DESIGN OF A SMALL, LOW COST MARTIAN LANDING DEVICE APPLIED TO SCIENTIFIC SURFACE EXPLORATION OF PLANET MARS

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Abstract

In the context of a planned low cost mission to the planet Mars (via Ariane 5 ASAP with a maximum spacecraft mass of 360 kg) that utilizes the orbiter as the primary space vehicle, an additional small landing device (lander) is foreseen to carry out a supplementary scientific investigation of the Martian surface. This landing device is the focus of this presentation.

In particular, based on the practical experience of the Babakin Space Centre, a cost effective landing device has been developed that is based on components which already exist. The atmospheric entry module has a mass of 16 kg, which includes the mass (4 kg) of the surface module (Mars station). The available mass hence for the Entry-, Descent- und Landing phase (EDL) subsystem is 12 kg. An aerodynamic heat shield and thereafter a parachute is used to slow down the entry module from approximately 5,000 m/s to approximately 25 m/s, after which the landing of the surface module is aided by a protecting airbag system which dampens the landing (soft landing) so as to avoid any damage to the surface module.

After the surface module has been correctly orientated so that its operation can begin, it may fulfil its purpose for carrying out scientific research of the lower Martian atmosphere. Due to limited space and available energy supply, the type and amount of scientific equipment within the surface module has to be restricted.

1. Concept for the Micro-Mars Mission

In the context of various international missions for research of the planet Mars, the planned Micro-Mars Mission has an important role to obtain scientific results by way of a low cost lander mission. Consequently, in 2002 a phase-A study, financed by the DLR-, EADS- und partner institutes, was carried out to evaluate the feasibility of the mission, including launch and operating costs, subject to a budget limit of 50 Mio Euro [1].

The following collaborators were involved:
- DLR, Berlin-Adlershof
- EADS Space Transportation, Bremen
- University of the Bundeswehr, München
- Technical University, München
- Technical University, Braunschweig
- University of Applied Science, Bremen
- EPFL, Lausanne, Switzerland
- Babakin Space Centre, Moscow, Russia

The very ambitious design-to-cost requirement means that it is necessary to use existing subsystems and intelligently integrate them in the development of the overall system. Fortunately the outcome of the phase-A study has shown that it is technically and financially feasible to pursue the Micro-Mars Mission.

The Micro-Mars spacecraft consists of an orbiter with 15 kg of equipment and the 16 kg Martian entry module. The orbiter consists of a robust central tube structure and is designed with a multiple mission concept in mind to reduce cost expenditure thanks to minor modifications that would be required for the development of alternative planetary vehicles. However, the lander is specifically designed for a Mars mission to cater for typical forces and temperatures during the atmospheric entry as well as Martian surface conditions.

The planned 15 kg on the orbiter available for experiments will make use of the following equipment:
- High resolution camera
- Magnetometer
- Radio science experiment
- Radiation experiment (Dosimeter)

The high-resolution camera is the primary payload of the orbiter. The camera experiment will provide a detailed scan the Martian surface, which will also help to fill voids in scanned data from previous Mars missions. Such data is very useful for the planning of future Mars missions where an improved understanding of the geophysical landscape can be obtained through analysis of the images. The camera system is an ongoing development of the DLR Institute for Planetary Research and has the highest resolution worldwide of up to 0.7 m per pixel. These images of the orbiter will be complemented by detailed measurements of magnetic field, gravitational field and surrounding radiation of Mars.

The Micro-Mars spacecraft (orbiter und lander) will be launched in 2007 or during a later start window as a piggyback payload (ASAP) on the Ariane 5 into a GTO, with a mass of 360 kg that includes fuel accounting for roughly 57% of S/C mass, and which will enable a delta-V manoeuvre of up to 1,600 m/s [2]. After separation from the Ariane 5 upper stage, an independent propulsion system will carry out the necessary trajectory manoeuvres during the 10 month journey to Mars and the 2 year in orbit operation phase (course corrections and manoeuvres for aligning the camera as well the antenna for communication with Earth). A bi-propellant propulsion system with 205 kg of fuel (MMH + MOH) is used to source four 22 N and four 10 N engines in order to provide control for orientation and position. For orientation sensoring, three reaction wheels, a gyro sensor, a sun sensor and two star sensors are available. Communication between the space vehicle and earth is achieved via S-Band (2.1 GHz) and X-Band (8.48 MHz), which are standard frequencies for missions to Mars. The orbiter and lander communicate via the standard UHF-Band (430 MHz). In so doing Micro-Mars may also play a role as a backup communications S/C for other Mars missions, for example to transfer data from another surface module in the event of its data relay spacecraft malfunctioning [1].

