SCI-AP Preprints October 2004

Recent papers related to studies initiated by the Planetary Explorations Studies Section of the Science Payloads and Advanced Concepts Office

International Astronautical Congress 2004
Vancouver, Canada, 4-8 October 2004

- ESA's Technology Reference Missions
- Small Body Sample Return to Deimos
- System concepts and enabling technologies for an ESA low-cost mission to Jupiter/Europa
- The Interstellar Heliopause Probe

- Analysis of a Solar Sail Mercury Sample Return Mission

2nd international Planetary Probe Workshop
NASA Ames Conference Centre, Moffet Field, USA, 23-26 August 2004

- ESA Venus Entry Probe Study

11th SPIE International Symposium on Remote Sensing
Gran Canaria, Spain, 13-16 September

- On the study of highly integrated payload architectures for future planetary missions

SCI-AP

European Space Agency
Agence spatiale européenne

Science Payload and Advanced Concepts Office (SCI-A)
Planetary Exploration Studies Section (SCI-AP)
2200 AG Noordwijk, Postbus 299
The Netherlands

European Space Agency
Agence spatiale européenne

SCI-AP

Science Payload and Advanced Concepts Office (SCI-A)
Planetary Exploration Studies Section (SCI-AP)
2200 AG Noordwijk, Postbus 299
The Netherlands

International Astronautical Congress 2004

Vancouver, Canada,

4-8 October 2004

SCI-AP

European Space Agency
Agence spatiale européenne

Science Payload and Advanced Concepts Office (SCI-A)
Planetary Exploration Studies Section (SCI-AP)
2200 AG Noordwijk, Postbus 299
The Netherlands

TECHNOLOGY REFERENCE STUDIES

A. Lyngvi, P. Falkner, A. Atzei, D. Renton, M. L. v. d. Berg and A. Peacock
Science Payload & Advanced Concepts Office
European Space Agency, ESTEC, The Netherlands
aleksander.lyngvi@esa.int, pfalkner@esa.int, aatzei@esa.int, drenton@esa.int, mvdberg@rssd.esa.int, apeacock@esa.int

ABSTRACT

ESA's Science Payload & Advanced Concepts Office (SCI-A) has introduced Technology Reference Studies (TRS) to focus the development of strategically important technologies of likely relevance to future science missions. This is accomplished through the study of several technologically demanding and scientifically interesting missions, which are not part of the ESA science programme. Presently the Planetary Exploration Studies Section of SCI-A is studying four TRS; the Venus Entry Probe, the Jovian Minisat Explorer, the Deimos Sample Return and the Interstellar Heliopause Probe. These TRS cover a wide range of mission profiles in the solar system with an even wider range of strategic important technologies.

All TRS mission profiles are based on small satellites, with miniaturized highly integrated payload suites, launched on Soyuz Fregat-2B.

This paper describes the current four TRS in further detail and shows how these missions are used to identify and prepare the development of enabling technologies.

1. INTRODUCTION

Most science missions are in many respects technologically very challenging. It is very important to define and prepare critical technologies far in advance to ensure that they are developed in a timely manner and that associated cost, risk and feasibility of potential future mission concepts can be estimated properly. Technology Reference Studies (TRS) are set up to provide a set of realistic requirements for these technology developments far before specific science missions get proposed by the scientific community.

2. TECHNOLOGY REFERENCE STUDIES

The TRS[1] are chosen to cover a wide range of different scientific topics ranging from astrophysics, fundamental physics to planetary exploration. Currently four mission concepts are under study in the field of planetary exploration: The Venus Entry Probe (VEP), the Deimos Sample Return, (DSR), the Jovian Minisat Explorer (JME) and the Interstellar Heliopause Probe (IHP).

These four studies cover a variety of mission profiles with very different technological challenges. Through their study a set of detailed requirements for technology development activities can be determined.

The TRS are a tool to focus technology development activities and to define their required environmental conditions, but they are not part of ESA's science mission programme. The current four planetary TRS have been carefully selected to address a wide range of technologies that have to be applicable to many other scientific mission profiles as well. For instance, the technological challenges for the VEP are not only applicable to an in-situ atmospheric mission to Venus, but in many respects also to missions to other planetary bodies with dense atmospheres, such as Titan. The technologies for the JME apply to several of Jupiter moons not only to Europa and are relevant for many outer planets missions as well. The technologies for the DSR enable return of samples from different low gravity bodies and the technologies developed for the IHP will also apply to

missions to the outer planets, as they share similar environmental and technical constraints.
One of the main goals of the TRS is helping to reduce the cost of future science missions. The studies are based on low cost spacecraft, allowing for a phased exploration strategy with multiple small spacecraft and lower overall risk compared to a single high resource mission approach. The low cost approach is ensured by carefully chosen constraints on the mission concept.

The TRS must be compatible with a single Soyuz Fregat 2B launch vehicle launched from Kourou. Envisaged technologies should have a technology readiness level compatible with a launch in the 2010-2020 timeframe. This is to ensure that only realistic mission scenarios are studied and that the technology requirements can be properly defined.

3. VENUS ENTRY PROBE

More than twenty missions have been flown to Venus so far, including fly-bys, orbiters, and in-situ probes. These past missions have provided a basic description of the planet, its atmosphere and ionosphere as well as a complete mapping of the surface by radar. The upcoming comprehensive planetary orbiters, ESA's Venus Express (launch 2005) [2] and Planet-C from JAXA (launch 2007)[3], will further enrich our knowledge of the planet. These satellite observatories will perform an extensive survey of the atmosphere and the plasma environment, thus practically completing the global exploration of Venus from orbit. For the next phase, detailed in-situ exploration will be required, expanding upon the successful Venera atmospheric and landing probes (1967 - 1981), the Pioneer Venus 2 probes (1978), and the VEGA balloons (1985).

The objective of the VEP[4] is to establish a feasible mission profile for a low cost in-situ exploration of the atmosphere of Venus by employing an aerobot and several atmospheric microprobes. Typical scientific questions that the VEP aims to answer are:

- How and why has the atmosphere evolved so differently compared to Earth?
- What are the source(s) of the present atmosphere and what role do minor atmospheric constituents play in the atmospheric chemistry and greenhouse effect?
- What are the dynamics of the Venus atmosphere and what are the driving factors behind these?
- What is the size distribution of cloud aerosols, their physical and chemical composition and what is the aerosol density variation in the vertical profile?
- What is the history of the resurfacing and volcanism?

The mission profile consists of a pair of small satellites and an aerobot that drops several microprobes during cruise phase. In this profile the VEP composite is launched into a direct Venus trajectory and enters a highly elliptical Venus orbit (250 km x 66 000 km) after 120 to 160 days. The Venus Polar Orbiter (VPO) will subsequently be lowered into a polar 2000 km x 6000 km orbit where it will perform remote sensing primarily dedicated to support the in-situ atmospheric measurements by the aerobot and to address the global atmospheric scientific objectives. The Venus Elliptical Orbiter (VEO) will stay in the highly elliptical orbit until the entry probe is released. Then the VEO will decrease the apoapsis in a range from 20000 to 7500 km where it will perform radar measurements of the planet. The entry probe will deploy the aerobot (Figure 1) that will float in the middle cloud layer of Venus where it will perform in-situ science measurements. The deployed microprobes will perform simple measurements during the descent.

