orbiter as well as other orbiters at Mars (e.g. MGS, Mars Express). A maximum diameter of 16 cm prohibits the structure from increasing in size to accommodate more solar cells, so the maximum output of the solar array at end of life is 0.4 W. The beacon transmit frequency is 301.5 MHz, and the receive frequency is 437.1 MHz. Its range is 3000 km.

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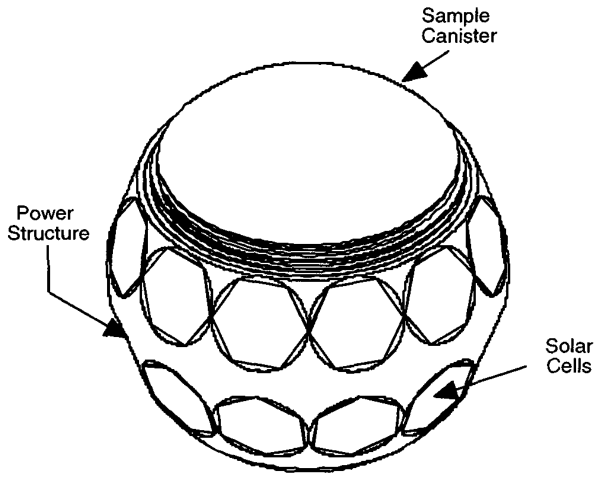

Figure 8 - Orbiting Sample

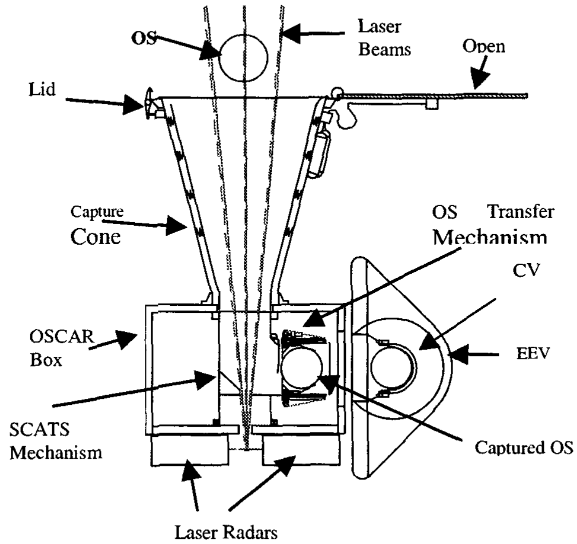

Figure 9 - Capture of the OS by the Orbiter

The power structure also has corner cubes around it so that the laser radar (LIDAR) on the orbiter can detect it. Once the orbiter arrives at Mars, it searches for the OS. The OS will have hopefully been located prior to the arrival of the orbiter by the other satellites orbiting the planet, although this is not required. Once the '05 orbiter has found the OS, the SCATS is used to capture it as shown in Figure 9. The LIDAR finds the OS and the orbiter maneuvers so that it is caught in the SCATS capture cone. A lid is then closed on the cone so that it will not bounce out. The lid pushes the OS into the throat which leads to the transfer mechanism.

Once the OS is in the transfer mechanism, it is placed in one of the Earth Entry Vehicles (EEV). Inside the EEV, it is encased in a Containment Vessel (CV) as shown in Figure 10. At this point, the OS is configured for return to Earth.

The containment vessel is designed to prevent the release of Martian samples into earth's atmosphere when the EEV lands. It is designed to survive any credible impact. If the OS breaks open, the EEV will contain the samples.

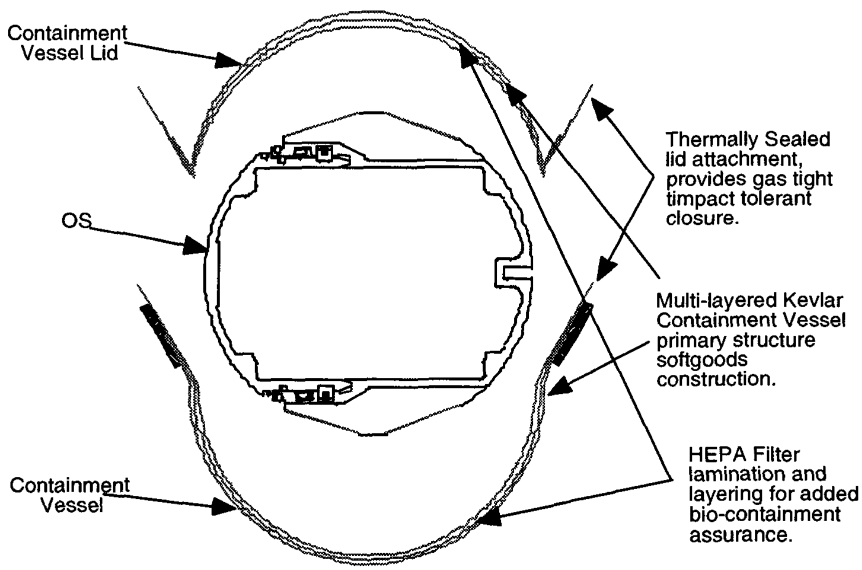

Figure 10 - OS/CV System

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10. ORBITER

Major requirements

The Orbiter for the 2005 Mars Sample Return mission has demanding requirements :

- launch mass : < 2700 kg, including the OSCAR main payload (NASA provided) and the four European Netlanders,
- heritage and simplicity when possible (design to cost),
- approximately 3250 m/s of ΔV, a figure significantly less then would be necessary if aerocapture techniques were not used for Mars Orbit Insertion,
- stringent planetary protection requirements,
- Single Fault tolerant in all mission phases, including critical autonomous phases.

