MICRO-MARS, A SMALL ORBITER AND LANDER TO PLANET MARS

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Abstract

The proposed Micro-Mars Mission can contribute substantially to the international Mars exploration programme within the framework of a future low cost mission. The Micro-Mars Mission concept consists of an orbiter integrating a total scientific payload of 30 kg to investigate Mars including a light-weight lander of 15 kg. The spacecraft will be launched in 2007 as piggyback payload by an Ariane 5 ASAP into GTO 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-band and X-band to earth and in UHF-band between orbiter and lander.

The orbiter will be designed as multipurpose platform, allowing the utilisation for missions to other planets to reduce development cost. The Mars orbit is highly elliptical with a periapsis below 200 km to enable high-resolution remote sensing. The payload is based on scientific needs for high-resolution remote sensing from orbit by taking into account future mission plans in order to avoid overlaps. A major goal is the investigation of surface material and small-scale geologic processes by imaging which is also needed for the detailed characterisation of landing sites and the preparation of future missions. These observations will be complemented by detailed measurements of the magnetic field, the gravity field, the radiation environment as well as atmospheric and ionospheric studies from orbit.

The light-weight micro lander is a very challenging task in the area of micro technology and a 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).

Introduction

Three years ago the idea to prepare a small, low cost mission to Mars has been created by DLR Berlin as prime investigator and Astrium GmbH Bremen as prime contractor for the space segment. To save launch cost it was foreseen to design a spacecraft of 360 Kg for a launch on ARIANE ASAP-5 into GTO. Meanwhile we have performed the 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 For the baseline mission scenario (fig. 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 more than 1 week whereas the orbiter with a high resolution camera as main payload element shall have a lifetime of up to 3 years inclusive the 11 months journey to the Mars orbit. For low cost reason only one ground station on earth will be available, allowing a maximum of 8 hours contact time between the orbiter and the earth station for data dumping.

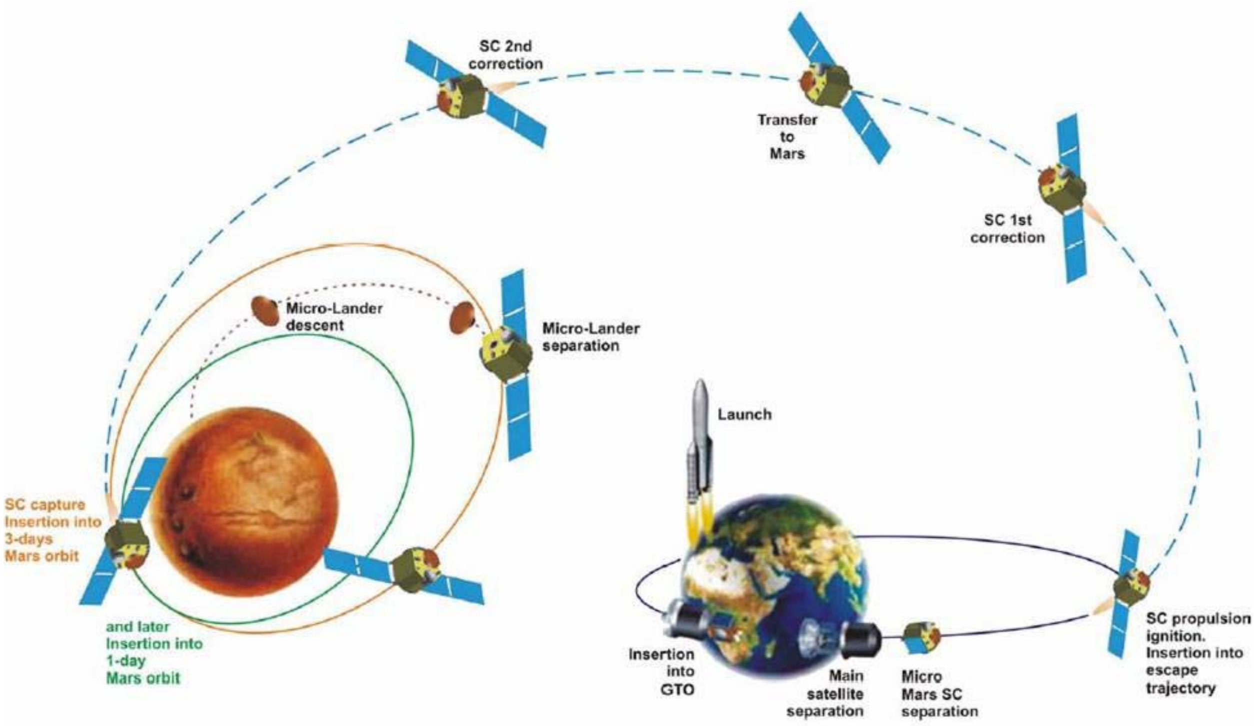

Fig. 1: Mission Scenario

<u>Scientific Objectives</u>

The Micro-Mars Orbiter and Lander mission has to be seen within the framework of the international Mars exploration programme. 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)