Figure 1: The Venus Aerobot

Several challenges have already been identified during the ongoing VEP study. The entry, descent and deployment scenario is a very critical issue as specific subsystems for the entry vehicle are not available and have to be developed. The current baseline for the entry probe is steep entry with a 45° sphere-cone aeroshell (Figure 2). The entry angle is limited to 30-40°, constrained by the maximum allowed peak acceleration of 200 g for the payload. The peak heat flux is around 20 MW/m^2, which requires a dedicated heat shield development and qualification effort. Just above 1.5 Mach a disk-gap-band parachute will be deployed by a pyrotechnic mortar, which then will slow the probe down to a velocity of around 20 m/s where the balloon will be deployed.

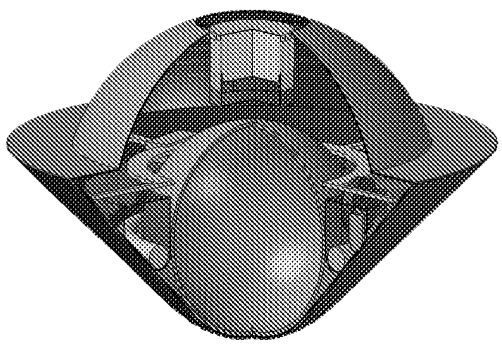


Figure 2: The Venus Entry Probe

The aim of the aerobot is to circumnavigate Venus twice, which will require a lifetime of at least 14 days. For such a long duration flight, an overpressure balloon with Hydrogen gas is considered the most suitable. Microprobes are released to compensate for gas leakage and to perform measurements during decent in the Venusian atmosphere. Additionally, gas release mechanisms and gas replenishment systems are also being considered in order to provide the required mission lifetime. The balloon envelope material needs to have an extremely low leakage rate, and will possibly employ welded seams.

The gondola has a highly miniaturized payload package with an extremely low average power demand. Power is provided by amorphous-silicon solar cells, which are mounted on the gondola surfaces.

The microprobes require substantial development, as they should be limited to around 120 g to meet the stringent mass requirements of the aerobot. One of the key technical challenges of the microprobes is the miniaturized localization and communication subsystem, currently subject to an ESA technology development activity provided by Qinetiq[5].

4. DEIMOS SAMPLE RETURN

During Deimos and Phobos' presence in Mars orbit, ejecta material from all over the planet's surface has accreted onto the two moons during different eras. Modelling suggest that approximately 10% of the upper regolith material on Deimos, likely originated from Mars[6]. This Mars component generally consists of Noachian basin forming (4.6-3.8 billion years ago) and late heavy bombardment impacts material (4.0-3.8 billion years ago).

Believed to be similar to fossils, asteroids retain some records of the formation of the solar system, making them attractive targets for sample return missions. Deimos is smaller than Phobos, with a gravity less than 0.1 % that of Earth. It is also less irregular in shape than Phobos and has a smoother appearance due to partial filling of some of its craters. These factors, along with Deimos' larger orbit, make it the more attractive target for a dedicated TRS.

The DSR[7] (Figure 3) mission profile is defined for returning a 1 kg sample of Deimos regolith back to Earth. The returned sample will provide information about two different solar system bodies, a D-type asteroid, Deimos, and the planet Mars.

The DSR is launched in the current mission profile into a highly elliptical Earth orbit before transfer to Mars. The DSR mission profile assumes an insertion into a 500 km x 100 000 km Mars orbit before the orbit is circularised to obtain co-orbit with Deimos at approximately 20 069 km. The orbit will be slightly different from that of Deimos' to allow for observations of the body before landing and sampling. After sampling, the DSR spacecraft will return to the same highly elliptical Mars orbit from where the DSR will do the Mars Earth transfer followed by a direct entry at Earth return.

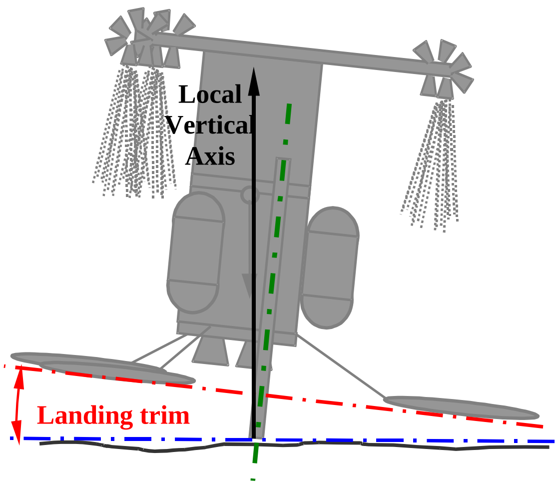


Figure 3: DSR lander

The current mission profile requires a total Delta-V of approximately 2.7 to 3.3 km/s depending on the launch date, with several launch opportunities in the 2010 to 2020 timeframe.

The DSR is providing the background and detailed requirements for low gravity body sample return missions. Several key technologies have already been identified and defined for future development.

The sampling mechanism is of prime importance. Currently, the most promising alternative for the sampling mechanism is a touch-and-go concept, in which the spacecraft only briefly touches the surface while it collects the sample. This sampling method has lower complexity compared to most other alternatives, such as a robotic arm or mole, where landing and anchoring of the spacecraft is required.

A high degree of autonomy is required during the sampling manoeuvres. Due to the communication delay between Earth and Deimos, a highly autonomous guidance, navigation and control system is required to guide the spacecraft during its approach to the surface, sample collection and return to orbit, without interaction from Earth mission control.

The Earth return vehicle also requires substantial development. Several studies have already been performed on such systems mostly in the frame of Mars sample return mission scenarios. DSR greatly benefits from these studies.

An additional challenge is given due to the required planetary protection. The contamination chain from sample collection must be broken to ensure cleanliness of the re-entry vehicle and the sample canister must remain intact during re-entry and has to survive any kind of impact scenario to prevent Earth contamination. Furthermore the sample integrity of the canister must be guaranteed during all phases of the transfer back to Earth.

5. JOVIAN MINISAT EXPLORER

Until now, a limited number of missions have visited the Jovian system: Pioneers 10 and 11 were the first, providing information on the Jovian radiation and magnetosphere in the early 1970s, followed by the Voyagers 1 and 2 at the end of the same decade, which provided multi-band imaging, as well as radiation and atmospheric observations of Jupiter and the Galilean moons.

Ulysses was the first spacecraft to visit Jupiter (1992) since the Voyager missions in the 1970s, since it used a Jupiter gravity assist to swing out of the ecliptic plane towards an orbit around the poles of the Sun. Its visit of Jupiter supplied valuable information on the Jovian radiation and magnetic environment. The last mission focussing on Jupiter was Galileo; it was launched in 1989 and has just ended its mission after being deliberately targeted into the Jupiter atmosphere. This spacecraft provided the most extensive study of the Jovian system until now, including in-situ measurements of Jupiter’s atmosphere by means of an atmospheric probe. Meanwhile Cassini has delivered on its way to Saturn additional data during its Jupiter fly-by in December 2000.

