MERCURY SURFACE EXPLORATION SCENARII

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ABSTRACT

Several versions of a Mercury Surface Element, part of the ESA BepiColombo Mercury Cornerstone mission to be launched in 2009, have been studied. An overview of the main requirements, drivers and constraints identified is given in this paper, concerning system design, payload accommodation and deployment, thermal control, lighting, energy and communications issues.

1. INTRODUCTION

A proposal for a mission to the planet Mercury was initially submitted to ESA in 1993, and since that time it has been the object of several investigations by the Agency, the European scientific community, and industrial contractors. The latest version of such mission (the BepiColombo mission) has recently been selected by ESA's Science Programme Committee (SPC) for launch in 2009 as ESA's Fifth Cornerstone. BepiColombo features two spacecraft in orbit around Mercury (a Planetary Orbiter and a Magnetospheric Orbiter), as well as a Surface Element, necessary for the detailed geochemical and geophysical exploration of the surface. Details of the mission are provided in (Ref. 1, 2, 3).

2. MISSION DESIGN

The BepiColombo mission design is based on 3 science elements and 2 propulsion modules. The science elements are the Mercury Planetary Orbiter (MPO), the Mercury Magnetospheric Orbiter (MMO), and the Mercury Surface Element (MSE), depicted in figure 1. The propulsion modules are the Solar Electrical Propulsion Module (SEPM) and the Chemical Propulsion Module (CPM). The MPO is launched first (August 2009), with an SEPM and CPM, on a Soyuz-Fregat vehicle from Baikonur. A composite of the MMO and MSE, with another SEPM and CPM, is then launched within a month on another Soyuz-Fregat. The propulsion modules for the first and second launch are made identical, by taking the worst-case requirements into account, thus offering the advantage of procurement at recurring cost.

The SEPM carries the MMO and MSE composite to the vicinity of Mercury (reached after a 3.5 year cruise), after which it is jettisoned. The CPM is then used for injection of the composite into the highly elliptical, 400 x 12,000 km, Mercury polar orbit required for the MMO operations. Soft landing the MSE requires essentially a flexible, high-performance, liquid-fuel propulsion system, able to bring the spacecraft to zero velocity, in a controlled attitude, in close proximity of its intended landing site. The velocity increment for injection into Mercury orbit is only of the order of 350 m/s, while the combined velocity increment for de-orbit and landing is >4 km/s. It is envisaged to use the CPM as a propulsion system for the descent and landing of the MSE (this becoming the sizing case for the CPM design). The MSE lander (the surface station, including the payload and the power, data and communications subsystems) is a self-contained package, which separates from the integrated CPM in close proximity of the ground.

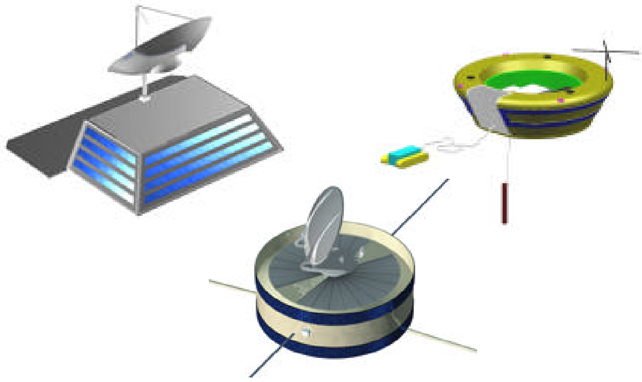

Figure 1. *The BepiColombo science elements, MPO, MMO and MSE*

2.1 Landing Site

The MSE provides Mercury-surface chemical and physical properties measurement for a duration of at least one week. Landing on Mercury shall occur in the latitude band between 84° and 86°, in either hemisphere, with no requirements on longitude from the scientific side. The near-polar location, combined with the epoch of landing, provides a milder environment than a lower-latitude one from the thermal standpoint: the mean surface temperature stays between –50^o and +70^o C. However, the high latitude reduces considerably solar illumination of the surface (typically up to 40% of the surface may be shadowed by local features), with consequent limitations on solar power availability.

2.2 System Design Drivers

The main design drivers for the MSE can be summarised as follows:

- propulsive braking greatly limits the useful payload mass on the surface;
- power and operations (including navigation during landing) require targeting an illuminated area;
- soft landing requires a shock absorption device;
- the thermal environment constrains the landing location and the duration of surface operations;
- the Model Payload requires a compromise between mobility, soil penetration, and avoidance of chemical contamination.