Mission Architecture

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

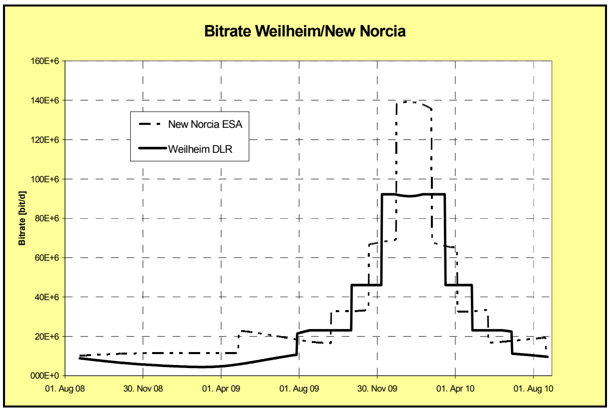

Fig. 2: Daily data dump to earth for the ground stations at Weilheim and New Norcia

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 Mars Orbiter with the integrated Mars lander will be equipped with a propulsion system responsible for the Mars transfer injection inclusive correction maneuvers and the Mars capture manoeuver into a 3-days Mars orbit. This will be followed by a release maneuver for the Mars lander and a Mars orbit change maneuver into a 24.5 h orbit controlled by the Mission Control Center located a DLR GSOC with its own ground station. Daily contact time of about 8 hours for the scientific data dump to earth will be available. The Micro-Mars lander shall collect scientific data for some days and will transmit them to the orbiter during their contact times.

Mission Operations

From launcher separation until preparation of Mars capture manoeuver the Micro Mars spacecraft shall be in a sun-oriented attitude mode without the periods of orbit change manoeuvers, which 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 and only 1 short TM/TC contact per day between the S/C and the MCS necessary to control the stored housekeeping data of the day.

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

Tab. 1: Orbit Maneuvers

After injection into Mars orbit, the lander will be released and placed on the Martian surface. Further manoeuvers are required to reach the final orbit and to start the scientific operational phase (tab. 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.

Mission Phases	Phase Duration	Attitude Orientation	Pointing Accuracy	Ground Contact	Power Consumption
Launch GTO Phase	10-18 days	Sun	5 deg	1x per day	low
Mars Transfer Injection Phase	hours	Thrust vector	0.1 deg	required	battery low
Mars Transfer Phase	11 months	Sun	5 deg	1x per day	low
Mars Capture Phase	hours	Thrust vector	0.1 deg	required	battery low
Mars operational orbit Phase	12-24 months	Sun, 15 hours Nadir, 1 hours Earth, 8 hours	5 deg 0.1 deg 1 deg	n.a. n.a. required	low battery high
Lander Separation Phase	hours	Thrust vector	0.1 deg	required	battery low

Tab. 2: Operational Mission Phases

Orbiter Scientific Instrumentation

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 tab. 3. 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

Tab. 3: Orbiter Experiment Mass and Power budget

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 pixels. 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 pixels. 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 400mm focal length telescope with an aperture of 60mm which is build in C/SiC. The modular detector and sensor electronics integrates the focal plane assembly with the CCD-detector and the sensor chip module. The detector is a fast frame-transfer CCD with 1024 x 1024 pixels 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.

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.

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.

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 200 mm.