The Micro-Mars orbiter will be put into a highly elliptical orbit around Mars with a periapsis altitude of approximately 200 km and an apoapsis altitude of 30,000 km above the Martian surface. Such an orbit is useful for scientific experiments carrying out measurements over various altitudes where a single orbit will occur every Martian day (24.5 earth hours). Furthermore, the high inclination of the orbit will allow a daily overflight of the poles of the planet which is also of scientific benefit.

At the start of the operation phase at Mars the orbiter and lander are still mechanically connected to one another. A special impulse manoeuvre is carried out near the apoapsis in order to dispatch the lander into a desired descent trajectory towards the Martian surface. At this point the lander is separated from the orbiter, after which the orbiter carries out another impulse manoeuvre to return it back to its original orbit, whilst the lander continues on its descent trajectory and automatically enters the EDL phase.

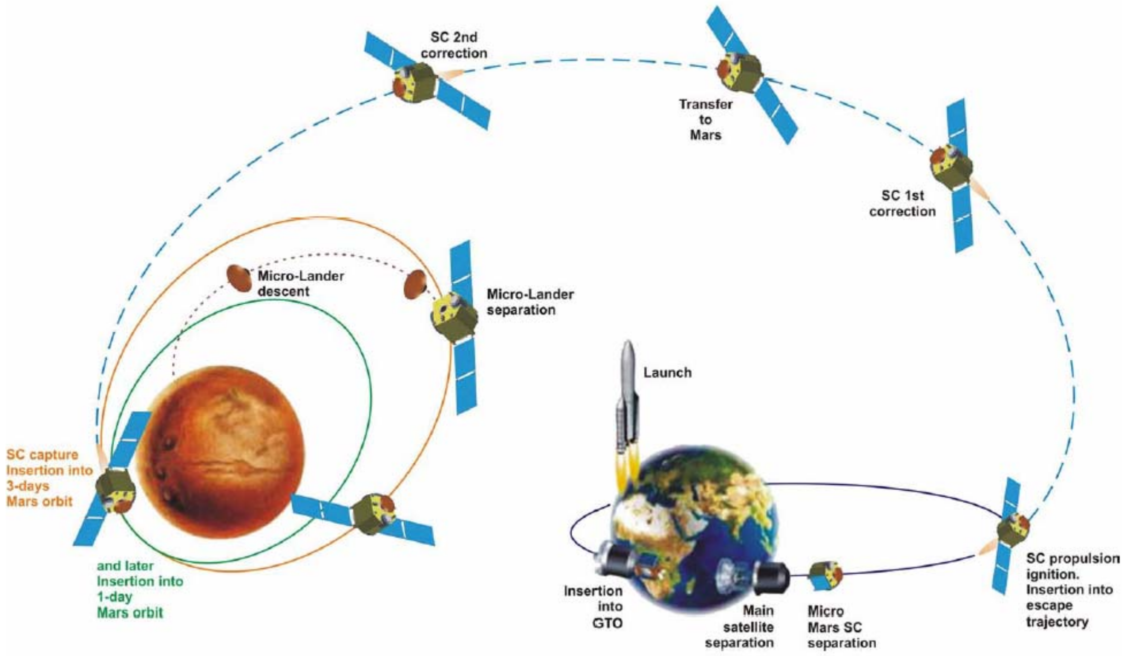


Figure 1: Various phases of the Micro-Mars Mission

Figure 2 shows the Micro-Mars S/C configuration during its journey to Mars. The lander is still attached to the orbiter and is seen by the 0.7 m disc segment characteristic of the aerodynamic heat shield. The solar arrays of the orbiter have a span of 5.5 m and an effective area of 4 m² when extended, which will on average provide 310 Watts of power during the sun phase of its Mars orbit.