Trades

The overarching trade is between the mass optimization (reduction) and the design to cost approach. The 2700 kg mass allocation is a real challenge because of the tremendous amount of propellant needed to perform the sample return mission. Some numbers to illustrate the level of difficulty : 2700 kg on the launch pad in 600 kg dry mass at the end of the mission, due to the fact that mass must be carried to Mars and then back to Earth, a dry mass increase of 1 kg can result in a launch mass increase of 3.3 kg to maintain the necessary propulsive capability.

Key trades are currently under analysis in the frame of the two parallel studies being conducted by two French aerospace companies. Although the trades and design are specific to each contractor, Centre National d'Etudes Spatiales (CNES) has defined recommendations and guidelines.

Staging: A two-stage configuration is better for mass and robustness of the mission. The cruise stage shall primarily carry the four Netlanders and provide solar power during the cruise to Mars and shall be jettisoned before the atmospheric entry for aerocapture. During the atmospheric pass, the second stage is protected by a heatshield, to be jettisoned after the insertion into mars orbit. This second stage shall support the rest of the mission.

Propulsion: A unified propulsion system for the two stages is preferable to a multiple staging (the cruise propulsive phase is not large, ~100 m/s of ΔV).

Aerocapture center of gravity: The location of the center of gravity is very constrained by the aerodynamic stability required during the aerocapture. It must be inside a box of roughly 100 mm x 20 mm (see Figure 11). This constraint is driving many of the trades for the overall configuration of the Orbiter.

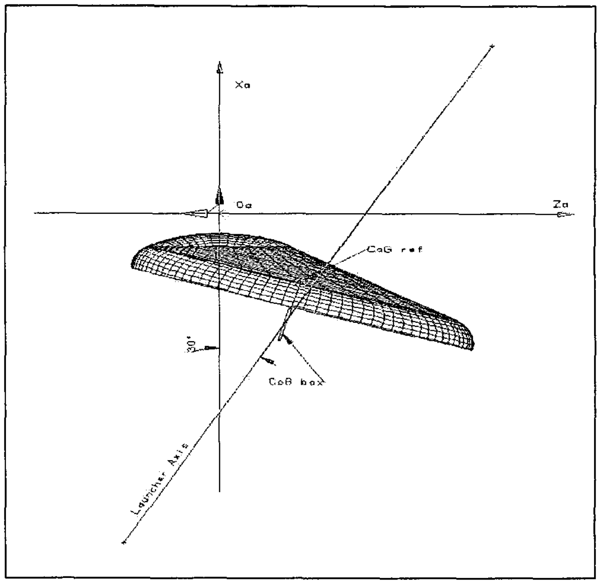

Figure 11 – Aerocapture Heat Shield

Attitude Control actuators: Preliminary analyses indicate a preference for reaction wheels because of the attitude stability required, the fuel consumption budget, and to enable operational flexibility.

Attitude trades for Direct To Earth communications gives the following results. During the cruise to Mars : Sun pointing (3-axis or spin) and a fixed MGA. In Mars Orbit: a) Nadir pointing during the sample search phase to minimize the gravity gradient torque, yaw steering, and a 1 degree of freedom (dof) HGA, b) Target pointing, roll steering and a 1 dof HGA, c) or inertial pointing and a 1 dof HGA, depending of the phase of the mission.

General configuration

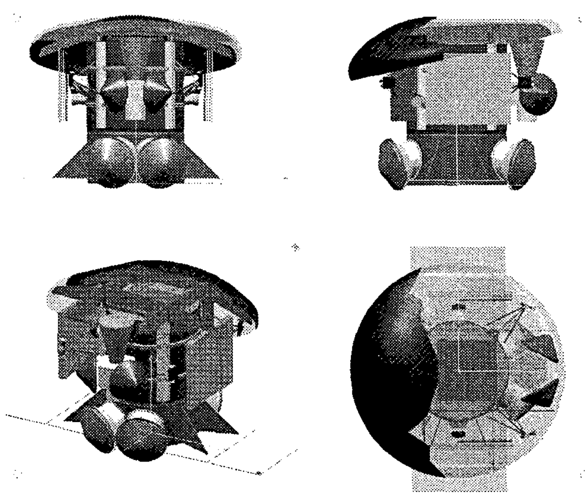
Figure 12 – Orbiter Configuration

Propulsion System

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The main features of the propulsion system are:
Thrusters: a main thruster (> 400N, Isp ~320s) and 16 to 20 Attitude Control Subsystem (ACS) thrusters (10N or 22N), grouped into 4 clusters (see figure 13).