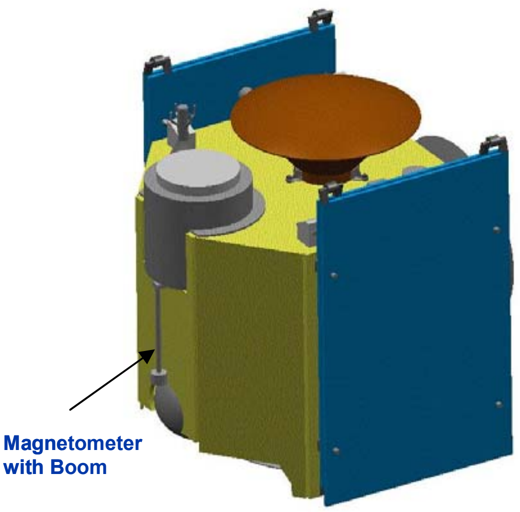

Fig. 3: Orbiter and Lander in Launch Configuration

Mars Orbiter Vehicle Design

The Mars Orbiter will be launched by ARIANE 5 ASAP and has to fit into the mini adapter envelope during launch.

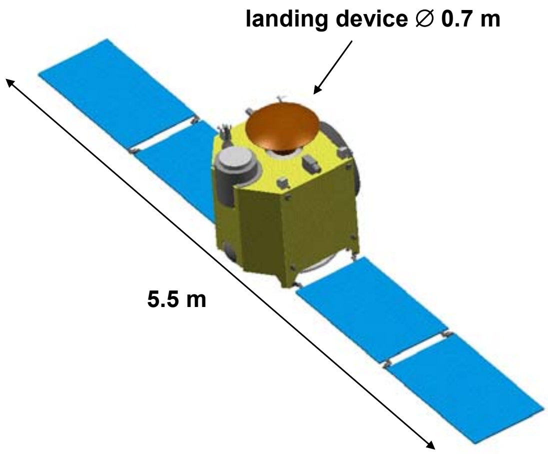

Fig. 4: Flight Configuration

The primary structure of the orbiter, which integrates also the micro lander, consist of a central tube with attached platforms for payload and subsystem units. 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 and also for the wheels downloading. 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). The necessary electrical power will be provided by 2 deployable solar arrays in combination with a NiH2 battery system and a power control unit. 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.

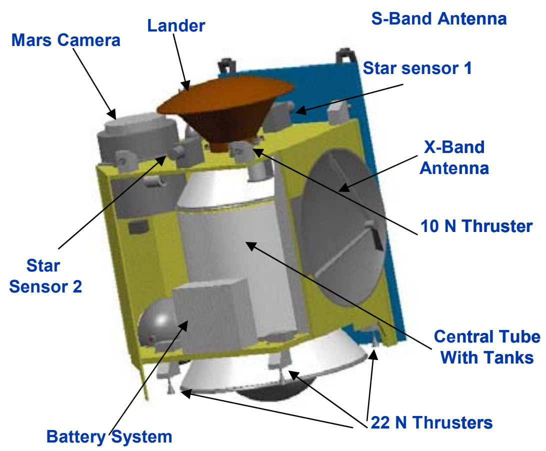

Fig. 5: Inside of the Orbiter

Thermal control will be provided by a simple passive system with some additional heaters controlled by thermostats. Fig. 5 gives an impression of the orbiter inside elements. Fig. 7 presents the orbiter and lander launch configuration and tab. 4 shows the mass budget.

