MICRO MARS: A LOW COST MISSION TO PLANET MARS WITH SCIENTIFIC ORBITER AND LANDER APPLICATIONS

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ABSTRACT

The proposed Micro-Mars mission can contribute substantially to the international Mars exploration program within the framework of a future low cost mission. The concept consists of an orbiter integrating a total scientific payload of 30 kg including a light-weight lander of 15 kg. The spacecraft will be launched as piggyback payload by an Ariane-5 ASAP with a total launch mass of 360 kg. It will use a bipropellant propulsion system with 210 kg of fuel and four thrusters of 22 N, and four of 10 N for orbit and attitude control. Further attitude actuation shall be performed by three reaction wheels and a gyropackage, a star sensor and a sun sensor for attitude sensing. Communication will be performed in S- and X-band to earth and in UHF between orbiter and lander.

From a highly elliptical orbit with a periapsis below 200 km, four instruments will perform high-resolution remote sensing observations and the payload consists of a camera system, a magnetometer, a dosimeter, and an ultrastable oscillator for radio science.

The light-weight micro lander is a challenging technological experiment by itself. It is equipped with a suite of scientific instruments which will supplement the orbiter measurements and concentrate on the environment (temperature cycle, atmosphere, magnetosphere and radiation).

1. INTRODUCTION

In the year 2000, DLR Berlin as prime investigator and EADS Space Transportation GmbH, Bremen as prime contractor for the space segment started the idea of a small, low-cost mission to Mars with a total launch mass of about 360 kg on the ASAP-5 platform [1]. Meanwhile, we have performed a phase A study under contract of the German Space Agency (DLR Bonn), together with DLR Berlin as PI, several German Universities (Berlin, Braunschweig, Bremen, Köln, Mainz, München, Münster), MPI Mainz and DLR Köln for the scientific payload, and two additional partners, EPFL Lausanne/Switzerland and MD Robotics/Canada.