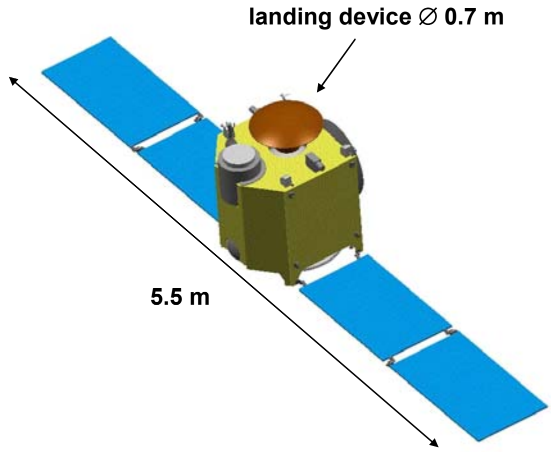


Figure 2: The Micro-Mars S/C including the lander and extended solar arrays

2. Micro-Mars lander

The 16 kg Micro-Mars lander comprises the following components:

Subsystem	Mass, kg
1. Surface module	4.00
2. Aerodynamic shield	2.60
3. Parachute system	3.66
4. Air bags	3.00
5. Fall-line	0.82
6. Shadow part of thermal protection	0.50
7. Back cone	0.50
8. Pyrotechnic devices	0.62
9. On-board cable network	0.30
Total mass	**16.00**

Table 1: Mass tabulation of the lander subsystems

The EDL subsystems comprise 12 kg of the total lander mass. For the surface module – also referred to as the Mars station – 4 kg remain. This mass is the driving factor behind the development of the station, which has a diameter of ∅ 25 cm. With reference to the principle technologies available for vehicle entry, a comparison was made between the classical heat shield technique, based on a hard, aerodynamic heat shield and the so-called Inflatable, Re-entry and Descent Technology (IRDT), based on an expanding heat shield. At present the former, EDL technique is preferred, primarily due to the reliability experienced during other Mars missions and the lower complexity during the EDL phase.

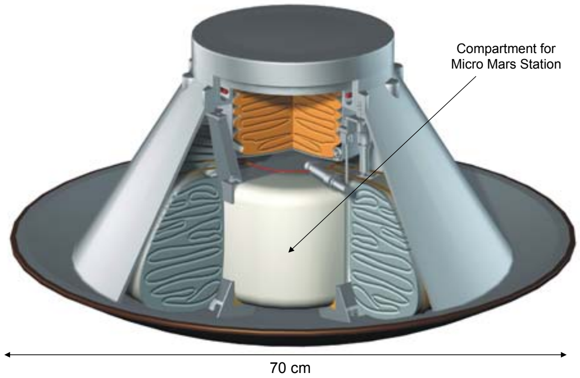


Figure 3: The Micro-Mars lander

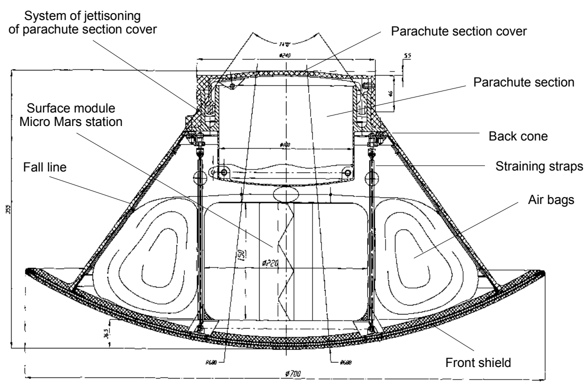


Figure 4: Cross-section of the lander with subsystems (see Table 1)

Figure 5: Model of the Micro-Mars lander (scale 1:1). The Mars station can clearly be seen in the centre

3. Entry, Descent und Landing (EDL)

The Entry, Descent and Landing mission phase renders the Micro-Mars lander without any supporting rocket propulsion. Via the delta-V manoeuvre as described in Section 1, the atmospheric descent trajectory is initiated, for which the following initial conditions are valid for the EDL phase calculations, based on the Mars-GRAM 2001 atmospheric model [3]:

- Entry altitude h: 120 km
- Entry velocity v: 5,000 m/s
- Entry angle: -11°
- Axial rotation: 0.5 rpm
- Mass m: 16 kg
- Effective heat shield area A: 0.385 m²

At an altitude of h=120 km, the effect of the atmospheric pressure starts to become apparent – aero braking is hence effective. The spin of the lander brought about during its mechanical link with the orbiter remains at 0.5 rpm and contributes to the axial stability of the lander. An acceleration sensor uses the atmospheric forces to determine the altitude and hence controls the pyrotechnic devices that are responsible for activating and separating the necessary subsystems during the EDL phase (see Table 2). Furthermore, the sensor is also used as an experimental instrument as it reports its measurements during the various EDL phases via a UHF link to the orbiter. This is used as a verification of the current atmospheric model for Mars (e.g. Mars-GRAM 2001).