The major constraint on system design has been the need to maximise the useful system mass on the surface of Mercury (scientific instruments, plus supporting deployment devices and system avionics). This is a challenging task, due to the absence of atmosphere on the planet, which forces the adoption of a purely propulsive descent and landing system (which is very mass-demanding).

3. MISSION OVERVIEW

The operations of the MSE mission are described in the following sequence (figures 2 and 3):

- The composite (MMO, MSE, CPM) separates from SEPM. The MSE lander is not active, and the composite is controlled by MMO. The composite mass at this stage is 584 kg for the baseline August 2009 launch window (on the basis of performance specified for the Soyuz/Fregat launcher and SEPM).
- CPM fires (thrust 4 kN, I_{sp} 315 s, Δv = 359 m/s) and injects the composite into orbit around Mercury.
- The composite loiters in a highly elliptical orbit (up to 33 days), waiting for the correct landing conditions (surface illumination).
- CPM and MSE separate from MMO, which starts its scientific orbital mission. The composite mass is 418 kg. MSE takes control of the composite.
- CPM fires (burn time 67 s, Δv = 850 m/s), in proximity of MMO orbit periherm, and injects MSE on a transfer orbit with a periherm of 10 km above one of the polar regions.
- Approximately 75 minutes after transfer orbit injection, CPM fires again (burn time 138 s, Δv = 3464 m/s), and reduces the velocity of MSE to 0 in close proximity of the ground (120 m). The (dry) mass of the composite is reduced to 106 kg.
- MSE is detached from CPM and is separated by 3 small 90 N solid-fuel thrusters (2 s burn time). The CPM crashes into the ground within 8 s of separation. The separated MSE mass is 44 kg.
- The lander inflates its airbag system during its free fall.
- The lander hits the ground (at about 100 m from the CPM crash site) after a 10 s ballistic flight and rebounds several times, increasing its distance from the CPM debris.
- The lander comes to rest. Airbags are separated and deflate. The lander falls to the ground.
- The lander deploys the communications antenna and jettisons the protecting lid of the radiator (if need be).
- The lander deploys the payload and starts operations.
- One of the relay orbiters (MPO or MMO) comes into visibility of the lander and a communications session is started.
- Operations continue for a minimum of 7 days, and can be extended to a maximum of 35 to 70 days (according to the selected landing date and the availability of sunlight on the solar array).
- Mercury night-time sets in, solar power is no longer available, and the MSE mission is terminated.

The main MSE design features of relevance during the critical descent and landing phases are discussed in some detail hereunder.

3.1 Separation and Descent

The AOCS for the autonomous flight of CPM is based on:

- a sensor suite: star tracker, inertial measurement system (3 gyroscopes, 3 accelerometers), optical range/range-rate sensing system;
- actuators: main engine, 4 RCS engines, propulsion drive electronics.

The need to maintain the shock level at landing below limits which are acceptable to the payload imposes the adoption of a Guidance, Navigation & Control system capable of allowing a drastic reduction of the landing speed, and therefore the adoption of an airbag landing system. The proposed range and range-rate sensing system is based on an optical device, tracking features on the ground from one image to the next. This tracking requires modest onboard resources (compared with a radar altimeter), because the inertial measurement system will provide information of the vehicle motion between images, so that a feature can be searched for within a small window of uncertainty. The relative motion of tracked points will reveal the vehicle altitude, knowing the distance between the points from which the images were taken. This distance is not directly known, due to dispersions in onboard knowledge of the velocity, but, when a measured acceleration is applied to the vehicle, the known change in velocity can be used to scale the distance to the ground. Notably, several ground features distributed over the image, depending on the local lighting conditions and surface morphology, are tracked (notice that this requires that approach to landing occurs over a sunlit portion of the Mercury surface). RSS filtering yields an estimate of the average ground plane in the image.

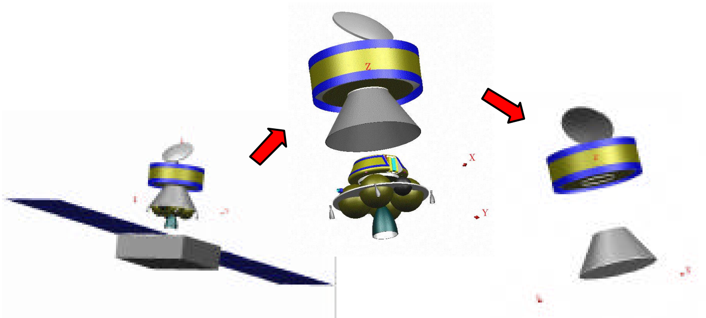

Figure 2. *MMO and MSE deployment sequence: SEPM jettison, separation of MMO from CPM+MSE, jettison of Interface Cone from MMO.*