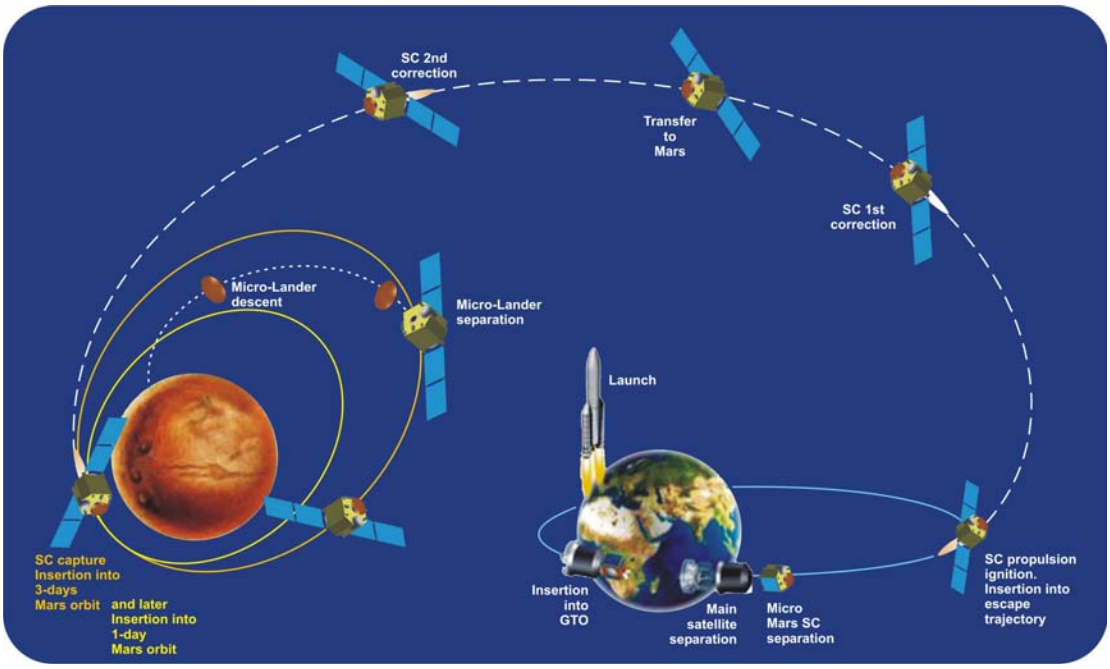

Figure 1: Micro-Mars mission scenario

For the baseline mission scenario (figure 1), a launch in 2007 was assumed with Ariane-5 piggyback (ASAP 5) using one mini-adapter launch opportunity. The 360 kg spacecraft allows a 15 kg payload for the Mars orbiter and an additional very small micro lander of about 15 kg total mass for a soft landing on the Martian surface. The lander shall allow for a payload mass of about 2 kg and a lifetime of > 1 week, whereas the orbiter shall be designed for a lifetime of 3 years including the 11 months journey to Mars. Mission operation is based on one ground station, allowing for a maximum of 8 hours contact time between the orbiter and the earth station for communication and data downlink.

2. SCIENTIFIC OBJECTIVES

The Micro-Mars orbiter and lander mission has to be seen within the framework of the international Mars exploration program. This was taken into account by defining its scientific goals and by selecting the baseline payload. Small satellite missions like Micro-Mars are very well suited to close existing gaps in and to provide complementary data to the ongoing exploration effort. This has led to the formulation of the scientific objectives as:

- Characterization and investigation of small-scale surface features and related geologic processes through very high resolution imaging from orbit in the decimeter-resolution range
- Refinement of the Martian magnetic field and its anomalies from orbit and determination of magnetic field properties on ground
- Local and regional measurements of the Martian gravity field
- Detailed analysis of the atmosphere by radio sounding, imaging, and in-situ measurements on the Martian surface
- Detailed investigation of the ionosphere by measuring the magnetic field in orbit and on ground as well as by radio sounding
- Investigation of the galactic and solar radiation environment during cruise, in Mars orbit and on ground
- Preparation for future unmanned and manned landing missions by:
    - Characterization and provision of context information for future landing sites from orbit
    - Determination of the biologically effective radiation during cruise, in orbit around Mars and by in-situ measurements
    - Improvement in our understanding of the environmental conditions on the Martian surface (temperature cycle, atmospheric properties, opacity)
- Support probable other Martian lander missions by providing the capability of telemetry contact
- Unique capability to perform high resolution remote sensing observations of Deimos (as well as Phobos)

3. MISSION ARCHITECTURE

The overall main objective of this mission is to provide a small and low-cost system. Therefore, the entire low-cost project is driven by the design-to-cost philosophy which requires to launch the spacecraft by a piggyback opportunity - in this case by ASAP 5 - into GTO. The Micro-Mars mission architecture consists of the following elements:

- Launch into GTO by Ariane-5 piggyback
- Mars orbiter including its payload
- Mars lander including its payload
- Mission control center and one ground station

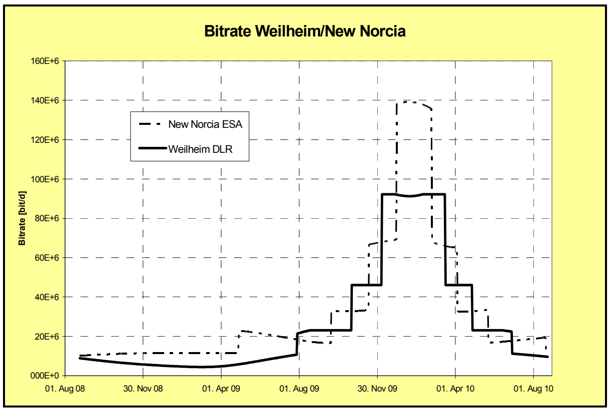

Figure 2: Daily data dump to earth for the ground stations at Weilheim and New Norcia

The Micro-Mars orbiter with the integrated lander uses a propulsion system responsible for the Mars transfer injection, the correction manoeuvers and the Mars capture manoeuver into a 3-days Mars orbit. This will be followed by a release manoeuver for the Mars lander and an orbit change manoeuver controlled by the mission control center located at DLR GSOC with its own ground station. Daily contact time of about 8 hours for the scientific data dump to earth will be available (figure 2). The Mars micro lander will collect scientific data for some days and will transmit them to the orbiter during its contact times.