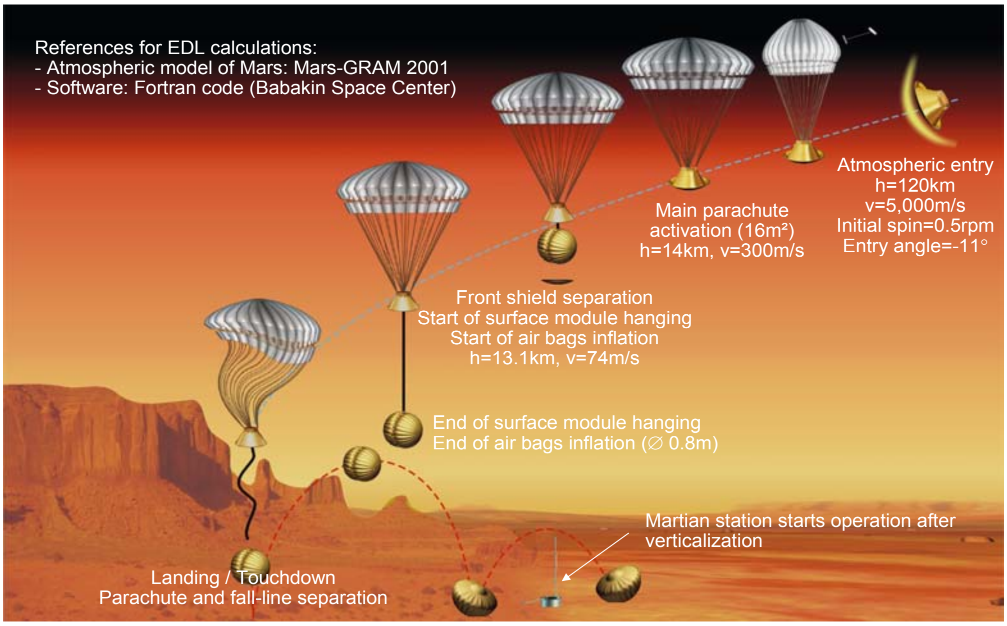


Figure 6: Graphical depiction of the various EDL phases

Parameters of EDL phase (parachute concept) v_{entry} = 5,000 m/s, θ_{entry} = -11 ±1.0°, m = 16.0 kg

Phase of descent	Time, s	Altitude, km	Velocity, m/s	Trajectory angle, deg	Dynamic pressure, Pa	Mach number	Note
Entry into atmosphere	0	120	5,000	-11.0 -10.00…-12.00	0	23.35	Nominal and deviation of parameter are give
Maximal g-load	117.5 101.5...140.5	35.0 32.5...38.6	3,165 3,155...3,382	-6.84 -5.01...-8.4	2,128 1,598...2,623	16.24 16.20...17.36	Nx_{max}=-8.09 -6.28...-9.61
Separation of the parachute container cover, PS deployment	219.9 192.6...262.5	14.1 11.4...16.6	299.9 280.9...318.7	-34.22 -32.93...-36.26	179 165...195	1.37 1.27...1.46	Nx = -0.62 ± 0.02 at the descending branch of Nx
Main parachute unreefing	226.9 199.6...269.5	13.3 10.6...15.8	117.7 107.2…128.6	-40.7 -39.33...-42.58	29.8 26.4...33.8	0.53 0.48...0.59	Δt =7s by independent command
Beginning of SM suspending on the fall-line and inflation of the inflatable shock-absorbing device	229.9 202.6...272.5	13.1 10.4...15.6	73.9 67.34...80.69	-45.79 -44.23...-47.92	11.96 10.6...13.53	0.34 0.30...0.37	Δt =10s from the moment of PS deployment
Termination of the SM suspending, separation of the aerodynamic shield from the SM	235.9 208.6…278.5	12.9 10.3...15.4	28.29 26.26...30.27	-64.4 -62.28...-66.93	1.78 1.64...1.94	0.13 0.12...0.14	The parameters of motion of upper units are given
Landing at the level of h=2 km	603.3 493.2...719.2	2.0	23.6 22.40...25.00	-90	3.31 3.15...3.49	0.102 0.097...0.108	
Landing at the level of h=0 km	691.7 576.7...812.5	0.0	21.6 20.50...22.90	-90	3.32 3.16...3.50	0.093 0.088...0.098	

Table 2: Listing of important calculation results for the EDL phase

The first contact with the ground occurs at a vertical velocity of approximately 25 m/s where a maximum impulsive deceleration of 200 g is experienced by the surface module. The surface module is however well protected by a large airbag, which has an approximate diameter of ∅ 80 cm.