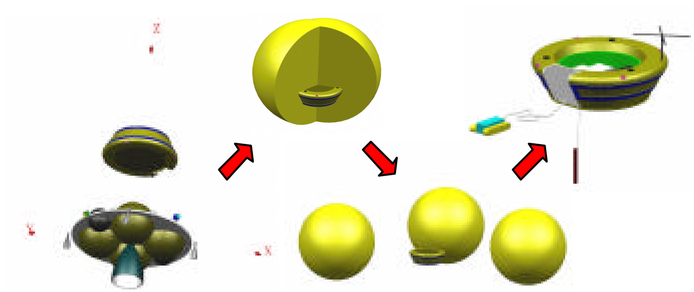

Figure 3. *Separation of MSE from CPM, MSE landing and airbag separation, MSE on ground with deployed payload (thermal protection cover is jettisoned).*

The AOCS star sensor and range sensor are mounted on CPM, to ensure adequate fields of view.

The main CPM engine was selected on the basis of the following criteria:

- high thrust required for descent optimisation (near-optimal in the range 1.5 to 4 kN; 4 kN selected on the basis of commercial availability);

- high I_{sp} required, for mass saving (315 s advertised by manufacturer, improvement to 320 s possible, by nozzle and combustion chamber modifications);
- re-startability (3 burns) required to cover the whole injection, descent and landing sequence;
- no thrust level modulation capability required (the robust landing system, based on airbags, does not require a precision-landing capability).

The consequent high thrust-to-mass ratio results in a highly dynamic system. Reaction control is implemented by four 20 N MMH/NTO thrusters, which are a reasonable compromise between the high thrust level required for torque compensation, and the low thrust level required to provide small minimum-impulse bits during the orbital phases. On the other hand, the short main engine burn times relax requirements on gyroscope drift performance.

The CPM+MSE transfer orbit profile and the final descent are shown in figure 4.

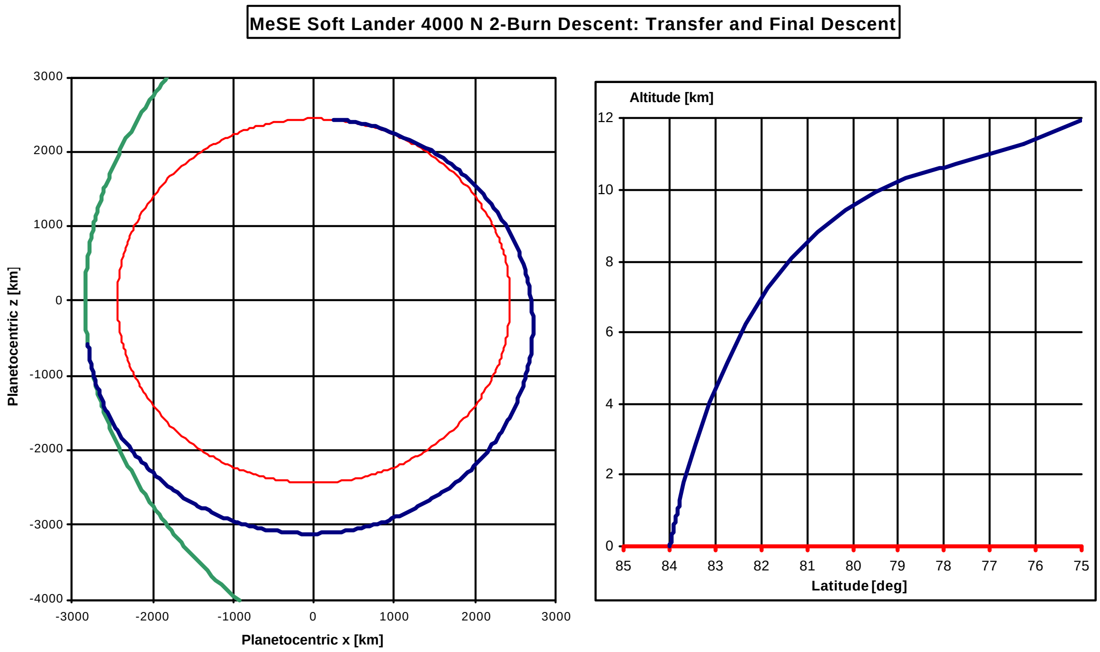

Figure 4. *MSE transfer orbit and final descent altitude profile.*

3.2 Landing

The MSE separates from the CPM, which crashes at >100 m distance from the MSE (thus avoiding chemical contamination of the MSE landing site). Ejection of the lander from CPM is initiated by 3 kick-off springs and is carried out by three 90 N separation thrusters, with 3 axial rails to guide the lander out of CPM during the operation.