4. ORBITER SCIENTIFIC EXPERIMENTS

The Micro-Mars orbiter payload consists of four different instruments which are described in more detail below. The power supply for each of the instruments is provided by the spacecraft power subsystem resulting in a major mass saving for the orbiter payload. Additional reductions could be achieved by integrating some of the instrument electronics within the onboard data handling system. A summary of the payload mass and power budgets is given in table 1. Concerning the power, it has to be noted that not all of the instruments will be operating in parallel and that the imaging system will be only switched on for a maximum of 30 minutes.

Experiment	Mass [kg]	Power [W]
Very-High Res. Imager	9.20	10.0
Medium-Res. Camera	1.50	2.1
Low-Res. Camera	0.30	2.1
Digital Unit (Camera)	1.40	7.0
Magnetometer	0.50	0.6
Magnetometer Boom	0.25	n.a.
USO Radio Science	0.50	4.0
Dosimeter	1.50	3.0 – 5.0
Orbiter payload total	15.15	28.8 – 30.8

Table 1: Orbiter experiments mass and power budgets

4.1 MICRO-MARS IMAGING SYSTEM

The Micro-Mars imaging system comprises three different camera heads with the Very High-Resolution Imager, the Medium Resolution Camera and the Low Resolution Camera. A common digital unit performs all tasks of camera control and command, data processing and data compression. Wavelet data compression is performed off-line by a signal processor and the digital unit has an internal buffer of 3 Gbit to store the raw image data before compression. It is a box with the dimensions of 150 mm x 75 mm x 50 mm and a mass of 1.4 kg.

Very High-Resolution Imager: It is equipped with a light-weight telescope (Richy-Chretien) with an effective focal length of 4 m and an aperture of 40 cm. It is manufactured from C/CsiC to reduce mass. Its length is 500 mm. The sensor electronics is based on the development at DLR for the ROLIS camera onboard the Rosetta Lander and for the SRC of the HRSC experiment onboard Mars Express. The focal plane is equipped with 3 interline CCD area arrays with 1024 x 1024 pixel. This enables the short integration time of about 0.1 msec. Time-delayed-integration (TDI) is applied to improve the SNR and to reduce the optics requirements. The 3 detectors are arranged in two rows of 2 and 1 CCD, respectively, in such a way that the gap in the front row between the two detectors is closed by the second row. The CCDs of the rows overlap yielding an effective detector area of 2800 x 1024 pixel. At periapsis altitudes of 200 km, the resulting spatial resolution will be 0.45 m/pixel.

Medium Resolution Camera: It consists of the optics with baffle, a filter wheel with 6 positions and the sensor electronics. The optics is a 400 mm focal length telescope with an aperture of 60 mm which is build in C/SiC. The modular detector and sensor electronics integrates the focal plane assembly with the CCD-detector and the sensor electronics into a compact 3-dimensional multi-chip module. The detector is a fast frame-transfer CCD with 1024 x 1024 pixel and a radiometric resolution of 14 bit. The resulting spatial resolution at 200 km is 6.5 m/pixel.

Low Resolution Camera: In order to allow for two color imaging which is needed to discriminate between bright dust, ices and condensates, the Low Resolution Camera consists of two identical cameras, but each equipped with a different color filter (blue and red, respectively). The detector is a 1k x 1k APS CMOS detector resulting in reduced system complexity and power demands. The optics has a focal length of 30 mm yielding a spatial resolution of about 2.3 km at 10,000 km altitude. Spatial resolution and SNR are by far sufficient to meet its scientific goal of weather monitoring.