The emphasis of the JME[8] is on the remote sensing of Europa, since it is one of the few places where liquid water may be found in the solar system, making it one of the prime candidates for the search for life outside Earth. The scientific objective of the JME is to perform detailed exploration of surface and subsurface of Europa with remote sensing instrumentation onboard of the orbiter and potentially additional deployment of a microprobe for in-situ analysis on the surface of the icy moon. As the orbit lifetime of JME is strongly limited by perturbations of Jupiters immense gravity, the science operation time once in Europa orbit is limited to 60 days, before the spacecraft will impact the Europa surface.

The current scenario foresees two small spacecraft, the Jovian Relay Spacecraft (JRS) and the Jovian Europa Orbiter (JEO) with 483 kg and 311 kg, dry mass respectively. The JRS will act as a relay satellite in a highly elliptical orbit around Jupiter, outside the high radiation zones, while the JEO will orbit Europa (Figure 4). The relay spacecraft will carry all subsystems that are not directly required for the Europa exploration. It will be subjected to less radiation than the Europa orbiter, carrying the communication system for the data and command link between Earth and the JEO, data processing and data storage units as well as a small, highly integrated scientific payload suite dedicated to explore the Jovian system. The Europa orbiter includes a highly integrated remote sensing payload suite, a communication system for communications with the JRS and Earth and potentially a high-velocity penetrating microprobe to allow for an in-situ investigation on the surface of Europa.

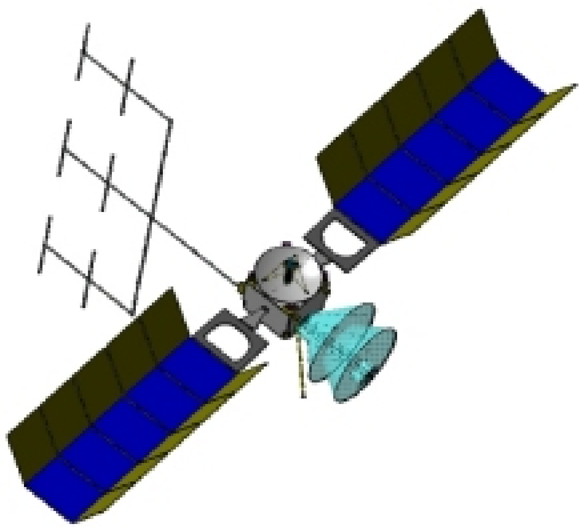


Figure 4: The Jupiter Europa Orbiter

One of the biggest challenges that JME will face is the extreme radiation environment at Jupiter and Europa. The spacecraft electronics need to be protected against radiation levels in excess of 5 Mrad (after 4 mm Al shielding). A combination of radiation hardened electronics in class of 1 Mrad, special adapted spacecraft subsystems and additional extensive shielding is required.

A specific constraint set for the study was that power generation onboard should be performed by non-nuclear methods. The solar power generators have to be designed for 1/25 of the solar flux at Earth using specific adapted GaAs triple-junction Low Intensity Low Temperature (LILT) cells, which require potentially costly development. In order to increase the efficiency of the solar power generators, solar concentrators are foreseen.

Also the communication system requires development to perform deep space inter-satellite links between JRS and JEO in both X- and Ka-band at high data rates (1 Mbps) and allow for communication with Earth. Current available systems do not provide these capabilities under the harsh radiation environment.

The long mission duration, the hostile environment and far distance from Earth ask for a highly autonomous mission, with the benefit of reduced manpower needed to operate the spacecraft. Additional autonomy is required for the commissioning and operational phase of the instruments. The commissioning phase must be very short, because of the orbit lifetime restriction, to allow for a meaningful science operation phase.

It is not possible to receive major parts of the science data in real time due to the limited communication opportunities with Earth. JEO science data are first transmitted to JRS and stored there, before transfer to Earth takes place within a one-year period. Only a small part of JEO data can be transmitted directly to Earth in almost real-time within the 60-day science operation phase and before JEO impacts onto the surface of Europa.

A high-speed hard penetrating microprobe is part of the mission profile requiring very challenging technology development for this system. The high velocity impact (in the order of several hundreds of meter per second) will require materials and subsystems capable of withstanding very high impact shocks and g-loads to guarantee operation of instruments and communication equipment during and after impact.

JEO will impact Europa after the science phase, imposing strict COSPAR planetary protection requirements to the spacecraft and its subsystems. Limitations on material selection and increased complexity and cost during design, manufacturing and integration phase are an unavoidable consequence. In-flight decontamination by the severe radiation in the Jovian system must be exploited as much as possible to relax some of the planetary protection requirements.

6. INTERSTELLAR HELIOPAUSE PROBE

The heliosphere is a plasma bubble blown up by the solar wind into the local interstellar medium. Its droplet shape results from the relative motion of the sun and the heliosphere. The termination shock marks the boundary between the interstellar medium and the heliosphere and is believed to be at a distance of 80-100 AU from the Sun[9]. This interface region is of particular interest for a mission to the interstellar medium and hence is the primary target for the IHP[10]. Key scientific questions to be answered are:

- *What is the nature of the interstellar medium?*

- *How does the interstellar medium affect the solar system?*

- *How does the solar system impact the interstellar medium?*

In order to investigate interstellar medium the IHP has to reach a distance of 200 AU in the direction of the Heliosphere nose, which is located at 7.5° latitude and 254.5° longitude in the ecliptic coordinate frame. A maximum of 25 years transfer time is foreseen.

The IHP requires extensive Delta-V to reach the necessary solar system escape velocity of approximately 10 AU/year. Solar sailing has proven to be the only feasible solution for IHP under the given low cost TRS constraints**.**

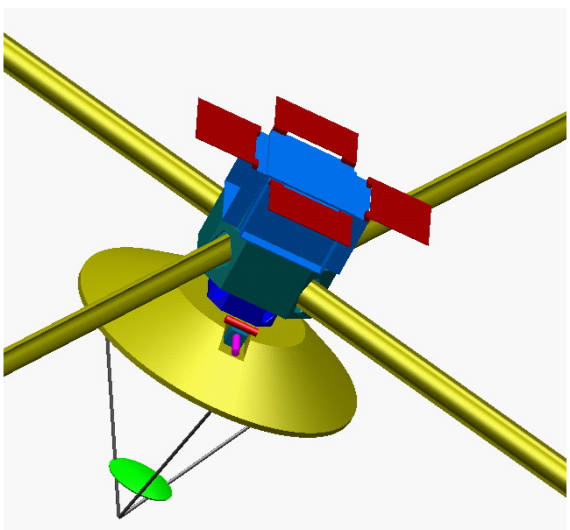


Figure 5: IHP with solar sail booms deployed

Solar sails utilize the photons emitted by the Sun to accelerate the spacecraft. The achieved acceleration in the order of few mm/s^2 is very low and strongly dependent on the distance from the Sun. A close approach to the Sun is required to obtain higher acceleration and hence to achieve the required high escape velocity. Thermal constraints on the sail, booms and spacecraft bus limit the closest distance to around 0.25 AU. The solar sail is jettisoned at 5 AU after an acceleration phase of around 5 years.