Airbags are used as impact attenuator, rather than a crushable structure (which would be more sensitive to the nature of the terrain). A maximum 30m/s touchdown speed is attained after a 120m free fall: this corresponds to the terminal velocity used by parachute-based landing systems under development for Mars, allowing the same airbag design to be used (and thus exploting heritage from the Beagle 2 and NetLander European missions). A maximum (worst-case) impact deceleration of 250 g for 38 ms is experienced under these circumstances. A whole range of ruggedised equipment (payload instruments, power system, avionics) to be used on Mars landers will be available for re-use in the MSE.

The landing system consists of 3 inflatable airbag envelopes (figure 3) tightened together by a rope into a spherical shape, 2.3 m in diameter, with a total volume of 6.4 m^3 (stowed: 0.011 m^3). The envelope material is a 0.2 mm thick rubber-coated aramid fabric; it is inflated by 3 pyrotechnic gas generators (>95% N_2) in about 1 s. When the tightening rope is cut after landing, the 3 envelopes spring back to their natural spherical shape because of their internal pressure, and separate from the lander, which falls to the ground from a height of about 1 m.

3.3 System Accommodation

The MSE has a mass of 44 kg, of which 7 kg have been allocated to the science payload. The overall configuration (fully deployed on the surface) and the mass budget are given in figure 5 and table 1. The subsystem and payload equipment layout is shown in figure 6.

The MSE lander has a conical body (900 mm diameter, 300 mm high), to ensure that it will come to rest on a preferred side. An autonomous attitude recovery capability after landing may be added at a later stage of the design, by the use of a deployable lid. The lander body has a sandwich construction with Glass Reinforced Polyimide Honeycomb. The lander body houses the instruments and ancillary equipment, and provides them with thermal protection. A microrover and a mole, part of the payload, are stowed in a dedicated recess on the side of the lander (figure 6), which may be protected by a flexible sunshade.

In principle, no deployment mechanisms need to be included in the MSE design, except for individual payload components (tethered microrover and mole), therefore increasing the system reliability.

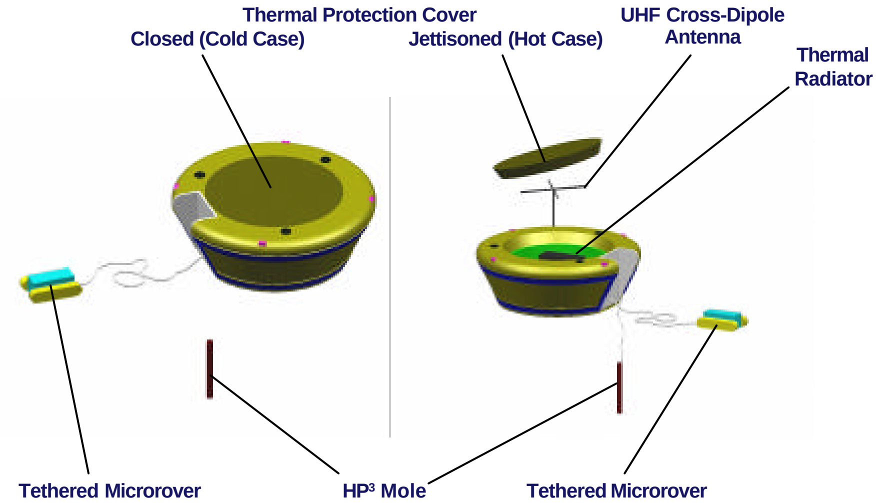

Figure 5. *MSE deployed configuration.*

Item	Total
	[kg]
Structure	15.3
Mechanisms	1.2
Thermal Control	3.2
AOCS	0.6
Data Handling	1.4
TT&C (UHF Relay)	2.5
Power	11.2
Harness	1.4
Payload	7.2
Total	**44.1**

Table 1. *MSE mass budget.*

4. MODEL PAYLOAD

A list of instruments included in the MSE Model Payload is given in table 2. Additional, or alternative instruments may be selected via an Announcement of Opportunity to be issued during 2002.

The overall mass and (minimum) energy requirements for the payload are assumed to be 7 kg (including margin) and 300 Wh. The data volume collected is 75 Mbit for the minimum lifetime of 1 week.