4.2 MICRO-MARS MAGNETOMETER

The Micro-Mars magnetometer consists of a triaxial fluxgate sensor with associated sensor electronics and a digital processing board. It is mounted on a boom with deploy mechanism in order to improve the measurement quality. The magnetometer range is ± 2000 nT and the inherent resolution is 10 pT. The electronics are integrated within the orbiter data handling system. Similar fluxgate sensors were already successfully flown e.g. on Phobos, Integral, and DS1 and are also part of the Rosetta payload. No DC magnetic cleanliness is planed. The high elliptic orbit implies the chance to calibrate the instrument in-flight. Special measures for magnetic cleanliness are not required.

4.3 MICRO-MARS RADIOSCIENCE

Two radio link modes are used to perform the radio science experiments:

- Two-way mode: the ground station transmits an X-band uplink signal, received by the spacecraft and transponded coherently back to Earth at two downlink radio signals at X-band and S-band. The frequency stability of the radio link is governed by the hydrogen maser of the ground station.

- One-way mode: the spacecraft transmits at two simultaneous and coherent downlink frequencies at X-band and S-band. The frequency stability is governed by the on-board Ultrastable Oscillator (USO).

Major requirements for the radio science experiment are an X-band uplink, simultaneous and coherent S- and X-band downlink via the HGA, S-band downlink operational if radio science experiments are performed, and periodical simultaneous ranging at S-band or X-band.

The one-way downlinks need to be stabilized to an accuracy of 10^{-13} at 3 seconds integration time in order to be capable to detect the slight phase changes of the carrier caused by the bending in the thin and rarified Martian atmosphere. This will be verified by an Ultrastable Oscillator (USO) connected as external reference source to both transponders. The selected USO is designed and will fly onboard of NASA´s Pluto Kuiper Belt spacecraft “New Horizons” towards the Pluto-Charon system. ITAR restrictions do not allow to present technical details.

4.4 MICRO-MARS DOSIMETER

The Micro-Mars radiation experiment on the orbiter shall measure the Linear Energy Transfer (LET) spectra in three orthogonal directions, time resolved counts and dose rates for charged and neutral components of galactic and solar particle radiation during cruise and around Mars. These objectives will be achieved by using the following instruments:

1) A DOSimetry TELescope (DOSTEL-3D) consisting of 6 planar silicon detectors placed at each side of a cube. The construction is based on the design and the experience with the dosimetry telescope which was flown already several times. It uses a detector head which is built up as a cube each side consisting of a rectangular silicon detector forming a 3D detector telescope. The design allows a pulse height analysis of the detector signals with 255 channels of 15.65 keV width for low energy deposits up to 4.0 MeV and 255 channels of 313 keV width for high energy deposits up to 80 MeV.

2) A scintillator consisting of a cube of organic B430 material surrounded by a detector consisting of the same material and acting as an anticoincidence detector. The light output of the scintillators will be measured with photo diodes instead of photo multipliers. Pulse shape analysis is used to discriminate against photons. The scintillator is sensitive to neutrons with energies > 1 MeV.

The volume of the entire Dosimeter experiment including electronics is 100 mm x 100 mm x 100 mm.

5. MISSION OPERATIONS

From launcher separation until preparation of the Mars capture maneuver the Micro-Mars spacecraft will be normally in a sun-oriented attitude mode. The periods of orbit change maneuvers (table 2), however, require a thrust vector orientation. During this 320 days long Mars transfer phase of sun-orientation, the payloads are switched off with the exception of the dosimeter which will monitor the radiation environment at low data rates.

Maneuver	Delta-V (m/s)	Propulsion Mass (kg)	Burn Time (min)
GTO to Hyperbola	1,500	141.93	80.4
Midcourse 1	50	3.61	2.05
Midcourse 2	20	1.43	0.81
Midcourse 3	30	2.12	1.20
Mars Capture	775.98	48.18	27.30
Lander Entry	28.70	1.55	0.88
Lander Separation	--	--	--
Peri-Lifting	11.96	0.58	0.33
Apo-Lowering (1day)	119.91	5.72	3.24