After several rebounds of the airbag, which can be as high as 50 m, the airbag comes to rest. A detonating device releases the 4 kg surface module, which has a mechanism to upright it, and hence ensures its correct orientation as shown in Figure 7.

Figure 7: Final steps in the EDL phase:
- First contact with the ground (Touchdown)
- Several rebounds
- Pyrotechnic separation of the airbags
- Upright orientation manoeuvre of the Mars station

Figure 8 depicts how the surface module can be up righted by means of an attached verticalisation ring, which is initially folded onto the module but released when a pyrotechnic bolt is fired.

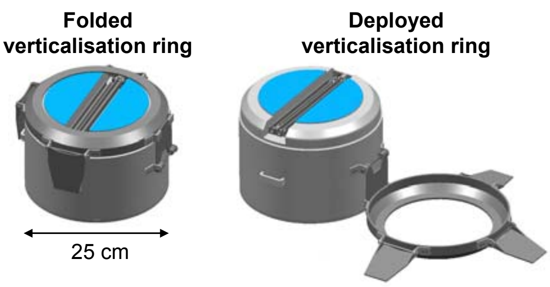


Figure 8: Concept for up righting the surface module via verticalisation ring

4. Payloads

Following a study by the Micro-Mars development team, the following payload was identified which can be mechanically integrated into the small surface module geometry and which can also be accommodated within the available energy budget:
- Magnetometer
- Infrared radiometer
- Dosimeter for measurement of UV radiation
- Dosimeter for measurement of charged particles

These four devices will make up the framework for the First Mars Observatory (FMO), driven by a common controller card (Figure 11). A fifth device comprises a small camera system that will film the surrounding landing environment.

The text within Figures 9 and 10 describe the payload package and scientific experiments of the FMO. The low energy demand but yet high technical capability makes the FMO very suitable for application within the Micro-Mars station. Sensors for material analysis such as a Mössbauer spectrograph and an alpha-proton-X-ray-spectrometer (APXS) cannot be incorporated in the surface module as a result of their high mass and large energy requirements.

First Mars Observatory (FMO)

The First Mars Observatory is an instrument package to measure environmental parameters.
The following sensors have been selected:

Four Channel Infrared Radiometer:
- to measure the radiation flux from the surface
- to determine the thermal inertia (K, c, Δ)
- to give clues on surface properties (temperature, roughness, emissivity)
- to deliver ground truth for orbiter instruments

Fluxgate Magnetometer:
- to measure for the first time the magnetic variation spectra on the surface
- to investigate for the first time the Mars - solar wind interaction by magnetic field measurements above and below the ionosphere (on orbiter and lander)

Charged Particles & Neutral Components Detector and UV-Detectors
- to measure the radiation climate on the surface of Mars
- to quantify the different components and their biological effectiveness
- to determine their variation with time (diurnal, seasonal and in dependence on solar activity)

Figure 9: Experiments of the FMO

Instrument Status
- The Infrared Radiometer and the Magnetometer are fully developed, qualified and flight models are already manufactured for the Rosetta Mission (Parts of the Lander experiments Mupus and Romap).
- The Charged Particles & Neutral Components Detector and UV-Detectors are based on the Experiment Radex (Radiation Climate on Mars as Relevant to Exobiology) which was selected for the EMF platform during one of the upcoming NASA Surveyor program Lander Missions.

Sensor accommodation

	Accommodation
Infrared Radiometer	towards surface
Magnetometer	outer position but not close to solar cells
Radiation + UV	towards sky
Electronics	inside the thermally isolated compartment

Required Recourses

	Mass	Power Consumption	Telemetry Rate	
Infrared Radiometer	100g	150mW	10kbit/day	
Magnetometer	120g	350mW	500kbit/day	
Radiation + UV	250g	300mW	200kbit/day	
Controller	100g	180mW	10kbit/day	
Box	120g			
Σ	**690g**	**980mW**	**720kbit/day**	**= 90kBytes/sol**