Solar sails require very thin and low mass sail materials with high optical reflectivity, specific thermal properties and additional lightweight booms and deployment mechanism. Even the smallest spinning disk sail requires a sail area of 50 000 m^2 for the IHP.

The large extension poses great challenges on storage and deployment of the sail, its supporting structures and on the Attitude Determination and Control System (ADCS) during and after deployment. Possible ADCS solutions are a gimballed boom between sail structure and spacecraft bus, or tip vanes or micro thrusters on sail structures. A suitable solution for this system must be developed and demonstrated in space in order to enable such a challenging mission concept.

The preferred solution for IHP is yet to be decided, however the solar sail will probably be a spinning disc or a square sail, rigidized with booms. The overall sail system mass must remain very low in the order of maximum 200 kg to obtain the required characteristic acceleration of 1 mm/s^2. The deployed booms are limited to 100 g/m specific mass and being able to withstand very high thermal fluxes due to the close approach to the sun.

A low mass mechanism is required that can safely jettison the sail from the spacecraft after 5 years of sailing with minimum risk of collision between the extended sail structure and separated spacecraft

Beyond the orbit of Jupiter the use of solar energy is very inefficient due to the low solar flux. The only real alternative for power generation so far is the use of nuclear energy. The sail size of the IHP is highly dependent on the overall system mass and hence also very sensitive to the mass of the power system. IHP and similar outer planets missions require European technology developments for radioisotope power generation and in particular in the field of thermal to electrical energy conversion.

The IHP communication system will be limited to an average downlink data rate of around 200 bps at 200 AU. Currently RF and optical communication are being considered. For the optical communication issues like the acquisition strategy, lightweight components and laser lifetime must be solved. For an RF system other challenges exists. A Ka band RF system with 80 Watts transmitted RF power requires an antenna of approximately 2.5 meters in diameter. The typical mass of the RF systems with current technology exceeds 60 kg, where the antenna structure forms a large part of the total mass. A significant mass reduction of the RF communication system is of great importance to the IHP and also for low cost outer planets missions. Development of lightweight antennas and highly efficient travelling wave tube amplifiers and solid-state RF amplifiers are required.

The design lifetime of the IHP must be more than 25 years. The consequences to all subsystems, components and materials must be evaluated in detail and specific test procedures must be developed. IHP must be highly autonomous with self-maintenance capabilities.

7. HIGHLY INTEGRATED PAYLOAD SUITES

Small spacecraft can accommodate only smaller payload masses. The Highly Integrated Payload Suite (HIPS)[11] approach is introduced to strongly reduce the payload resources requirements while fulfilling the scientific requirements of a specific mission. The payload is integrated as much as possible to share common functionalities like data processing, power supply, thermal and environmental control, between the instruments. Sharing of structures, optical benches, baffles, optics as far as possible within the physical limits is envisaged. Optimized payload power supply give great reductions in mass compared to individual power units. The high integration of the instruments also allows for significant reduction of harness.

Table 1 shows the HIPS for the discussed TRS with the estimated power and mass figures.

S/C Module		Strawman payload	Power (W)	Mass (kg)
VEP	VPO	- Microwave sounder - Visible-NIR imaging spectrometer - UV spectrometer - IR radiometer	55	25
	VEO	- Ground penetrating radar - Radar altimeter - UV / Visible camera	51	18
	Aerobot	- Gas chromatograph /Mass spectrometer with aerosol inlet - Nephelometer - IR radiometer - Meteorological package - Radar altimeter	5.2	4
JME	JEO	- Ground penetrating radar - Stereo camera - Near infrared mapping spectrometer - Radiometer - Magnetometer - Laser altimeter - γ-ray and neutron spectrometer - Radiation environment monitor	25	39
	JRS	- Radiation environment monitor - Plasma wave instrument - Narrow angle camera - Magnetometer - Dust Detector	11	18
DSR	-	- Stereo Imaging Laser Altimeter - Radio Science Experiment - Magnetometer - UV photometer	14	8
IHP	-	- Plasma Analyser - Plasma Wave and Experiment - Magnetometer - Neutral and Charged Atom Detector and Imager - Energetic Particle Detector - Dust analyzer - UV photometer - Visible NIR Imager - FIR Radiometer	15	22

Table 1: Instruments in the HIPS for different TRS

8. CONCLUSION

TRS are introduced to study potential future mission concepts with the main objective to identify and develop technologies that are needed to enable such concepts. Each TRS has identified a set of technologies and provided detailed requirements for the development. Some of the identified technologies are already under development while several others are proposed for future development within ESA technology programmes.

The TRS are helpful to concretize, select and prioritize technologies for ESA technology roadmaps and plans.

9. ACKNOWLEDGEMENTS

The authors want to greatly acknowledge the work performed by the different companies working on the individual TRS in particular, Cosine Research (HIPS), Surrey Satellite Technology (VEP), EADS Astrium (JME), Alcatel Space (DSR), Kayser-Threde (IHP).

10. REFERENCES

1. http://www.sci.esa.int/science-e/www/object/index.cfm?fobjectid=33170

2. Venus Express, Mission Definition Report, *ESA-SCI(2001)6*, 2001. Available at ww.rssd.esa.int/-SB-general/Missions.html.

3. K.-I. Oyama, T. Imamura and T. Abe, Feasibility study for Venus atmosphere mission, *Advances in Space Research*, Vol. 29, 265-271, 2002.

4. M.L. van den Berg, P. Falkner, A.C. Atzei, A. Phipps, J.C. Underwood, J.S. Lingard, J. Moorhouse, S. Kraft, and A. Peacock, Venus Entry Probe Technology Reference Study, submitted to Advances in Space Research, 2004.

5. Microprobe localization and communication prototype system under development by Qinetiq (ESA TRP contract 17946/03/NL/PA).

6. M. Gaffey, J. Bell and D. Cruikshank, Reflectance Spectroscopy and Asteroid Surface Mineralogy, Asteroids II, R. Binzel, T. Gehrels, M. Matthews (editors), The University of Arizona Press, 1989, pp. 98-127

7. D. Renton, P.Falkner, A. Peacock, The Deimos Sample Return, submitted to Advances in Space Research, 2004.

8. A.C. Atzei, P. Falkner, M.L. van den Berg, A. Peacock , The Jupiter Minisat Explorer a Technology Reference mission, (2003) *Proc. 5th IAA International Conference on Low-Cost Planetary Missions*, ESTEC, 24-26 September 2003. ESA SP-542, pp. 189-194, 2003

9. K. Scherer, H. Fichtner and E. Marsch, (eds.) *The outer Heliosphere: Beyond the planets* Copernicus Gesellshchaft e.v., Germany, 2000

10. A. Lyngvi, P. Falkner, A. Peacock, The Interstellar Heliopause Probe, submitted to Advances in Space Research, 2004.

11. S. Kraft, J. Moorhouse, A.L. Mieremet, M. Collon, J. Montella, M. Beijersbergen , J. Harris, M.L. van den Berg, A. Atzei, A. Lyngvi, D. Renton, C. Erd, P. Falkner, On the study of highly integrated payload architectures for future planetary missions. Presented at SPIE 2004.