Table 2: Micro-Mars course maneuvers

After injection into Mars orbit, the lander will be released and placed on the Martian surface. Further maneuvers are required to reach the final orbit and to start the scientific operational phase (table 1). The Mars orbit is a high elliptical 24.5 h orbit with a periapsis of 200 km altitude. Normally, the spacecraft will be in a sun-oriented mode. The main payload, the high resolution camera, however, will acquire images of the Martian surface mostly around periapsis, which requires a nadir orientation of the spacecraft for a maximum period of 30 minutes. If visibility conditions enable a direct contact to the earth ground station, an attitude change will be performed to point the high gain antenna directly towards earth and to transmit the payload data. During the first days of the Mars orbital phase, the lander will collect scientific data and store them until the orbiter will be in a position to receive these data and store them aboard until its next earth ground station contact.

6. MICRO-MARS VEHICLE DESIGN

The Micro-Mars mission will be launched by Ariane-5 ASAP and has to fit into the adapter envelope during launch (figure 3). The primary structure of the orbiter, which also integrates the micro lander, consists of a central tube with attached platforms for payload and subsystem units (figure 4).

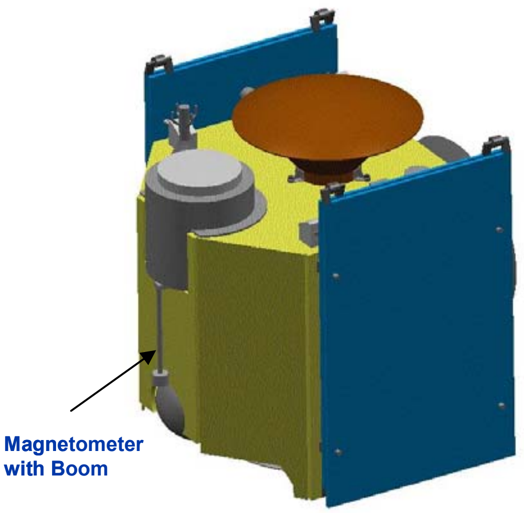

Figure 3: Orbiter and lander in launch configuration

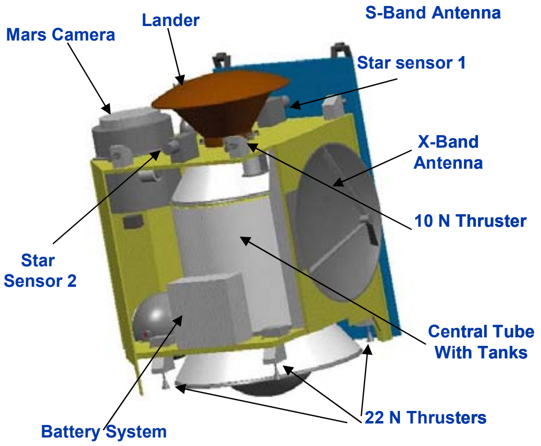

Figure 4: Inside view of the orbiter

The orbiter is equipped with a bi-propellant propulsion system, consisting of two large tanks inside the central tube, four 22 N thrusters for the orbit maneuvers and four additional 10 N thrusters allowing in combination with the 22 N thrusters a three axes attitude torque capability as back-up for the reaction wheels as well as for the wheels off-loading. The GNC subsystem uses an onboard computer commonly with the data handling system for guidance and control of the attitude. Attitude measurement will be achieved by 2 star sensors, a sun-sensor and an Inertial Measurement Unit (IMU).