Figure 10: Status and requirements of the FMO

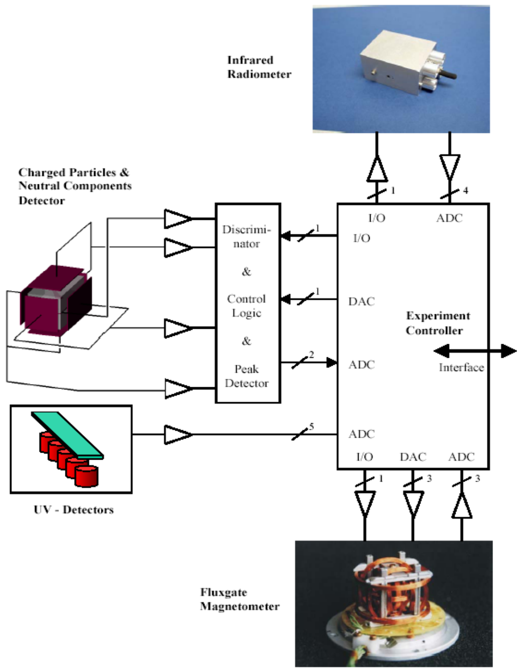


Figure 11: Control and experimental data capture of the FMO by means of a common electrical interface

5. Surface module / Mars station

The experiments of the First Mars Observatory (FMO) require the availability of a small boom, firstly to raise the camera but more importantly to raise the magnetometer to a sufficient height to be clear of the electrical system of the surface module, as the magnetic field of the latter would otherwise interfere with the magnetometer measurements.

Figure 12 depicts the Mars station after all its components have been deployed:

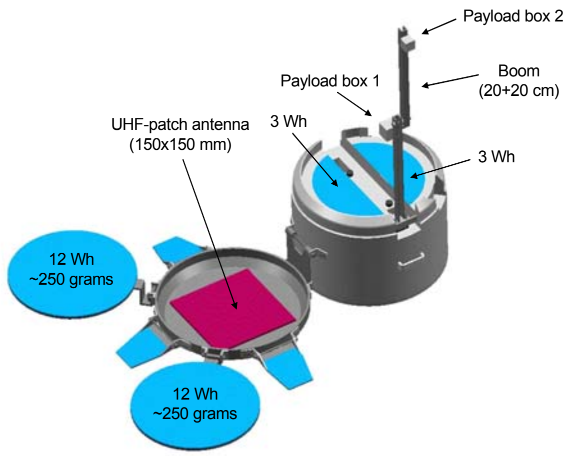


Figure 12: Mars station ready for operation

The solar cells can also be seen in Figure 12, where 24 Wh of energy are generated per Martian day via 2 round solar cell discs each of diameter ∅ 22 cm. Additional solar cell exposed surfaces generate a further 6 Wh of energy. A total of 30 Wh of energy is hence available per Martian day to drive the subsystems and experimental equipment of the Mars station as listed in Table 3.

Energy drain :

Item	Power usage (W)	Time (h)	Energy (Wh)
Avionics	0.250	24.000	6.000
Transceiver (including antenna)	18.000	0.167	3.006
Antenna	0.000	0.167	0.000
Thermal	1.500	8.000	12.000
Magnetometer	0.400	8.000	3.200
Radiometer	0.150	8.000	1.200
Dosimeter	0.300	8.000	2.400
Camera	1.000	0.500	0.500
Experiment Electronics	0.180	8.000	1.440
			29.746

Energy supply :

Item	Power input (W)	Time (h)	Energy (Wh)
Battery (rechargeable)			27.000
Solar panel (mean power)	2.500	12.000	30.000
RTG (optional, electrical only)	0.120	24.000	2.880
			59.880

Table 3: Drain and supply of energy on the Mars station

The main experiments of the FMO could, for example, be in operation for up to 8 hours per Martian day. A variation of the operation times is also possible. The biggest consumer of energy is the transceiver, which is required to communicate with the orbiter. This communication is however limited to 10 minutes per Martian day, as a result of the short time frame where the orbiter may be in direct contact with the lander. This short time frame is also a result of the surface module being placed in the equatorial region of Mars due to thermal limitations of the Mars station.

Communication is carried out on UHF-Band at 430 MHz. A 150 x 150 mm patch antenna is used for this purpose, which can be integrated within the verticalisation ring (Figure 12).

directly payload-related: 0.776 kg

Item	Mass (kg)
Avionics	0.200
Transceiver	0.230
Antenna	0.200
Thermal	0.200
Magnetometer	0.120
Radiometer	0.100
Dosimeter	0.250
Camera	0.086
Experiment Electronics	0.220
Battery (rechargeable)	0.222
Solar Panels (2x0.25kg + 0.05kg)	0.550
RTG (optional)	0.400
Cabling	0.100
Structure & Boom	1.000
Miscellaneous	0.122
	4.000

Table 4: Mass budget of the Mars station

A mass budget of the Mars station with all its subsystems and experimental equipment is listed in Table 4. Table 5 provides the corresponding dimensions available to date.