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

Tab 4: System mass budget

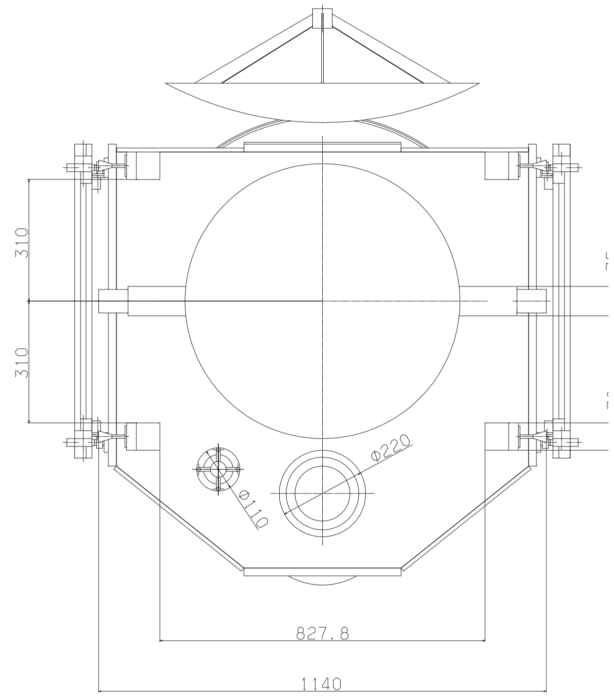

Fig. 6: Orbiter top view

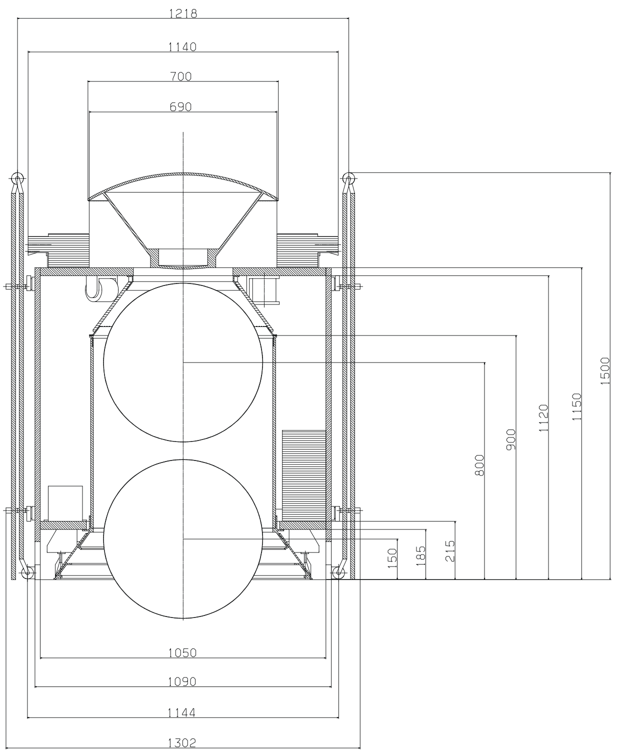

Fig. 7: Lander and orbiter with their dimensions

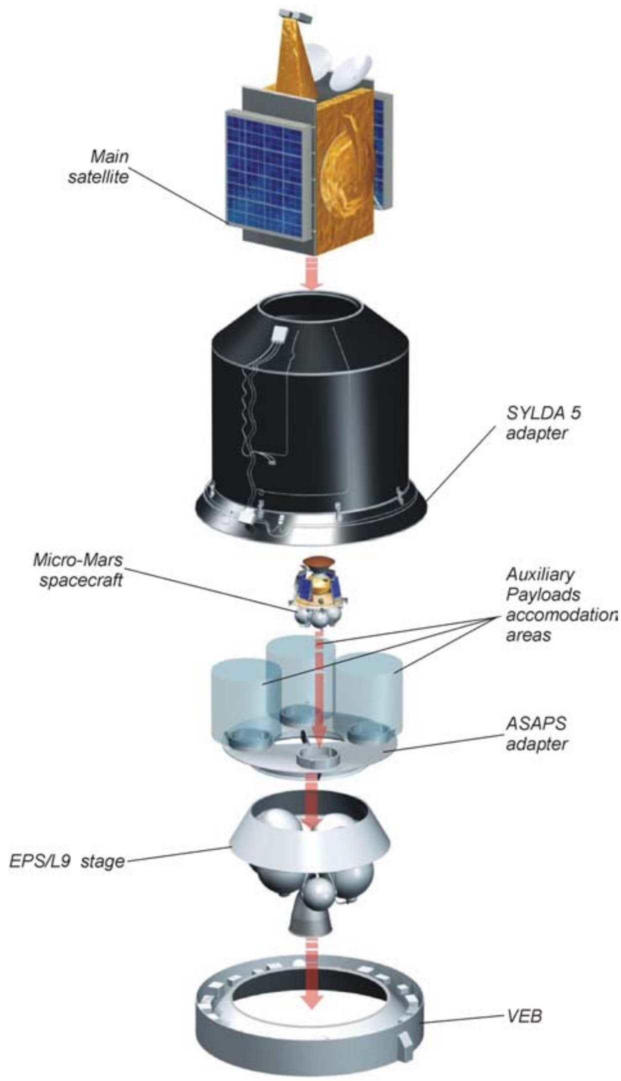

Fig. 8: Launcher Interface

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 then release the capsule and return into the previous orbit. The lander will descend on a ballistic flight trajectory with an entry angle of about -11degrees and a velocity of about 5,000 m/s. The capsule will land in the equator region using a heatshield for the atmospheric deceleration with a maximum of -8 g and a maximum heat flow of about 180 kW/m^2. Followed by a parachute landing phase with a parachute surface of 16 m^2, which will reduce the descend velocity to about 20 m/s. In the last phase until touch-down, the lander will provide an airbag-system to reduce the landing shock for the about 4.2 kg of subsystem avionics and scientific payload instrumentation. The total mass of the lander is about 15 kg. Fig. 9 shows the Micro-Mars Lander in a 3-D view whereas fig. 10 shows the deployed surface station (Small Mars Station).