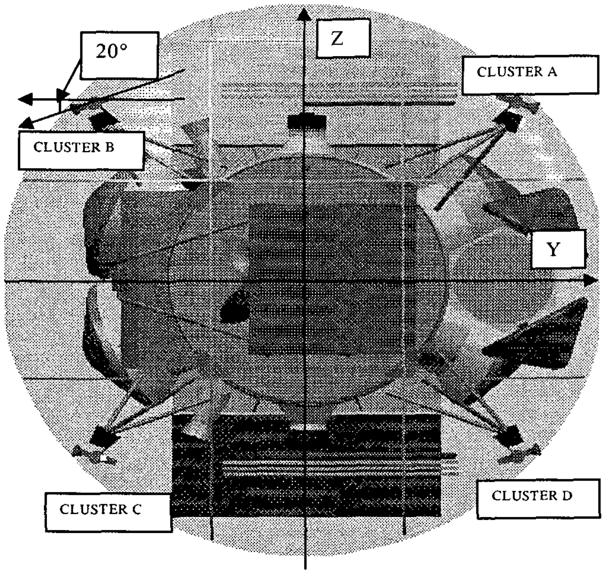

Figure 13 – Propulsion System

Bi-propellant under trade (MMH/NTO or Hydr/NTO)
2 or 4 tanks for 1400 kg of propellant

Communications

The sizing scenario for the downlink is as follows: a) 1 Mbit/day during cruise (3 passes/week using Deep Space Network (DSN) 34m antenna) , b) 200 Mbit/day during the orbital phase (played back to 34m DSN antenna during 25% of the orbit period), c) at least 10 bits/s during contingencies scenarios (using DSN 70m antenna). That leads to size roughly the RF power amplifier to 40W and the gain of the HGA to 30 dBi. In addition, a fixed MGA is used during the cruise to Mars, thanks to the SPE (Sun-Probe-Earth) angle, which remains < 40° for most of the phase, plus LGA's to improve the coverage in emergency cases. The X-band transponder is the JPL Furnished Small Deep Space Transponder.

Mass breakdown (approximately)

Structure, Thermal Control & Propulsion: 400 kg
Power, avionics, data handling & telecom: 260 kg
Heatshield: 250 kg
OSCAR Payload : 130 kg
Netlanders : 260 kg

Technology issues

A key technology issue is the capability of the propellant feeding device (in the tanks) to withstand 4 or 5 g during the launch phase and function under opposite acceleration (–2.5g) during the aerocapture. Studies are under progress to adapt existing tanks with membrane. Another key technology is autonomous guidance and control for aerocapture, especially given the uncertainties in Mars atmospheric density.

Computer & Data Handling

The current architecture is basically hybrid between a modular distributed architecture and a centralized ("star-shape") architecture. The main computer and the Oscar computer are JPL furnished items and are based on a Compact-PCI and RAD6K/PowerPC architecture; the communication bus is a MIL-STD-1553B bus (Figure 14).

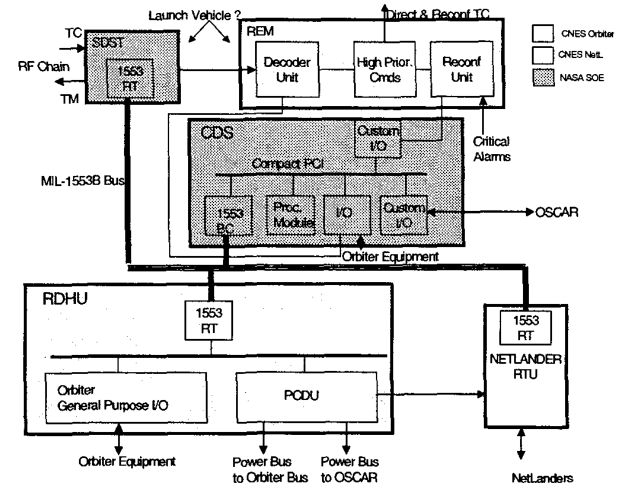

Figure 14 – C&DH Block Diagram

OSCAR Interfaces

The orbiter payload, also called OSCAR (Orbiting Sample Capture and Return), consists of tracking and rendezvous sensors, rendezvous software, capture mechanisms, and the Earth Entry Vehicle (EEV). OSCAR will be provided to CNES by JPL for integration onto the orbiter. The major OSCAR interface is the Rendezvous NGP (Navigation, Guidance & Piloting) closed-loop. OSCAR is responsible for the relative navigation (LIDAR sensor) and the guidance; the Orbiter bus is responsible of the absolute navigation (star tracker, IMU) and of the piloting. Practically, the interface is simple and clean, translation (ΔV's) and attitude (quaternion in Local Vertical/Local Horizontal (LVLH) frame) being the guidance parameters. For the realization of translations, the current baseline is to control them in closed-loop with accelerometers.