4.1 Heat Flow and Physical Properties Package

Geophysical investigations require that thermal, accelerometric, and densitometric probes be brought in contact with subsurface regions, to a depth of several meters. Therefore, the sensors of the Heat Flow and Physical Properties Package (HP^3) are mounted in a penetrating device, or mole, launched vertically down from a storage tube attached to the lander body. The HP^3 sensors consist of a string of thermistors that can be electrically heated, an accelerometer and a radiation densitometer. HP^3 measures the surface properties such as temperature, thermal conductivity and diffusivity, bulk density and mechanical hardness as a function of depth, down to about 2 to 3 m (maximum 5 m) in a regolith.

4.2 Alpha X-Ray Spectrometer

The Alpha X-Ray Spectrometer (AXS) is a very small device to measure the chemical composition of surface samples and is designed to be transported by a microrover. AXS contains a set of ^{244}Cm sources that emit energetic alpha particles which are backscattered or induce X-ray emission from the sample. The X-ray mode is sensitive to Na, Mg, Al, Si, K, Ca, Fe, P, S, Cl, Ti, Cr, Mn and Ni, the alpha mode to C and O. The sampling depth is about 10 µm and the integration time is 1 to 2 hours per sample. Such an instrument made the first in situ analysis of Martian rocks.

Surface mobility is an obvious requirement for the purpose of geochemical exploration, since selected, differentiated rocks have a much higher scientific yield than the average planet regolith, or than average mixtures of smashed or shattered rocks. A microrover attached to a tether can deploy instruments at selected sites several m away from the lander.

Instrument	Acronym	Deployment	Mass	Average Power
			[kg]	[W]
Heat Flow and Physical Properties Package	HP^3	Mole	0.5	0.3
Alpha X-ray Spectrometer	AXS	Microrover	0.3	1
Descent Camera	CLAM-D	None	0.5	3
Surface Camera	CLAM-S	None	4 x 0.2	3
Magnetometer	MLMAG	None	0.3	0.6
Seismometer	SEISMO	None	0.9	0.6
Mole	MDD		0.4	5
Microrover	MMR		2.0	3

Table 2. *MSE Model Payload instrument summary.*

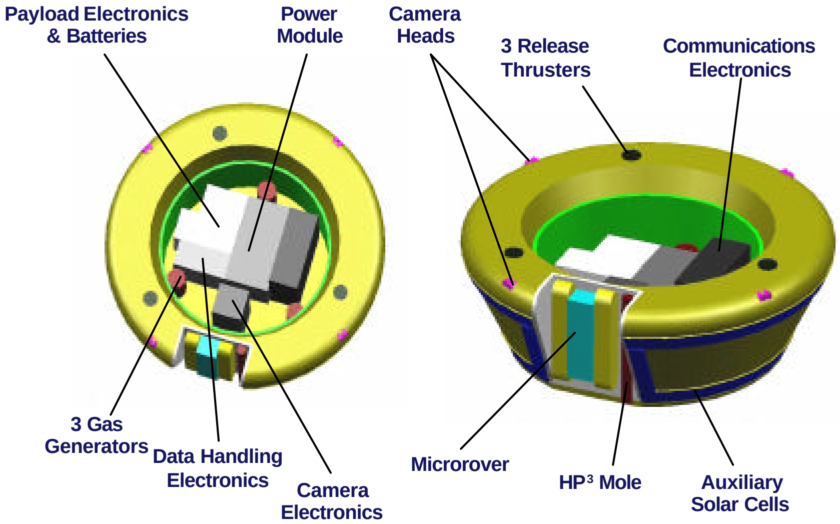

Figure 6. *MSE subsystem and payload equipment layout.*

4.3 Imaging System

The imaging system on MSE includes a descent camera (CLAM-D), a panoramic camera (CLAM-S) to characterise the surface at the landing site and, possibly, a Close-Up Imager (CUI) on the microrover.

The purpose of CLAM-D is to document the approach and landing, and to characterise the structure of the surface. The camera, equipped with a four-position filter wheel, could take (indicatively) 10 images during the descent, the last one as low as 100 m above the surface (just before MSE lander separation from the spent CPM). Communications from the MSE to the MPO are possible during descent and landing at a low data rate only, for the relay of basic housekeeping data: images from CLAM-D are therefore compressed and stored in MSE for later downloading.

CLAM-S consists of 4 pairs of microcameras (figure 6) installed at right angles on the lander upper rim (0.3 m height), each with a 120° field of view. This ensures a full stereoscopic coverage of the 360° field of view around the lander, with the possible exception of one quadrant, depending upon solar elevation (cameras on sunlit side may not be operated). With this arrangement, assuming an angular resolution of 2 mrad/pixel (and a 1024 x 1024 pixel CCD), the resulting ground resolution is, indicatively, 10 mm at 1.2 m from the lander (with 6.7 m^2 area coverage), and 80 mm at 3.4 m (44 m^2 coverage). A higher resolution and wider coverage may eventually be required, either for scientific purposes, or for the navigation of the microrover. Therefore, as an alternative, a telescopic mast, with azimuth and elevation drives, may be used to deploy an optical head. The camera itself would be recessed inside the lander body, for thermal control reasons, and the images relayed to it by mirror reflection or fiber optics. Flashlights may be added for imaging shadowed areas.