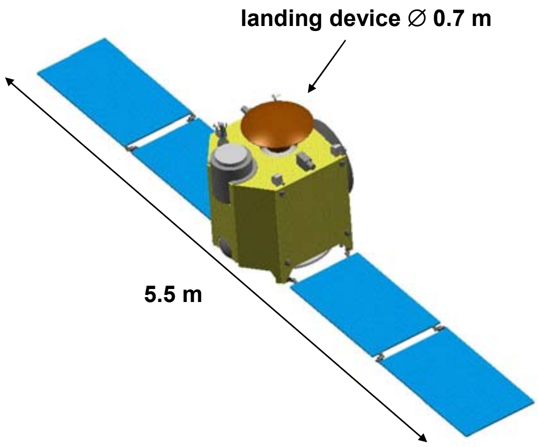

Figure 5: Micro-Mars flight configuration

The necessary electrical power will be provided by 2 deployable solar arrays in combination with a NiH2 battery system and a power control unit. After unfolding, the Micro-Mars spacecraft will have a dimension of 5.5 m (figure 5). The telemetry/telecommand system will collect the housekeeping and payload data and will distribute the received telecommands to the subsystems controlled by the onboard computer. This TTC system is connected with the communication system, which transmits the data via its X-Band downlink to the receiving station on earth (Weilheim/Germany). A one meter reflector antenna as high gain antenna allows a 1 kbit/s data rate. Telecommand will be transmitted from earth via an S-Band communication system.

Thermal control will be provided by a simple passive system with some additional heaters controlled by thermostats. Table 3 shows the mass budget of the Micro-Mars spacecraft and figure 6 depicts the system dimensions.

Payload incl. Lander	30
Propulsion S/S	36
Communication S/C	11.5
GNC S/S	6.5
Power Supply	16.0
Data Handling S/S	4.0
Structure S/S	40.0
Thermal Control S/S	7.8
Bi-Propellant	205.0
Margin	3.2
Launch Mass	360.0 kg

Table 3: Micro-Mars system mass budget

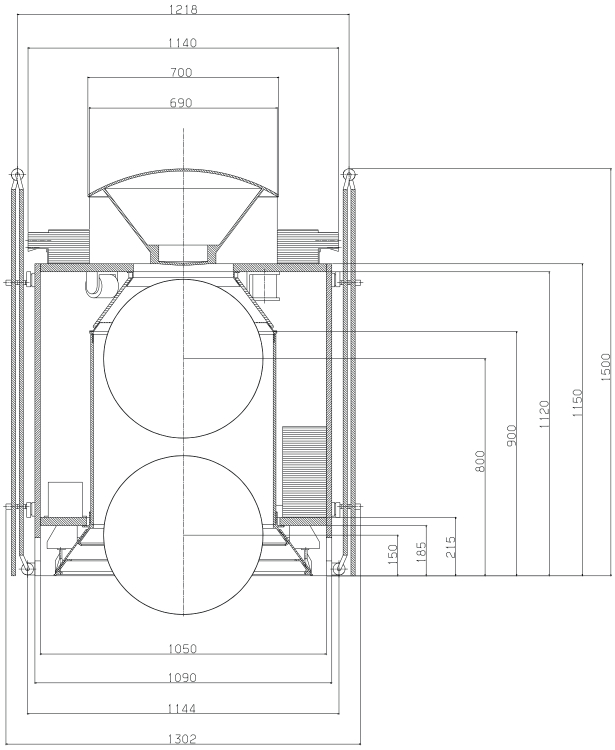

Figure 6: Lander and orbiter with their dimensions

7. MICRO-MARS LANDER DESIGN

The Micro-Mars lander will be a ballistic capsule system, transported to the Mars orbit by the orbiter, which will also perform the de-orbit maneuver, release the capsule and return into the original orbit. The lander will descend on a ballistic flight trajectory with an entry angle of about -12 degrees and a velocity of about 5,000 m/s.

The capsule will land in the equatorial region within ± 25° latitudes using a heatshield for the atmospheric deceleration with a maximum of -8 g and a maximum heat flow of about 180 kW/m^2. Subsequently, a parachute with a surface of 50 m^2 will reduce the descend velocity to about 20 m/s. In the last phase until touch-down, the lander is equipped with an airbag-system to reduce the landing shock for the surface station with a mass of about 4.2 kg of subsystem avionics and scientific payload instrumentation. The total mass of the lander is about 15 kg.