The Fault Protection is hierarchical as follows:

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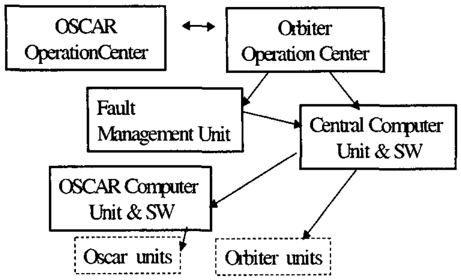

Aerocapture, a key challenge

Aerocapture has been selected as the technique for Mars orbit insertion. By modulating the bank angle (predictor-corrector algorithm in closed loop with accelerometers), a speed reduction greater than 2000 m/s is obtained. With the FPA (Flight Path Angle) in a entry corridor of –10.970° to –10.137°, the target orbit is 1400 km x 250 km (after an apoapsis burn) with an inclination of 45°. The selected shape of the heatshield (thermal flux of 400 to 500 kW/m^2) comes from the AFE (NASA Autonomous Flight Experiment) and provides a Length/Diameter (L/D) between 0.25 and 0.3 (for an angle of attack between –2° and +2°). A typical deceleration profile is given in Figure 15.

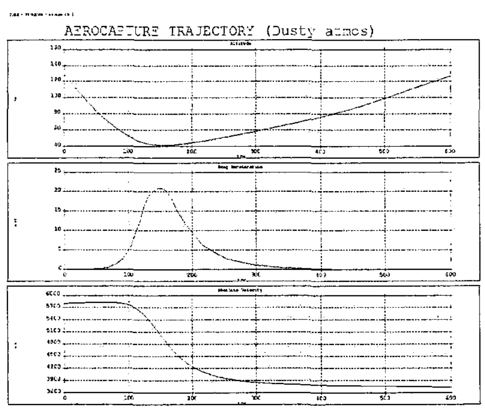

Figure.15 - Typical Aerocapture Deceleration

11. NetLanders

Scientific objectives

Mars exploration provides a unique opportunity to understand the formation and evolution of a planet similar to the Earth. This will be the major objective of the NetLander project, which is led by CNES in cooperation with many European institutes.

NetLander will bring new insights into the knowledge of Mars from the deep structure of the planet to its atmosphere and ionosphere. The mission is mainly a network mission, with some additional "multi-site" experiments. The so-called "network" experiments are those, which require identical instruments acquiring synchronized measurements at different locations on the planet. For example, seismology triangulation relies on seismological signals being detected by three stations in order to locate the seismic event. "Multi-site" experiments could be performed on one lander only, but still benefit from being performed at different places on Mars, because of the diversity of the planet.

Network science will answer two fundamental questions about Mars: What is the internal structure of the planet? What are the processes involved in the evolution of its atmosphere? A minimum number of three operational stations are needed to answer the first question: they will allow to locate seismic events, determine the direction and amplitude of the rotational axis of the planet and measure the horizontal gradient of temporal variations in its magnetic field. Four stations are therefore necessary to achieve a sufficient degree of reliability. The fourth station also allows to address additional scientific objectives: located at the antipode of the network, it may detect waves (already localized by the three other stations) transmitted through the core, thus providing an estimate of the core size. For meteorological studies, the main priority is to observe the same phenomenon simultaneously at different locations. It is true that with only four stations measuring atmospheric parameters, the network is far from providing a global coverage of the planet. However, only measurements from individual landers have been performed so far, and NetLander will allow a significant step forward in the knowledge of the atmosphere of the planet.

In addition to the network objectives, multi-site experiments will give more information about the local environment of the landers. Subsurface sounding will look for the presence of water (liquid or solid) underneath the lander. Radio links between the lander and the orbiter will be used to determine the Total Electronic Content (TEC) of the ionosphere above the lander. The on-board camera will provide additional geological and meteorological parameters.

Payload description

The NetLander reference payload comprises several instruments which will work together in order to answer the NetLander scientific objectives.

Each lander has one two-axis Very Broad Band (VBB) seismometer, one horizontal micro-sensor completing the two VBB axes, and one three-axis short period seismometer. The VBB is characterized by its very low noise and high sensitivity. Its long period performances allow the detection of the tides produced by the Sun or Phobos.

The atmospheric package (ATMIS) is composed of several sensors deployed along a boom. Temperature, pressure, humidity, optical depth, wind direction and velocity are measured. The ATMIS boom also carries the Electric Field Sensor (ELF).

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The tri-axial flux gate magnetometer (MAG) operates in the DC – 10 Hz frequency band.

NetLander Ionosphere and Geodesy Experiment (NEIGE) uses the NetLander telecommunication system for very accurate Doppler measurements (accuracy about 0.1 mm/s). In order to achieve this goal, some specific functions (e.g. up-link S-band carrier) have to be added to the UHF telecommunication system.

The Panoramic Camera (PanCam) provides panoramic, stereoscopic and multi-spectral imaging. The camera head is mounted on a boom and deployed about 1 m above the surface.

The Ground Penetrating Radar (GPR) sounds the ground at a frequency about 2 MHz, in order to achieve a satisfactory compromise between penetration (up to 2.5 km) and resolution (50-100 m).

Landing sites

The choice of landing sites will result from a compromise between scientific objectives and technical constraints.

The scientific requirements are primarily driven by the objectives of network science, which are expressed in terms of network shape, latitudinal and longitudinal coverage, distances between the stations. The best configuration includes 3 landing sites having a minimum separation of 30° between 2 sites, and the 4th station near the antipode of the triangle formed by the first three stations.