Item	Dimensions (cm)
Avionics & Experiment Electronics	10 x 8 x 6
Transceiver	9.14 x 5.84 x 2.54
Antenna	15 x 15 x 3.2
Thermal	
Magnetometer	Ø 6 x 7
Radiometer	5.3 x 4 x 3
Dosimeter (UV)	2.5 x 2.5 x 4
Dosimeter (Charge Particles)	5 x 2 x 1
Camera	4.25 x 4.3 x 3.35
Battery (rechargeable)	7.6 x 5.2 x 2.6
Solar Panels	Ø 23 x (0.2 + 0.3)
RTG (optional)	Ø 7 x 11
Cabling	
Structure & Boom	Ø 25 x 15
Miscellaneous	

Table 5: Dimensions for the Mars station subsystems and experimental equipment

The experimental requirements as shown in Figures 9 and 10 together with the details provided in Tables 3 to 5 lead to a preliminary design of the Mars station as shown in Figure 13.

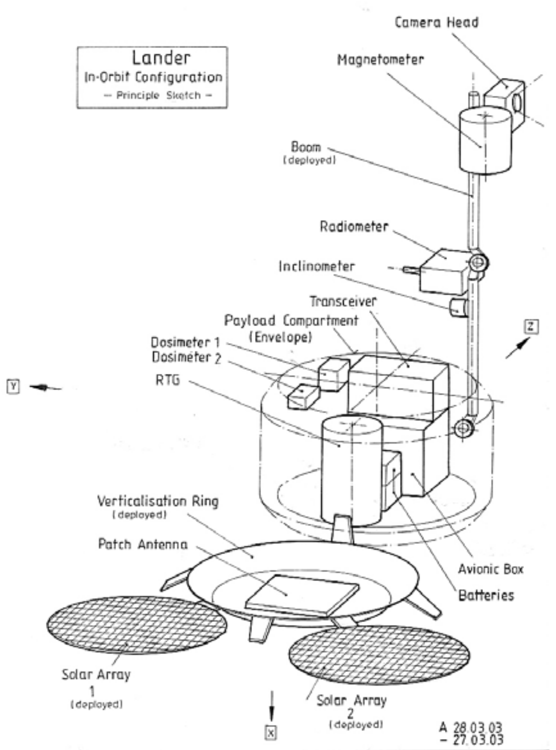


Figure 13: Preliminary study for the design of the Mars station

It can be seen from Figure 13 how all the subsystems and experimental equipment can be integrated within the overall diameter of 25 cm which has been allocated for the surface module structure made primarily out of carbon fibre composite materials.

6. Aspects of the power subsystem

A critical element of all lander based Mars missions is the power subsystem, which caters for both the electrical and thermal requirements of the surface module. By means of choosing a landing site in the equatorial region of Mars, many thermal problems can already be reduced. Despite this, temperatures of -70°C can still be experienced which freeze up many subsystem components. Particularly problematic is sourcing batteries that can operate at such temperatures. The use of additional devices based on radioactive substances such as Radioisotope Thermal Generators (RTG) can help to increase temperatures and by means of heat pipes to stabilise the temperature of devices, in particular that of the batteries, within their specified operating ranges.

Consequently a Plutonium 238 based RTG module (Figure 14) with 204 Wh of thermal energy per Martian day was considered and is listed as an optional device in the tables of Section 5. In using such a device, the operating life of the surface module can be extended to several months depending on a thermal analysis study, which is still to follow. If it should not be permitted to use an RTG module, it could be replaced by a second battery pack. However, the temperature problem would still not be solved and the operating life of the surface module would be limited to days.

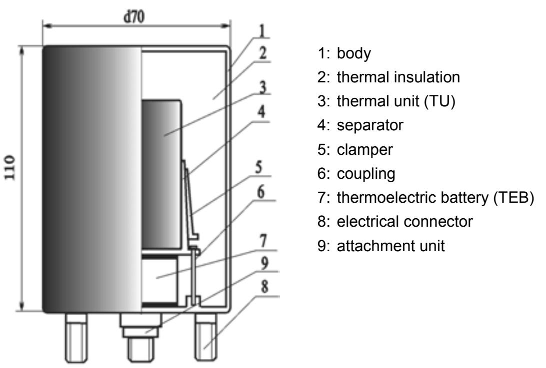
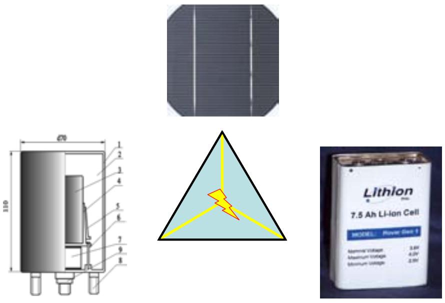


Figure 14: RTG module (204 Wh of thermal energy and 2.88 Wh of electrical energy per Martian day)

Independent of the thermal limitation, the electrical energy will be generated, as already described, by using solar cells (30 Wh per Martian day).