SMALL BODY SAMPLE RETURN TO DEIMOS

D. C. Renton, P. Falkner, A. Peacock

drenton@rssd.esa.int
Science Payload & Advanced Concepts Office
Planetary Exploration Studies Section (SCI-AP)
European Space Agency, ESTEC, Noordwijk, The Netherlands

ABSTRACT

ESA's Science Payload and Advanced Concepts Office (SCI-A) has recently introduced the Technology Reference Studies (TRS) as a technology development tool to provide a focus for the development of strategically important technologies that are of likely relevance for future scientific missions. This is accomplished through the study of several technologically demanding and scientifically interesting missions, which are not part of the ESA science programme.

The goal of the Deimos Sample Return (DSR) TRS is to study the means of collecting a scientifically significant sample from Deimos' surface and returning it to Earth. The DSR mission profile consists of a small spacecraft, launched on a Soyuz-Fregat 2B. After transferring to the Martian system, the spacecraft will enter into a co-orbit with Deimos where it will perform remote sensing observations and ultimately perform a series of sampling maneuvers. Upon completion of sampling the spacecraft will return to Earth, where the sample canister will perform a direct Earth entry.

This paper will outline the preliminary mission architecture of the DSR TRS, as well as the critical technology drivers. This will include an outline of sampling tools and methods appropriate for a small, low gravity body, as well as planetary protection and re-entry technologies.

1 INTRODUCTION

1.1 Technical Reference Studies (TRS)

The Deimos Sample Return (DSR) is one of ESA's Technology Reference Studies (TRS), which have been introduced to provide a strategic focus for technology development.[1] The TRS have a baseline of a single or a pair of small satellites, with highly miniaturized and highly integrated payload suites. The motivation for this approach is to use low resource spacecraft to create a phased strategic approach to exploration, which will reduce the risk and cost, compared to a single, high resource mission.

Retrieving a sample from a small, low gravity, solar system body is significantly different from retrieving a planetary sample. Whereas a sample return from a larger body, such as a planet, would require a lander, an entry and/or descent system, and a launcher to return to orbit, retrieving a sample from a small body is considerably different. A dedicated launch vehicle after sample collection is not required for such a low gravity environment, and the descent requirements are also significantly altered. The differences and challenges involved in a small body sample return is one of the key reasons that the DSR was chosen as one of ESA's TRS. Deimos specifically was chosen due to the belief that a sample of regolith from the Martian moon contains material from both a class D asteroid and Martian material that was deposited on Deimos during the late heavy bombardment period.

1.2 Deimos

Deimos is one of two moons that are in orbit around Mars. The origins of Deimos and Phobos are unknown, although there are two prevalent theories. One hypothesis states that they are asteroids that were captured into orbit about the planet, while the other theory believes that they were created alongside Mars during the formation of the solar system.

Asteroids are attractive targets for sample return. They are believed to be similar to fossils, retaining some records of the formation of the planets. Deimos is smaller than Phobos, with an acceleration due to gravity less than 0.1 % that of Earth. It is also less irregular in shape than Mars' other moon Phobos and has a smoother appearance due to partial filling of some of its craters. Although Phobos is also of scientific interest, these factors, along with Deimos' larger orbit, make it the more attractive target for such a mission.

Deimos is classified as a D class asteroid. D-type asteroids have low albedos and a generally featureless spectrum. Their spectrums have high values in the infrared region and albedos ranging from 0.04 to 0.07. Deimos' has the highest albedo of any D-asteroid it is 60 % higher than average. This is believed to be cause by the presence of Mars ejecta on the asteroid surface.[2]

The surface mineralogy of a D-class asteroid is inferred to be carbon and perhaps organic-rich silicates. However, current surface mineralogy characterization is not definitive for any of the classes, so D-type asteroids could have differing surface mineralogy. It has been theorized that Deimos' surface is composed of carbonaceous chondrites.[3]

2 SCIENTIFIC RATIONALE

DSR aims to retrieve a scientifically significant sample of material from the Martian satellite and return it to Earth. The recovered sample will provide information about two different solar system bodies, a D-type asteroid, Deimos, and the planet Mars. Modeling demonstrates that approximately 10% of the upper regolith material on Deimos, likely originated from Mars.[2] Deimos has been in orbit around Mars since around the time of its creation and has accreted Martian ejecta. The ejecta were accreted during different eras and came from all over the planet's surface. This material has remained on the asteroid's surface due to Deimos' rubble-pile like structure. This structure efficiently dissipates shock energies and minimizes ejecta velocities, so the majority of ejecta will reaccrete.

The Mars component of the sample will generally consist of Noachian basin forming (4.6-3.8 billion years ago) and late heavy bombardment impacts material (4.0-3.8 billion years ago). This is much older than the material from the SNC meteorites that have been found on Earth, which have been ejected relatively recently. The SNC meteorites are 12 meteorites found on Earth that are believed to have originated on Mars approximately 1.3 billion years ago.

Deimos' regolith has been well mixed and it is expected that material from Mars will be found over the entire surface. This is anticipated due to the fact that the albedo of Deimos is about 60% higher than the average D-type asteroid.

The remaining 90% of the returned sample will consist of material from Deimos itself. D-type asteroids contain primitive material, which were not subject to significant alteration after accretion 4.5 billion year ago. This spectral type of asteroid is common in the outer asteroid belt and for the Trojans, but not in the inner solar system.

3 SCIENTIFIC OBJECTIVES

The main objective of the DSR TRS is to examine the feasibility of returning a meaningful sample from the Deimos surface to Earth. Consequently, no additional scientific measurements are currently planned, beyond those required for sample acquisition. This will reduce mission complexity and the resource requirements of the spacecraft. As a result the science objectives focus on the required size of the sample and its composition.

3.1 Sample Size

The amount of material that will be brought back will influence the science that can be performed. It would be advantageous to have enough material to apply all desired measurement techniques and tests. Most instruments require only a very limited amount of material (<< 1 gram) for investigations and those that require more, need only a few grams. The sampling size required for testing purposes is therefore, only several grams. However, a greater amount is required to get a good overview of the sampled area. Some redundancy is also necessary and there should be some additional material if further research is desired.

The areas of investigation that can be pursued vary with regards to the size of the returned sample. The expected sample size required for each area of investigation of interest has been examined.[4] According to expectations a 1 kg sample will contain about 100 g of Martian dust, which is expected to be the minimum required to perform all the desired research. A sample of this size will allow both complete coverage of Deimos and a clear view of several Martian ejecta originating from different episodes and different places. Therefore, the goal of DSR is to return a 1 kg sample of material.