The CUI mounted on the microrover would perform a close examination of rocks and regolith.

4.4 Magnetometer

A magnetometer on the lander (MLMAG) will characterise the magnetic properties of the surface and provide a reference for models of the intrinsic planetary field. Very simple sensors could measure the electric conductivities of the ground and of the exosphere, which are critical parameters for understanding the current pattern that shape the magnetosphere. It will be possible to derive the electrical conductivity of the ground by simultaneously recording the magnetic field fluctuations on MMO and MSE.

The MLMAG is mounted on the lander body. However, should magnetometric measurements need deployment of sensors to some distance from the bulk of the lander body, a boom (indicatively 0.5m long) may be used to deploy MLMAG.

4.5 Seismometer

A seismometer (SEISMO) is tentatively considered because it would significantly enhance the science return, especially if the lifetime of MSE could be increased beyond 1 week. This instrument would record tidal deformations and sound waves excited by quakes in the crust or in the mantle of the planet. The SEISMO sensor should be brought into contact with the Mercury soil.

5. THERMAL CONTROL

The thermal environment on the surface of Mercury is extreme, even in the polar regions which will be targeted by the MSE lander. A ground temperature variation between –50^o C and +70^o C occurs over a few days (figure 7), while sunlight may rise seasonally up to 10 times the one experienced in Earth orbit. This has a very significant impact on system design.

To cope with the low-temperature environment in a shadowed area, the MSE is fully insulated by a 30mm thick layer of carbon aerogel on the inner side of the outer conical wall. The MSE base is made from foam-core sandwich, with high resistance to impact and low thermal conductivity. Electronics are connected to the base via low-conductivity glass fibre struts. Should the landing occur in sunlight, a jettisonable cover would be expelled to enable a topside radiator to dump waste heat from the MSE. The radiator is slightly recessed inside the MSE structure and is protected from the Sun for elevations of up to 20^o. Thermal control of the lander in flight, from MMO separation (lander avionics activation) to landing (lid opening), relies on thermal inertia.

For additional protection from the Sun (but at the expense of added mass and complexity), MSE studies have considered using a deployable sunshade, unfolded and straightened by pre-stressed tapes. In case a deployable solar array were adopted, the sunshade (0.2 m^2, 50 µm-thick, white-painted titanium foil) may hang from the edge of the array, and rotate with it to follow the sun.

The Mercury surface thermal environment is especially severe for small, exposed payload appendages, deployment devices and probes at some distance from their mothercraft. The need to provide a low-temperature heat sink to sensors (typically: <0^o C for optical sensors, <-20^o C for geochemical probes) is particularly critical, if these are installed on a small-size, small-mass mobile deployment device. Thermal design of such items should, in principle, be able to cope with the following extremes:

- a cold case corresponding to a landing in permanent darkness (e.g. inside a crater or in the shadow of a rock or ridge), with no solar flux, and a surface temperature locally as low as -170^o C.
- a hot case corresponding to landing in a sunlit area at >84^o latitude, with a solar flux of up to 14.5 kW/m^2 (Mercury perihelion) with the Sun at very low elevation (<6^o), and a surface temperature as high as +70^o C.