Figure 15 depicts all the involved power subsystems of the Mars station (see also Table 3).

Solar Cells (30 Wh per sol)



Radioisotope Thermoelectric Generator (204 Wh thermal, 2.88 Wh electr. per sol)

Lithium Ion Rechargeable Battery (27 Wh)

Figure 15: Power subsystems of the Mars station

7. Data processing / Avionics

Data processing for the Mars station is carried out using a small COTS low power microcontroller card and is already considered as part of the avionics subsystem in Tables 3 to 5. Similar solutions have been adopted in the NASA "Deep Space 2" Mars mission. A SRAM storage capacity of 2 MByte is sufficient to store experimental data for up to three days. A total of 600 KByte of experimental data is collected per Martian day, which can be uploaded to the orbiter within the 10-minute window available for radio contact. The text in Figure 16 lists several details for the OBDH, whilst Figure 17 lists aspects relating to the volume of data which will be handled.

The data upload rate from the lander to the orbiter is at 8,000 bit/s. Most data will relate to the experimental measurements whilst typical housekeeping data will also be handled. The data download rate from the orbiter to the lander is achieved by telecommand at 20 bit/s.

Low-cost approach: space qualified (MIL 1553), but COTS components:

- Similar to *Deep Space 2* electronics and software architecture, but with modified and enhanced memory (SRAM, 2 MBytes) to save and hold data during some Martian days via a *history buffer* concept:
  - Experimental data
  - Surface module housekeeping data
  - Imaging data
- Dataprocessing unit:
  - Creditcard-sized OBC
  - Microcontroller-based: 80C51, C167 or similar
  - Common i/o interfaces like RS485, CAN onboard
- Power electronics board
- Memory:
  - SRAM, 2 MBytes
  - Needs ~ 3.5 orbiter↔lander contacts for upload
- Software: C, Assembler

Figure 16: Details of the avionics and OBDH of the Mars station

Data volume per Martian day (kBytes):	
- Experimental data:	90 (see *proposed payloads)*
- Surface module housekeeping data:	10
- Imaging data:	500 (compressed raw image data)
Sum:	**600 kBytes**
SRAM volume:	2,000 KBytes
→ Data capacity covers 3 Martian days:	3 * 600 KBytes (+ margin)
→ Data dump per lander↔orbiter contact:	600 kBytes (see *communications*)

Data volume per Martian day = data dump per orbiter↔lander contact

Due to the landing site selection of the Martian station and the foreseen orbit of the orbiter there will be one orbiter↔lander contact per sol during the first operational week of the Martian station. Later possibilities of orbiter↔lander communications depend on line of sight between the two spacecrafts.

Figure 17: Data volumes and data transfer details

8. Summary

During the phase-A study of the Micro-Mars Mission, the foundation for the low cost mission has been developed. This involves an initial design of the lander, an investigation of the entry and landing process (EDL phase) and the selection of an appropriate experimental package together with its operation within a very limited energy budget. The feasibility for using a 16 kg landing module has been verified. A detailed study will follow in the next stage of the project development. From a strategic point of view, a low cost mission to Mars is a suitable candidate for contributing to the ESA Aurora Program.

9. Abbreviations

APXS	Alpha-proton-X-ray spectrometer
ASAP	Ariane structure for auxiliary payload
COTS	Commercial off the shelf
g	Gravitational acceleration
FMO	First Mars observatory
GTO	Geostationary transfer orbit
MMH	Mono methyl hydrazine
MON	Mixed Oxides of Nitrogen
OBDH	Onboard data handling
PS	Parachute system
RTG	Radioisotope thermal generator
S/C	Spacecraft
SM	Surface module
S/S	Subsystem
UHF	Ultra high frequency
Wh	Watt hours

10. Literature

[1] Micro-Mars Mission, phase-A study, final presentation, EADS Space Transportation, 2003
[2] ASAP-5 manual, Arianespace, Paris
[3] Micro-Mars Mission, scientific-technical report, Babakin Space Centre, Moscow, 2002