3.2 Sample Composition

The sample should consist of regolith material from the surface. Optimally this should also include several small pebbles. In addition, the sample should not be composed entirely of 'surface dust' and should have some subsurface material, providing a good mix of regolith.

Images of the satellite's surface indicate that Deimos has a regolith zone, which has an estimated mean depth of 10 m. Optical data indicates that this zone is homogeneous across the surface. As evidenced by its many craters, Deimos has been subjected to heavy bombardment by meteorites in its past. The majority of ejecta from this bombardment would reaccrete due to Deimos' rubble-pile like structure that efficiently dissipates shock energies and minimizes ejecta velocities. The surface material thus became widely dispersed. Therefore, the composition of the surface sample does not depend on sampling location. The samples would be similar from any part of the surface.

However, there is one science restriction for the sampling site selection. Newly formed craters by unknown objects should be avoided. Samples from these locations would be likely to contain a high concentration of material from the impacting body and therefore less of the desired material from Deimos and Mars.

4 DSR ARCHITECTURE

The preliminary architecture consists of a small or mini spacecraft, launched into the Mars-Deimos System on a Soyuz Fregate 2B (or equivalent "low cost" launcher). The spacecraft will be launched into a 200 x 25 000 km Earth orbit, after which it will begin its transfer to the Martian system. Upon reaching Mars, the spacecraft will be placed into a highly elliptical orbit (500 x 100 000 km) during orbit insertion, before performing a series of maneuvers to enter into a co-orbiting trajectory with Deimos (20 069 km circular orbit). The spacecraft will then enter into an observation mode, performing measurements of Deimos surface and gravitational properties before obtaining the samples. Once the samples are obtained they will be transferred into a canister inside the Earth Entry Vehicle (EEV). Unnecessary components of the spacecraft, such as the sampling mechanism and empty tanks, will then be separated and left in Martian orbit to reduce propellant requirements for the transfer back to Earth. Upon approaching Earth the EEV will separate and perform a direct re-entry.

Launch Date	Earth-Mars Transfer	Mars Departure Date	Mars-Earth Transfer	Stay Time (days)	Mission Duration (yrs)	Total DeltaV (km/s)
10-Nov-11	0.5 rev.	08-Aug-13	0.5 rev.	331	2.71	2.67
7-Dec-13	0.5 rev.	15-Mar-15	1.5 rev.	169	3.45	3.04
17-Jan-16	0.5 rev.	19-Mar-18	0.5 rev.	515	2.75	3.29
25-Oct-17	1.5 rev.	16-Jun-20	0.5 rev.	122	3.16	3.03
12-May-18	0.5 rev.	21-Feb-19	1.5 rev.	79	3.05	2.73
10-Nov-19	1.5 rev.	10-Jul-22	0.5 rev.	154	3.42	3.08

Table 4-1: Optimal High Thrust Transfers

The instrumentation on-board the spacecraft will be composed of a Highly Integrated Payload Suite (HIPS) in order to reduce resource and size requirements.[5] In order to safely obtain a sample from the surface, if any kind of close approach or touch down is to be made, the gravitational and surface characteristics of Deimos must be known. Therefore the payload will likely contain a set of remote sensing instruments in order to characterize the asteroid and sampling sites to aid in determining the navigation and control sequences for the sampling maneuvers. In addition to those required for characterization activities before sampling, imaging and range finding instruments will be required during the maneuvers.[6]

4.1 Mission Analysis

The mission analysis for a sample return to Deimos has been examined for launch in the 2010-2020 time frame.[7] Both low and high thrust scenarios were analyzed along with gravity assists, optimal stay times and Martian orbits, as well as other ΔV reducing measures. The study has determined that it is feasible to return a significant mass using both chemical and combined chemical and Solar Electric Propulsion (SEP) scenarios. However, due to the higher cost of a SEP system, the baseline will be a Chemical Propulsion (CP) system.

4.1.1 High Thrust Transfers

The baseline scenario for high thrust transfers uses half revolution transfers to and from Mars. A direct entry is envisioned at Earth return since an Earth orbit insertion would not be feasible with the mass constraints of using a Soyuz Fregate 2B launch vehicle. The main concern of this scenario is the long required stay times at Mars of about 450 days. These can be decreased with a 1.5 revolution transfer scenario, however the transfer time is increased and the overall mission time remains relatively unchanged.

For the nominal high thrust scenarios the optimum total ΔVs are around 7 km/s and the optimum stay time ranges between 330 and 550 days. The optimum transfers are outlined in Table 4-1. The type of transfer is also noted for each segment, whether it is a 0.5 (short) or 1.5 (long) revolution transfer.

4.1.2 Transfer Mass Analysis

The useful masses available at Earth return were also analyzed for the optimum transfers and can be found with the mass breakdown in Table 4-2. The analysis assumes the use of a CP system with a specific impulse of 320s and that the maximum capacity of the Soyuz Fregat 2B, 2890.8 kg, is employed to launch into a HEO (200 x 25 000 km). The transfers have two CP stages in order to maximize the Earth returned mass as it was found that the CP-CP staged transfer has a mass advantage over a single CP transfer.

The masses at atmospheric entry in Table 4-2 represent the maximum spacecraft dry mass remaining upon reaching the Earth's atmosphere. For these transfer scenarios, the masses range between 200 and 300 kg. Leaving mass behind at Mars or increasing the performance of the CP system (i.e. increasing Isp) could increase this mass. However, preliminary analysis indicates that the mass at atmospheric entry for these transfer cases should be adequate for the needs of DSR.

Launch Epoch	**2011**	**2013**	**2016**	**2017**	**2018**	**2020**
Chemical Stage 1 Dry Mass (kg)	746	757.7	805	761	724.8	746.2
Chemical Stage 1 Fuel Mass (kg)	162.5	164	170	164.4	159.7	162.5
Chemical Stage 2 Dry Mass (kg)	1065.3	1114.3	1101.5	1110.4	1160.3	1132.6
Chemical Stage 2 Fuel Mass (kg)	198.3	203.1	201.9	202.7	207.6	204.9
Transfer Rate (kg/year)	121.2	75.9	80.9	82.9	79.3	74.4
Mass at Atmospheric Entry (kg)	**328.7**	**261.7**	**222.4**	**262.2**	**248.3**	**254.5**

Table 4-2: Mass at Atmospheric Entry for High Thrust Transfers

4.2 Deimos Observational Orbit

Once the spacecraft has entered into a co-orbiting trajectory with Deimos it will be placed into an observational orbit in order to observe the surface before performing sampling maneuvers. This observational orbit will be achieved by slightly modifying the eccentricity and inclination of the spacecraft's orbit, with respect to that of Deimos (see Table 4-3).

Orbital Characteristics	Deimos	Spacecraft
Eccentricity	0.0005	+ 0.0005
Inclination (deg.)	1.79	+/- 0.05

Table 4-3: Orbital Characteristics for Observation Orbit

The difference in eccentricity will produce a relative circular motion about Deimos with a distance of ~11.5 km from the surface and a 30 hour period. The slight difference in inclination will allow observation of the North and South Poles. This relative circular motion can be seen in Figures 4-1 and 4-2.