The selection of sites is constrained by the mission scenario, with one orbiter carrying all 4 landers to Mars. Other constraints come from the lander design: landing site elevation is limited (+ 2 km) to allow sufficient deceleration and safe landing, with reasonable parachute size. Energy requirements and thermal issues limit the latitude: preliminary estimates resulted in choosing latitudes in the range: - 40°, + 40°.

NetLander design

Each NetLander probe is attached to the orbiter by a Spin-up and Eject Device (SED). Its goal is to provide the linear velocity necessary to separate the probe and the spin rate, which will help stabilize the probe during the entry phase.

The lander comprises two main sub-assemblies:
- the Surface Module,
- the Entry, Descent and Landing System (EDLS).

The EDLS protects the Surface Module during all mission phases until its deployment on the surface of Mars (Figure 16). In particular, it is designed to withstand the thermo-mechanical loads during the atmospheric entry phase and at landing. The impact shock on Mars must be limited to 200 g / 20 ms.

The atmospheric phase begins when the atmosphere is detected by accelerometer measurements. During the ballistic entry phase, the heat shield reduces the velocity of the probe, and protects the lander against high thermal fluxes.

The parachute system is activated when the probe velocity is low enough to allow parachute deployment. These conditions have to be obtained at high enough altitudes to maximize the efficiency of the parachute phase. At Mach 1.5, the pilot chute can be opened. The main parachute is deployed at sub-sonic velocity (Mach 0.8).

Because of the low atmospheric density on Mars, the efficiency of the parachute system is limited: an additional landing system is necessary to reduce the landing shock. This landing system will likely consist of inflatable balloons around the Surface Module.

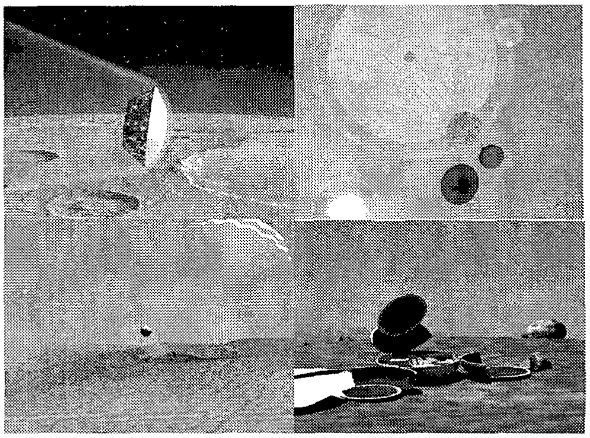

Figure 16 - Entry, Descent, Landing and Deployment

The total mass of the NetLander probe is 50-60 kg when it enters the Martian atmosphere. After landing and ejection of all EDLS elements, the mass of the Surface Module is around 20 kg, of which 4-5 kg are allowed for scientific instruments.

After landing, the Surface Module determines its orientation. Opening the main petal turns it into its upright position, if necessary. After reaching a stable position (Figure 17), the Surface Module deploys its antenna and the booms carrying the panoramic camera, the ATMIS package, and the magnetometer. The seismometer will also be mechanically decoupled from the primary structure by releasing the instrument from its mounting point and letting it fall on the Martian surface with a cable connecting it to the Surface Module.

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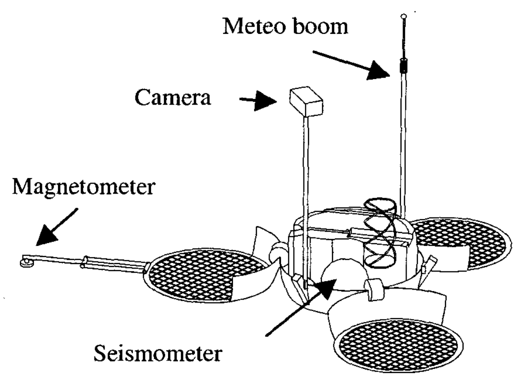

Figure 17 - Deployed Surface Module

Solar panels serve as the main energy source. The primary battery is reserved for supplying the required power for the NetLander during descent, landing, and initialization phases on the Martian surface. The secondary battery is used as energy storage for NetLander nighttime operations on the Martian surface. The solar arrays are accommodated on the inner surface of the three petals, having a total surface area of roughly 1 m^2.

All system electronics are accommodated together with the Thermal Control Subsystem in one common Electronics Box, which is surrounded by thermal insulation. The inside of the Electronics Box will be kept at temperatures between + 50 and –50 °C by Radioisotope Heater Units (RHU) and a controllable Heat Rejection System. The use of RHUs promotes effective operations and facilitates survival in case of a global dust storm.

The estimated energy demand by science instruments and the payload service electronics is 20 to 100 Wh/sol. In the beginning of the mission, energy demand is higher due to more intensive measurement operations. The power subsystem will be scaled to meet energy demands also at the end of the mission (one Martian year).

12. Orbiting Sample Capture and Return

The key components of the Orbiter Payload are the Orbiting Sample Capture and Return System (OSCAR) and the Supplied Orbiter Equipment (SOE); these components and their relationship to the orbiter are outlined in Figure 18.