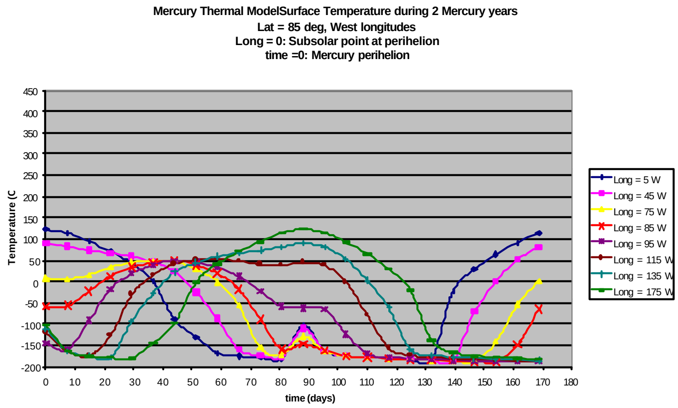

Figure 7. *Variation of surface temperature at high latitude.*

6. ENERGY

Beside the severe mass constraint, which will force the widespread adoption of efficient miniaturised systems, the payload delivered to Mercury surface will be faced with very limited available system resources (in terms of energy, and of retrievable data volume), and will therefore require a significant degree of optimisation. An energy-limited MSE design has been adopted, with a primary battery of high energy density (1.7 kWh available energy). This approach allows operations to be carried out even in a completely shadowed area for about one week; the minimum mission therefore does not rely on the availability of solar energy, and has a high probability of success. An energy density of 250 to 300 Wh/kg is sought, the most promising technologies being Li-Thionyl Chloride ($Li/SOCl_2$), Li-Sulfuryl Chloride (Li/SO_2Cl_2) and Li-Manganese Oxide (Li/MnO_2).

An indicative a llocation of the available energy is: 300 Wh to payload, 800 Wh to OBDH and power, 500 Wh to TT&C. The payload energy allocation may be further broken down as follows:

- 120 Wh to HP^3. This corresponds to 9 hours of mole penetration at 5 W (i.e. a maximum penetration depth of 5 m, assuming 1,700 kg/m^3 regolith) and 73 hours of HP^3 instruments operations at 1 W.
- 150 Wh to AXS and CUI. This corresponds to 66 hours of microrover operations at 2 W (i.e. up to 330 m traverse at 5 m/h) and 6 hours of AXS operations at 3 W (1 hour at each of 6 targets).
- 30 Wh to MLMAG, corresponding to 100 hours of operations at 0.3 W.

Alternative studies have envisaged the adoption of a solar array deployed on a mast (total height >1 m above ground), and this is regarded as an option (with impact on system mass and reliability). Pointing and power transfer is performed by a solar array drive at the base of the mast. Array pointing is required to be offset at an angle of up to 70° away from the Sun direction, for thermal reasons (maximum cell temperature of 130^o C). 50% of the array surface is covered with OSRs.

Sun acquisition is done by solar array current and temperature sensors reading. GaAs solar cells are selected for higher efficiency.

7. LIGHTING

A consequence of the landing in a polar region (>84^{o} latitude) will be the extremely variable lighting conditions, with extended portions of the surface shrouded in darkness by any small surface obstacle. The probability of landing in shadow at a latitude of around 85° may be high (indicatively, 40% of the terrain may be expected to be in shadow), although Mercury topography in these regions is poorly known and terrain models used may yield widely different results.

An optional small solar generator (strings of solar cells fixed to the lateral surface of MSE, for about 0.5 kg additional mass) could nevertheless be added and used to feed directly the load, or part of it, thus allowing extended surface operations.

8. RADIATION

At Mercury, the main radiation threat consists of solar energetic particles, which have a flux density about 10 fold with respect to 1 AU (figure 8). In addition, BepiColombo cruise and operations at Mercury will take place at a time close to solar maximum (occurring in 2011). The radiation effects on the spacecraft at Mercury include total dose, solar cell degradation and single-event effects. The total dose effects on electronic parts and solar cell degradation are comparable to scientific missions in Earth orbit (e.g. the 10-year XMM mission), provided 4 mm Al equivalent shielding thickness is implemented (figure 9). Single-Event Effects (SEE) due to protons pose problems in solar maximum periods, as follows:

- Memories: SEE during solar proton events are about a factor of 10 higher than for missions at 1 AU, therefore EDAC (Error Detection and Correction) or other methods to reduce the SEU rate are necessary. It is expected that the SEE rate from galactic cosmic rays will be less than at 1 AU due to the attenuation of the GCR flux by the solar wind. The maximum SEU rate is about 10^{-10} SEU/(bit day); for a comparison, the XMM requirement is 10^{-11} SEU/(bit day) for memories. Proton effects dominate in Mercury orbit and have to be evaluated in detail.
- Linear parts: linear parts on Mercury spacecraft suffer 10-fold the SEU effects of 1 AU missions. In periods of solar events, the SEU rate can reach up to 1,000/day. Galactic Cosmic Ray (GCR) effects remain constant.
- Proton-induced SEU: the effects of protons on optocouplers are dramatic. Damage due to low energy protons (0.05 MeV) may be critical, but shielding by optics is sufficient to avoid this problem (no direct radiation exposure to payload system CCD). Effects of high-energy proton fluxes on CCD are unclear; research on CMOS sensor technology is in progress for suitable space radiation applications.