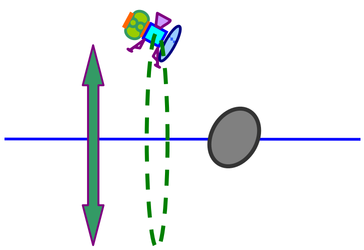


Figure 4-1: Spacecraft Relative Motion

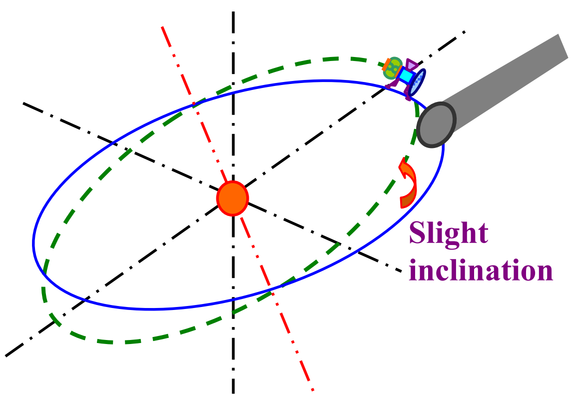


Figure 4-2: Spacecraft's Modified Orbit

This observational orbit will permit the examination of a large number of sampling sites on the surface, enabling the selection of the most optimal locations for sampling maneuvers. The repetitive motion about Deimos will also aid in the accurate mapping of its gravitation field, which will be key in the determination of the navigation sequence for sampling maneuvers.

5 PLANETARY PROTECTION

"The Outer Space Treaty of 1967 specifically requires that all space exploration must be done in a way that avoids harmful contamination to celestial bodies or adverse changes in the environment of the Earth from the introduction of extraterrestrial materials."* The impact of both back and forward contamination during this mission must therefore be addressed.[8]

It has been determined that there is little danger from contaminating Deimos with Earth materials, however it would compromise the scientific integrity of the returned sample. Therefore, forward contamination of the samples and sample sites must be prevented.

It has also been determined that the prevention of back contamination is not strictly required for bodies such as Deimos. Specifically, a report by the US Space Studies Board states that containment is not warranted for samples returned from the Martian moons, Phobos and Deimos.[9] However, the current COSPAR Planetary Protection Policy recommends further study before any such mission is undertaken.

If it were concluded that the prevention of back contamination is warranted, several stringent requirements would be necessary for DSR. The exterior of the re-entry vehicle would need to remain uncontaminated by the sample or any other Deimos material during sample collection. The containment of the sample would also have to be verifiable before re-entry and the sample capsule would need to be sufficiently robust in order to withstand a crash landing. All of this must be done in accordance with current Planetary Protection guidelines.

---

* http://www.astrobiology.com/adastra/bring.em.back.html

The current strategy for the DSR TRS is to adopt the more stringent requirements, protecting against back contamination. This will ensure the feasibility of the design in the case that further studies of Deimos determine that a returned sample could be hazardous. This approach also has the benefit that such a DSR mission could potentially act as a technology demonstration mission for several Mars Sample Return technologies.

6 ENABLING TECHNOLOGIES

Several technologies have been identified as enabling for a DSR mission. These technologies fall into two categories, those required to collect a sample from Deimos' surface and those required to insure protection against back contamination. The enabling technologies for sample collection include a highly autonomous guidance navigation and control system, as well as the sampling mechanism itself. The technologies required for planetary protection compliance include a sample containment mechanism, to break the contamination chain, and a robust Earth entry vehicle.

6.1 Sampling Mechanism

There are various methods that can be used to retrieve a sample from a small solar system body such as Deimos. The two main options involve whether or not the spacecraft will make contact with the surface. The sample could be obtained through: collecting the sample directly from the surface or creating a debris cloud of asteroid material and collecting a sample from that cloud. Both of these options will necessitate the development of new technology in order to optimize the sampling method for a small body and to accommodate its use on a small or mini spacecraft. However, collecting samples from a debris cloud is extremely limiting in possible sample size and it would be difficult to collect the required 1 kg sample.

One option for a rendezvous sample collection would be a touch-and-go. The spacecraft would briefly make contact with the surface; collect the sample and return to orbit. Anchoring of the spacecraft would not be required which would simplify operations and reduce spacecraft mass. In addition, for a touch-and-go methods the relative speed between the spacecraft and the asteroid would not necessarily need to be as low as for a precision landing, thus decreasing the required ΔV. It also might be possible to conserve the momentum of the spacecraft while reversing the direction of travel. This would further decrease the ΔV needed to return to orbit.

A touch-and-go sampling maneuver could prove optimal in terms of spacecraft and mission requirements, however it introduces several challenges in collecting the actual sample. The sample would have to be collected in a very short time frame and this could prove exceedingly difficult considering the amount of sample required. It is unlikely that a 1 kg sample could be collected during a single maneuver so several maneuvers would have to be performed. However, this produces the added benefit of providing multiple sampling sites.

Several mechanisms could be used in such a touch-and-go maneuver, where the spacecraft briefly makes contact with the surface before returning to orbit. A scoop could be used to collect the sample, scooping a quantity of regolith into a collector when the spacecraft touches the surface. A compressed collector device could also be used. It would imbed itself into the surface as the spacecraft impacts and then, as the spacecraft returns to orbit, the collector would be withdrawn from the surface with the desired sample contained within. Hyabusa, the Japanese asteroid sample return mission uses a touch-and-go sampling approach. Upon a brief contact with the surface, a projectile is fired and the debris is then funneled up a cone as the spacecraft retreats from the surface.[10] A similar device could also be useful for DSR, however the means of collecting a sample of sufficient size would need to be addressed.

The design and development of a sampling mechanism capable of collecting a 1 kg sample of regolith from a small body is critical for the feasibility of a DSR mission. The mechanism should also be compatible with the optimal touch-and-go type-sampling maneuver.

6.2 Highly Autonomous Guidance, Navigation and Control System

Performing a rendezvous or landing maneuver on the surface of a small body, with only a small gravitational field, presents several challenges. The requirements for the approach and rendezvous or landing will largely depend on the sampling method selected. However, the survival of the spacecraft after the sampling maneuver is critical, so the approach towards and any contact with the surface must be strictly controlled.

Due to the lag time in communication between the Earth and the Martian system, real time control during these critical maneuvers will not be possible. Therefore a highly autonomous guidance, navigation and control system must be developed to ensure feasibility of such a mission.

6.3 Earth Entry Vehicle (EEV)

In order to comply with back contamination protection requirements the Earth Entry Vehicle (EEV) must ensure containment of the sample, in addition to bringing it safely to Earth. The containment seal on the EEV door must be verifiable before Earth entry will be permitted and containment of the sample must be ensured during entry and landing. The EEV will also need to enable rapid localization and recovery of the sample. Several current and past studies have examined potential designs of re-entry vehicles for Mars Sample Return (MSR) missions. These studies could prove beneficial for the design and development of the EEV for a DSR mission. There also exists the potential for a DSR mission to provide technology demonstration for a MSR Earth re-entry vehicle.