MORS includes the guidance software and sensors to locate, track, and rendezvous with the Orbiting Sample (OS). The sensors include: 1) the Radio Direction Finder (RDF), which can locate the OS at a maximum range of 3000 km; the Light Detection Radar (LIDAR), which will determine range and bearing measurements to the OS at a maximum distance of at least 5 km; and an observational camera, which is not used for rendezvous guidance but for taking images of the capture event. SCATS includes the capture cone and mechanisms to transfer the OS from the capture cone into the EEVs. The EEV takes the OS safely from the orbiter to the surface of the Earth.

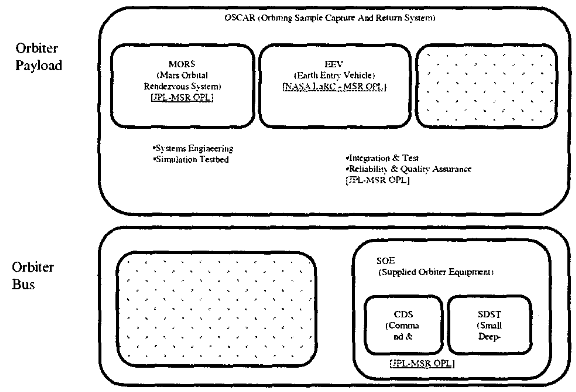

Figure 18 - OSCAR Components

Driving Requirements

The orbiter payload, also called OSCAR, consists of tracking and rendezvous sensors, rendezvous software, capture mechanisms, and the two Earth Entry Vehicles (EEVs). OSCAR will be provided to CNES by JPL for integration onto the orbiter. The primary functions of OSCAR are to:

- Provide instrumentation and software to find and track the two OS
- Send maneuver requests to the orbiter during the terminal rendezvous phase in order to autonomously rendezvous with each OS
- Capture the two OS
- Transfer and latch both OS into the Earth Entry Vehicles (EEVs)
- Seal the EEVs
- Jettison unnecessary equipment prior to leaving Mars (to reduce departure propellant usage on the orbiter system)
- Spin-up and release the EEVs for Earth entry
- Ensure safe delivery of two OS to the surface of the Earth

Key Trade Studies

Key trade studies to date have included:

- Whether or not to fly a search camera as a backup to the radio direction finder to look for the OS at long ranges (decision: no search camera)
- 1 vs. 2 EEVs, including impacts to SCATS and OSCAR for these options (decision: 2 EEVs)
- Whether or not to have a separate processor for the OSCAR guidance during rendezvous (decision:

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OSCAR has a separate processor)

- Payload Electronics Box: Is it more mass efficient to have two PEBs and jettison one at Mars vs. one PEB which is not jettisoned and can fire the EEV separations pyros (decision: two PEBs is more mass efficient)
- Configuration trades: capture cone and mechanisms, OSCAR general configuration,
- Ratio of jettison mass to returned mass

System Design

The configuration and system block diagram are shown in Figures 19 and 20, and the mass list in Table 3.