Measures for minimising the radiation effects can be taken at system level (structures), at unit level (housing, electronics), and at part level. A reduction of solar proton induced effects can be expected if structure materials have a low yield for secondary effects, e. g. neutrons. Materials with high carbon content, e. g. CFRP, C/SiC, B_4C, are recommended. The current design assumption is a CFRP structure with low-Z material. At unit level, housing must be made from high and low-Z materials (second-layer shielding), while electronics must use SEU-safe designs. At parts level, spot-shielding is to be used, if necessary, to minimise mass, in combination with radiation-hardened, i.e. single-event robust parts (SOI and SOS technology).

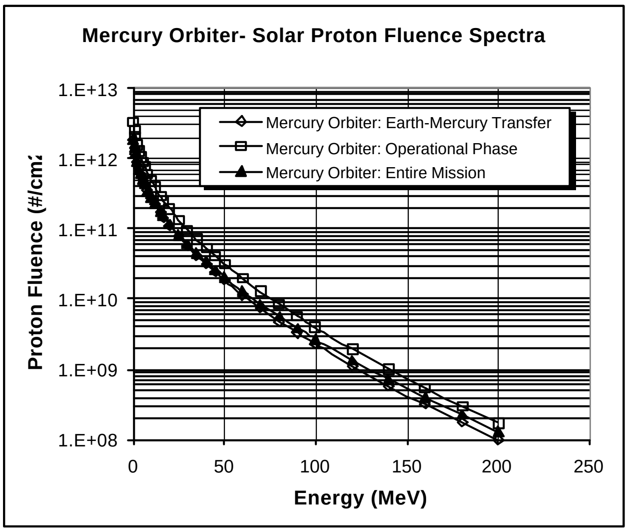

Figure 8. *Mercury mission proton fluence.*

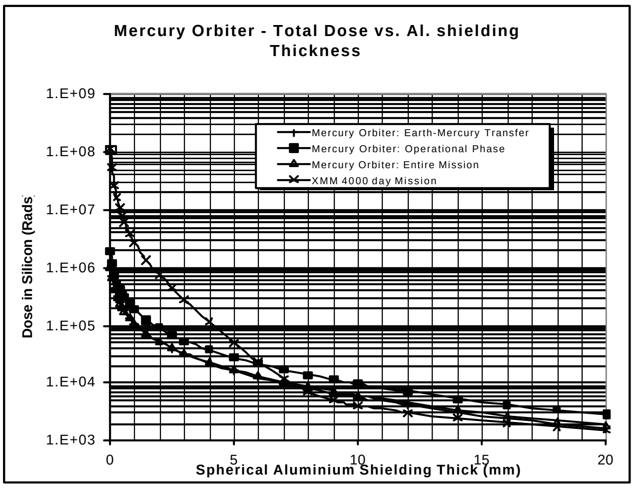

Figure 9. *Total dose comparison, Mercury versus XMM.*

9. COMMUNICATIONS

Limitations on communications between Earth and the deployed MSE payload will be caused by the low available data rate, and by visibility windows (contact may be restricted to as little as <10 minutes every 9.3 hours). This will impose a high degree of autonomy to be built into the payload systems.

The MSE science data are stored in a mass memory and transmitted to one of the companion orbiters at each overhead pass. Either the MPO or the MMO can be used as relay. A mean usable data rate of 4 to 9 kbit/s is provided by the UHF telemetry system (0.5 to 1 W RF). A self-deployed cross-dipole antenna is mounted on top of the lander body.

In case the MMO is used as relay, the payload can use a total of 75 Mbit for the (minimum) 7 days of operations. This corresponds to 18 contact periods of 480 s, occurring every 9.3 hours. Indicatively, 68 Mbit may be used for imagery (compressed), in order to retrieve:

- high-resolution and colour images taken by the CLAM-D during descent (24 to 48 Mbit, e.g. 12 to 24 high-resolution images x 1 Mbit, plus 48 to 96 colour images x 0.25 Mbit, binned 2x2).
- stereo pairs of the surface from the CLAM-S (16 to 32 Mbit e.g. 8 to 16 stereo pairs x 2 Mbit).
- close-up images from the CUI on the microrover (e.g. 4 to 8 images x 1 Mbit).

The rest of the data volume is shared among the HP^3 (e.g. 2 Mbit from 73 hours of operations), the AXS (2 Mbit from 6 hours of measurements), and the MLMAG (3 Mbit, 100 hours at 6.8 bit/s). Nearly twice this total data volume (138 Mb) is provided in the case of MSE to MPO link (4 times more frequent contact periods due to the lower near-circular orbit). In addition, by appropriately phasing the MSE descent with the MPO orbit, it is possible to ensure visibility of the MPO from the MSE for at least a portion of the descent trajectory, so that telemetry of vital parameters from the MSE at a low data rate can be maintained until landing.

10. REFERENCES

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