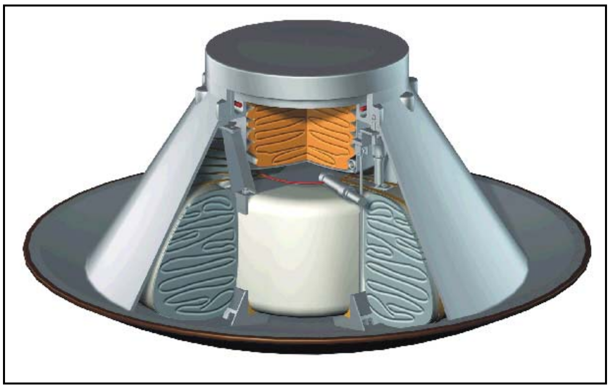

Figure 7: Micro-Mars lander in 3-D view

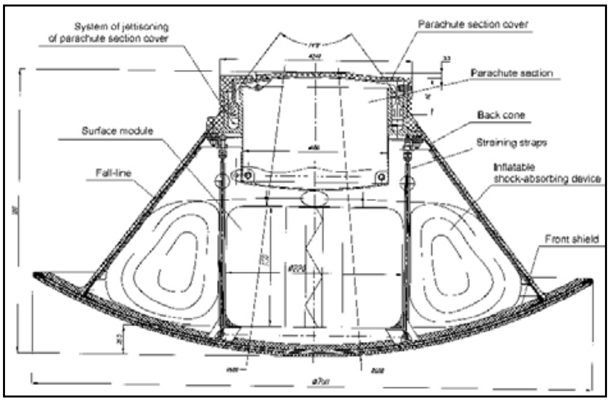

Figure 8: Micro-Mars lander with its dimension

Figure 7 shows the Micro-Mars lander in a 3-D view and figure 8 gives the main dimensions of the lander. The deployed surface station with the lander payload accommodation is shown in figure 9. The different phases for descent and landing are schematically shown in figure 10.

A number of different lander experiments were investigated covering the fields of environmental measurements, and geochemical measurements. Major constraints for selecting the baseline payload were low mass, low power consumption, and requirements with respect to mobility and system complexity. By taking into account the scientific advantage of complementary simultaneous measurements on ground and in orbit, a strawman payload was selected which concentrates on measurements of the Martian environment and comprises the following instruments:

- A magnetometer including an inclinometer to measure in parallel the properties of the magnetic field in orbit and on ground
- A dosimeter experiment measuring the dose rates of ionized radiation as well as the intensity and timely variations of the UV radiation on ground in addition to similar measurements in orbit
- A radiometer to measure remotely timely variations in temperature and to derive the physical properties of the surface
- A suite of atmospheric sensors to determine pressure, wind velocity and direction, and other parameters
- A camera equipped with an APS CMOS detector to determine the landing site position, to provide the context information of the surroundings, and to perform atmospheric measurements.

Experiment	Mass [g]	Power [W]
Dosimeter	150	0.30
Magnetometer	190	0.40
Radiometer	100	0.15
Atmospheric sensors	460	0.20
Camera	100	1.00
Lander payload total	**1000**	**2.05**

Table 4: Lander experiments mass and power budgets

A summary of the lander payload mass and power budgets is given in table 4. Many of the experiments are mounted on a boom (figure 9).

In order to cope with the available power, instrument operations on the lander will be phased in time.