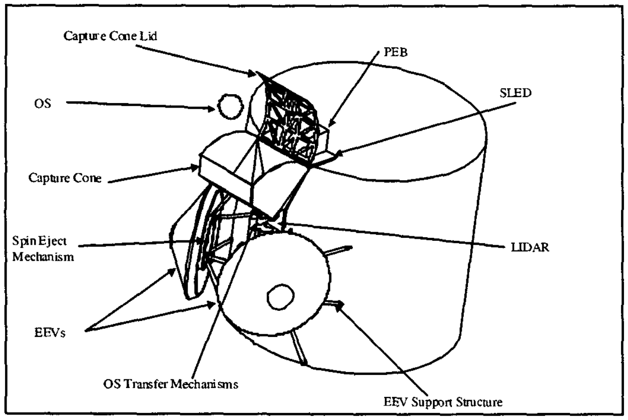

Figure 19 - OSCAR Configuration Drawing

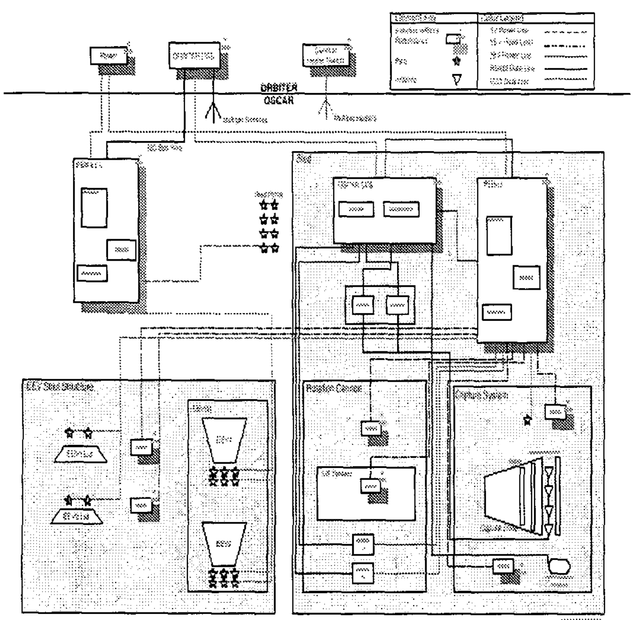

Figure 20 - OSCAR System Block Diagram

Table 3. OSCAR Mass List

OSCAR Mass Profile										
		Launch Mass				Jettisoned Mass	Returned Mass	Approach Mass	EOL
Subsystem	Hardware Items	Mass w/o contin-gency (kg)	Contin-gency (%)	Mass w/ Contin-gency (kg)	Allocation (kg)	Mass w/ Contingency (kg)	Mass w/ Contingency (kg)	Mass w/ Contingency (kg)	Mass w/ Contingency (kg)
Struct/ Mech	OSCAR	14.52	28.81	18.70	TBD	0.00	18.70	0.00	18.70
SCATS	Capture System	18.23	24.25	22.65	20.70	18.44	4.22	0.00	4.22
Power/ Pyro	PEB-O, Micro-PEB, Pyro System	7.83	30.00	10.18	TBD	7.59	2.59	0.00	2.59
Thermal		3.50	30.00	4.55	TBD	2.65	1.90	0.00	1.90
CDH	MORS Processor	4.10	20.00	4.92	TBD	4.92	0.00	0.00	0.00
Observation Camera		1.00	30.00	1.30	TBD	1.30	0.00	0.00	0.00
RDF		2.15	30.00	2.80	TBD	2.80	0.00	0.00	0.00
LIDAR		8.00	30.00	10.40	TBD	10.40	0.00	0.00	0.00
EEV		38.72	25.00	48.40	TBD	0.00	48.40	48.40	0.00
OS	Samples	0.00	0.00	0.00	TBD	0.00	7.20	7.20	0.00
Syst. Cont.	System Contingency	98.05	10.00	9.81	TBD	4.90	4.90	2.45	2.45
TOTAL				133.71		53.00	87.91	58.05	29.86

Technology Drivers and Heritage

The only key technology driver is the LIDAR. The other components have heritage from other flight projects or applications.

Key Interfaces with the other Flight Elements

OSCAR's key interfaces is with the Orbiter (mechanical, electrical, software, data storage and rate, data interfaces). OSCAR has key interfaces with the Sample Transfer Chain on the following three items: SCATS, the containment vessel which envelopes the OS in the EEV as redundant layer of containment assurance, and the OS (beacons, mass, size, and shape).

Key Challenges

Key challenges for OSCAR include MORS software development with interfaces to the orbiter software, the EEV development and test program, and the LIDAR development.

Reliability and redundancy issues

Most of OSCAR is either block or functionally redundant (except for primary structure and other TBD waivers). One potential single point failure is the capture cone – if the first OS got stuck in the cone, there is currently no way to get the second OS into the EEV. A trade study will be done to see if the additional mass to avoid this situation is tolerable.

13. EARTH ENTRY VEHICLES

The two Earth Entry Vehicles (EEVs) transport the Orbiting Samples (OS) through Earth's atmosphere and deliver them safely to a recoverable location on the surface. During this entry, descent, and landing, each passive EEV dissipates 1.9 giga-joules of kinetic energy while limiting the mechanical and thermal loads experienced by the OS container. Limiting mechanical and thermal loads on the OS preserves the sample's integrity and prevents loss of sample containment. The atmospheric flight of the EEVs is the final flight phase of the Mars Sample Return Mission.

EEV Requirements

The driving requirement on the Earth-entry capsule is to assure containment of the Mars samples during the intense Earth entry, descent, and landing phases of the mission. The design must also provide for easy sample recovery by

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avoiding a water landing and including ground recovery beacons. Vehicle mass at launch must be no greater than 24.2 *kg* each. The maximum dimension of each EEV is 1.0 *m*.

Key Trades

Delivery of the Mars Samples to Earth may include a direct entry to the surface or an Earth orbit insertion. The mass requirements and complexities of an Earth orbit insertion, which require a velocity change of 3630 m/s, an Earth orbit rendezvous, and eventual Earth atmospheric entry, increases risk over the simpler direct entry approach. Earth orbit insertion involves a factor of two to ten more critical events than direct entry. For these reasons, a direct entry approach is preferred. Successful direct entry at comparable energies was accomplished 30 years ago with the manned Apollo missions.

The EEV trade that received the most study surrounds the use of, or exclusion of, a parachute terminal descent system. A parachute system decreases ground impact speeds that may increase system reliability. Unfortunately parachutes and the associated deployment system, while highly reliable, do not possess the incredible reliability necessary to meet containment assurance requirements. If a parachute terminal descent system is included within the EEV, the vehicle must still be designed to survive the ensuing ground impact in the event of parachute failure. Packaging both a ground impact energy absorption system and a parachute system increases the ballistic coefficient of the EEV's that increases risk of heatshield failure. Additional risk is introduced with respect to inadvertent deployment of the parachute. A premature parachute deployment removes sections of the capsules thermal protection system. Finally, there is no mission need for a parachute system. Sample return missions carrying fragile samples may require a parachute, however, sample integrity tests on representative materials have demonstrated that the impact loads associated with a non-parachute EEV should not degrade the scientific value of the samples. In summary, the simpler direct to ground impact EEV design appears to possess higher reliability by virtue of its simplicity.