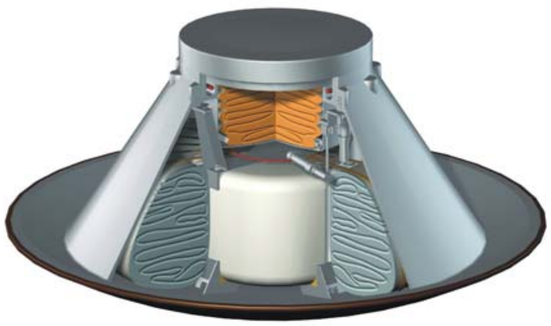

Fig. 9: Micro-Mars Lander in 3-D view

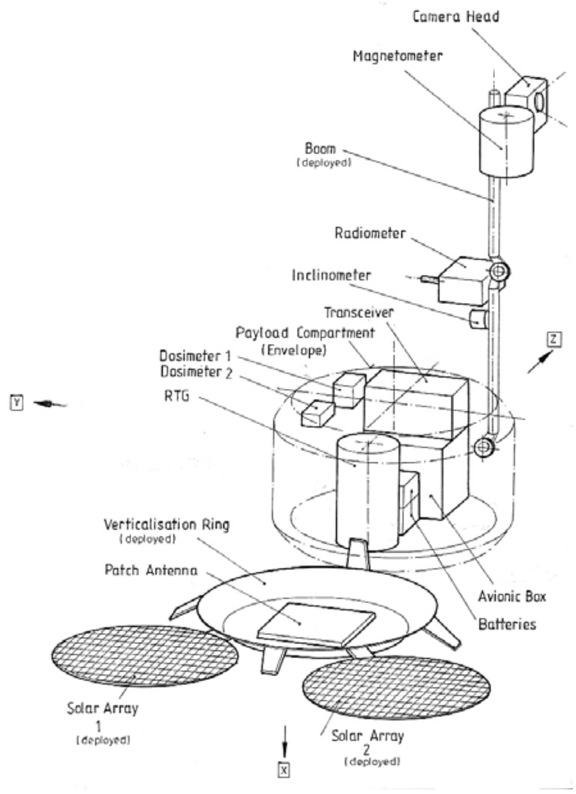

Fig. 10: Small Mars Station

Mission Type	Delta-v [m/s]	Propellant [kg]	Payload [kg]	Remarks
Mars orbiter & lander	2,497	205,0	30,0	Micro-Mars reference mission
Mars data relay	2,400	199,9	35,1	Orbit like reference mission
Mars mapper	2,400	199,9	35,1	Orbit like reference mission
Mars atmospheric orbiter	2,400	199,9	35,1	Orbit like reference mission
Phobos / Deimos lander	2,500	205,0	29,7	
Venus fly-by	1,050	107,4	127,6	
Venus orbiter	2,555	205,0	24,8	1,000km x 20,000 km orbit
Mercury fly-by / hard lander	3,200	205,0	-19,6	**Mission not feasible!**
Moon orbiter / hard lander	1,400	135,6	99,4	200 km circular orbit 0-90°
Near Earth Object (NEO) orbiter / lander	2,480	204,1	30,9	0.8 AU x 1.2 AU orbit assumed
Main belt Asteroid fly-by / hard lander	2,600	205,0	20,9	2.7 AU apohelium assumed
X-ray imaging	1,628	152,2	82,8	400 km x 20,000 km orbit, 50°
Infrared imaging	1,628	152,2	82,8	400 km x 20,000 km orbit, 50°
Magnetosphere science	1,293	127,3	107,7	1,000,000 km circular orbit, 5°
Radio science	1,628	152,2	82,8	400 km x 20,000 km orbit, 50°
Earth data relay	2,568	205,0	23,6	400 km x 20,000 km orbit, 90°
Earth high-resolution imaging	2,568	205,0	23,6	400 km x 20,000 km orbit, 90°
Environmental monitoring satellite	2,568	205,0	23,6	400 km x 20,000 km orbit, 90°

Tab. 5: Multimission capability

Summary

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.

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 Astrium GmbH Bremen.

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