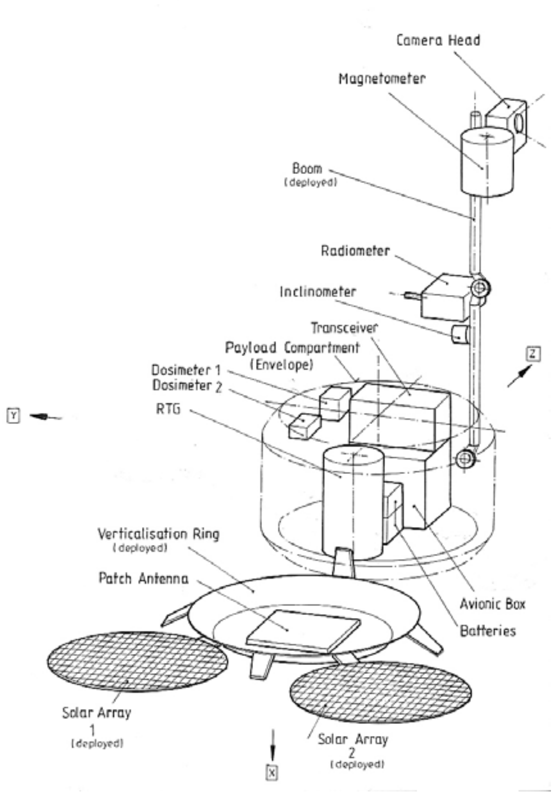

Figure 9: Micro-Mars surface station

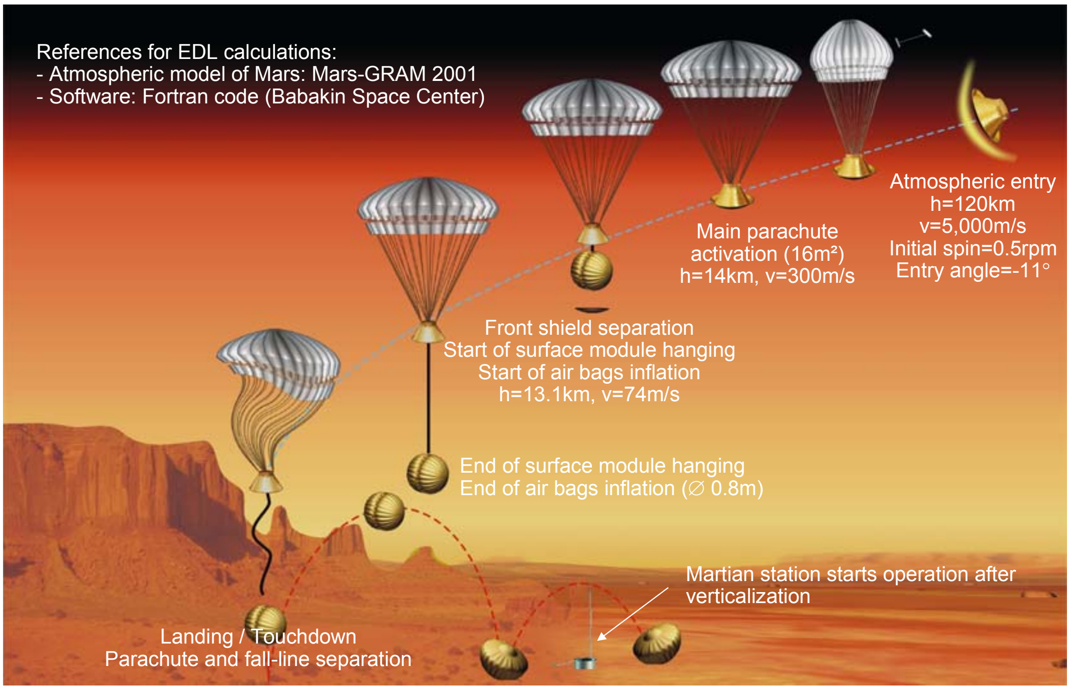

Figure 10: Micro-Mars lander descent and landing phases

8. SUMMARY

The Micro-Mars mission project has finished the phase A study and will start its phase B hopefully at the beginning of 2004 to meet the launch window in 2007 for an Ariane-5 piggyback launch into GTO. The injection window to Mars will be in September 2007 leading to an arrival at Mars in July-August 2008. The study did show, that a low-cost mission with a total mass of 360 kg including fuel is capable to transfer and operate a scientifically demanding payload in orbit around Mars as well as to place a small lander on the Martian surface. If this low-cost micro mission will be successful, the micro mission spacecraft has the capability to provide the S/C bus for further planetary missions, needing only minor modifications.

9. ACKNOWLEDGEMENTS

Babakin Space Center, Moscow carried out the lander design described in this paper [2] and also contributed to a preliminary orbiter configuration under a contract with EADS-ST GmbH Bremen.

10. REFERENCES

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