Return of two sample-containing OS introduces the trade between placing both OS in a single EEV or having two separate EEV's. There is a small mass benefit of placing both OS in one EEV primarily from reduced OSCAR mounting hardware. However, the smaller size and ballistic coefficient associated with each of the two EEV's reduces risk on the thermal protection system, decreases ground impact speed, and eliminates the potential for OS to OS interaction during the ground impact event. Two EEV's also provides additional mission resiliency.

Selection of the thermal protection system to meet the stringent containment assurance requirements includes a trade between low density, high performance, developmental materials and high density materials with flight heritage. Selecting a high density ablator such as fully-dense carbon phenolic as the primary heatshield may not reduce overall system risk. Carbon-phenolic's inefficient performance in this flight regime requires a large mass Thermal Protection System (TPS) that increases entry and impact loading.

Water impact is more benign than ground impact. However, the possibility of inclement weather, rough sea conditions, and sinking introduces substantial risk towards recovery of the capsules. Loss of either EEV not only represents loss of mission science but also loss of sample containment.

EEV Design

A representative EEV design is shown in Figure 21.

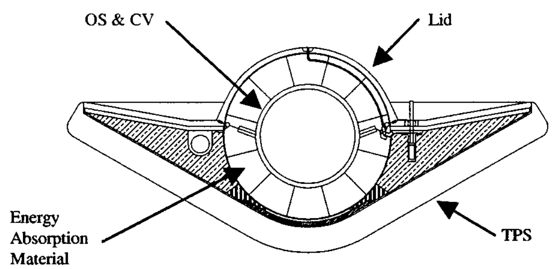

Figure 21 - One possible design of the Earth Entry vehicle with diameter = 0.8 m.

Each EEV includes a Containment Vessel (CV). The CV is an additional containment layer that is sealed after OS transfer. This additional containment layer is then encased in energy absorbing material comprised of a spherical cellular structure. The cellular structure is a radially oriented honeycomb structure with carbon foam filled voids whose energy absorption characteristics are tuned to a desired crush strength by variations of the composition of graphite and Kevlar in the inter-cell webs. The energy absorption performance of this structure prevents deforming loads from reaching the OS/CV containers even in the unlikely event that the entry trajectory leads to an impact with a rigid surface.

In the nose region of each EEV, the spherical cellular structure is packaged behind a TPS. Trade studies have not been completed on selection of the optimal TPS system. One possibility is a multi-layered TPS. The first layer outside the cellular structure is a carbon fiberform insulation layer followed by a carbon phenolic nosecap. Both of these layers mate to a carbon-carbon heatshield support structure upon which is bonded a low density heatshield such as Phenolic Impregnated Carbon ablator (PICA-15). The afterbody is also protected by a low density ablator such as Silicon Impregnated Reusable Ceramic Ablator (SIRCA-15F). The multi-layer thermal protection system makes use of materials utilized in the Galileo entry vehicle as well as the Stardust

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and Genesis sample return capsules.

The external shape of the vehicle is an axisymmetric 60-degree half-angle blunted cone forebody and a partially concave afterbody. The combination of this shape and the associated center-of-gravity location assures passive attitude control throughout all speeds regimes. The shape has the ability to reorient to a forward facing attitude if orbiter or spin eject failure lead to a backwards orientation of the vehicle at atmospheric interface.

The afterbody of each vehicle includes a removable lid for acceptance of an OS into the CV. The 3-point latch system is capable of retaining the lid during entry despite failure of any single latch.

The interior of the vehicle (outside of the spherical impact sphere) is fitted with carbon foam as a structural support element and additional energy absorber. Cut-outs within these foam sections house the two ground recovery beacon subassemblies.

Sensors indicate successful placement of the OS, placement of the lid, and engagement of the lid latches. Each EEV houses its OS during Earth-return cruise and, in conjunction with OSCAR thermal control, manages the environment of the samples during this flight phase.

Upon arrival at Earth, each EEV is positioned and then spin ejected separately from the Orbiter 1-4 days prior to atmospheric interface. During the ensuing exoatmospheric cruise, the spin stabilized EEV maintains an inertially fixed attitude and passively manages the thermal environment of the samples. The ground recovery beacons (two per vehicle) are activated prior to separation and operate for at least 24 hours after landing.

Reliability

Probabilistic Risk assessment is employed in the design trades studies and reliability determination of the EEV. The containment assurance requirements on the EEV necessitate that the design possesses greater reliability than previous entry systems and that its reliability be accurately quantified.

14. Key Issues and Conclusions

The biggest challenges to Mars Sample Return mission are:

- ensuring mission success
- mass
- cost
- Planetary Protection
- autonomous aerocapture
- autonomous rendezvous
- sample transfer mechanization and reliability

The MSR design development is not yet complete, but this current baseline represents our best effort to meet the above challenges with adequate margins. In response to the loss of the Mars Polar Lander mission, portions of this MSR architecture are being reevaluated, and the MSR Team is continuing to work to mitigate the above risks and successfully fulfill this historic mission.

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