6.4 Sample Handling/Containment

In order to comply with the planetary protection requirements to prevent back contamination, the sample must be contained and the exterior of the EEV cannot come into contact with any foreign material. Therefore a break in the contamination chain is required where the EEV is separated from any sections of the spacecraft that have made contact with Deimos material. Figure 6-1 shows a method for this contamination break during transfer of the sample to the EEV. The red outer casing provides a shield that prevents contamination of the exterior surface of the EEV. It will prevent contamination from any contact with the sample material as well as contact with any debris or dust cloud created when the spacecraft impacts the surface.

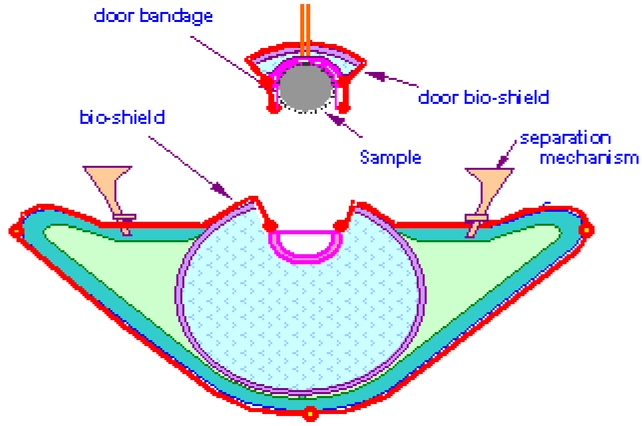


Figure 6-1: EEV Contamination Protection and Sample Containment*

As in the case of the EEV, sample handling and containment mechanisms have been studied for MSR missions. These studies could prove beneficial to DSR and such a mission could also provide technology demonstration for these technologies.

7 CONCLUSION

The Deimos Sample Return Technology Reference Study aims to focus the development of technology required for returning a sample from a small solar system body. In the preliminary stages of the study several enabling technologies have been identified. The sampling methodology and mechanisms required for DSR have the potential to be used in collecting a sample from any small solar system body. The sample containment mechanisms and entry vehicle could be used for any sample return mission requiring back contamination protection. The continuing study of the DSR TRS will continue to define a mission profile in order to examine feasibility and to refine technological requirements for such a mission.

* Figure courtesy of Alcatel Space (Cannes)

ACKNOWLEDGEMENTS

The technical assistance of Alcatel Space in the study of the Deimos Sample Return Technology Reference Study, under ESA contract 18050/04/NL/HB and EADS Astrium in the study of the DSR mission analysis, under ESA contract 17241/03/NL/HB, is gratefully acknowledged.

REFERENCES

1. Lyngvi, A., P. Falkner, A. Atzei, D. Renton, M. v.d. Berg and A. Peacock, *Technology Reference Studies,* proceedings of International Astronautical Congress, Vancouver 2004

2. Gaffey, M., J. Bell and D. Cruikshank, *Reflectance Spectroscopy and Asteroid Surface Mineralogy*, **Asteroids II**, R. Binzel, T. Gehrels, M. Matthews (editors), The University of Arizona Press, 1989, pg 98-127

3. Tholen, D. and M. A. Barucci, *Asteroid Taxonomy*, **Asteroids II**, R. Binzel, T. Gehrels, M. Matthews (editors), The University of Arizona Press, 1989, pg 298-315

4. Molster, F., *Scientific Reasoning for A Deimos Sample Return Mission*, Internal Report, Science Payloads and Advanced Concepts Office, Directorate of Science, ESA, 2003

5. S. Kraft, J. Moorhouse, A.L. Mieremet, M. Collon, J. Montella, M. Beijersbergen , J. Harris, M.L. van den Berg, A. Atzei, A. Lyngvi, D. Renton, C. Erd, P. Falkner, *On the Study of Highly Integrated Payload Architectures for Future Planetary Missions,* proceedings of SPIE Europe International Symposium - Remote Sensing, Spain, Sept. 2004

6. Schulz R., *Ground-Based Support for Landing and/or Sample Return Missions to Small Bodies*, Advances in Space Research, Vol. 25, Issue 2, 2000, pg 257-268

7. Kemble, S., M. Taylor, C. Warren and S. Eckersly, *TRM to the Inner Planets: Payload in Orbit Optimization*, Technical Note 3, ESA Contract 17241/03/NL/HB, EADS Astrium Limited. (Stevenage), 2003

8. Race, M. and J. Rummel, *Bring Em Back Alive—Or At Least Carefully*, Ad Astra Astrobiology Issue Expanded Edition, http://wwww.astrobiology.com/adastra/bring.em.back.html

9. Task Group on Sample Return from Small Solar System Bodies, Space Studies Board (SSB), National Research Council, Washington D.C., *Sample Return from Small Solar System Bodies*, Advances in Space Research, Vol. 25, Issue 2, 2000, pg 239-248

10. Fujiwara, A., T. Mukai, J. Kawaguchi, and K.T. Uesugi, *Sample Return Mission to NEA: MUSES-C*, Advances in Space Research*,* Vol. 25, No. 2, 2000

IAC-04-Q.2.a.02

SYSTEM CONCEPTS AND ENABLING TECHNOLOGIES FOR AN ESA LOW-COST MISSION TO JUPITER/EUROPA

Patrice Renard, Charles Koeck, Steve Kemble
EADS-Astrium, France & United-Kingdom
patrice.renard@astrium.eads.net, charles.koeck@astrium.eads.net, steve.kemble@astrium.eads.net

Alessandro Atzei, Peter Falkner
ESA-ESTEC, The Netherlands
aatzei@rssd.esa.int, Peter.Falkner@rssd.esa.int

ABSTRACT

The European Space Agency is currently studying the Jovian Minisat Explorer (JME), as part of its Technology Reference Studies (TRS). TRS are model science-driven studies contributing in the ESA strategic development plan of technologies that will enable future scientific missions.
The JME focuses on the exploration of the Jovian system and particularly the exploration of its moon Europa. The Jupiter Minisat Orbiter (JMO) study, which is the subject of the present paper, concerns the first mission phase of JME that counts up to three missions spaced in time by 6 years using pairs of minisats. The scientific objectives are the investigation of Europa's global topography, the composition of its (sub)surface and the demonstration of existence of a subsurface ocean below Europa's icy crust.
The present paper describes the candidate JMO system concept, based on a Europa Orbiter (JEO) supported by a communications relay satellite (JRS), and its associated technology development plan. It summarizes an analysis performed in 2004 jointly by ESA and the EADS-Astrium Company in the frame of an industrial technical assistance to ESA.
It addresses the interplanetary transfer, the hostile radiation environment, the power generation issue, the communication system, as well as the need for high autonomy on-board.

1 INTRODUCTION

ESA's Science Payload & Advanced Concepts Office has started a combination of activities that go by the name "Technology Reference Studies". The goal of the TRS's is to identify and develop critical technologies that will be required for future scientific missions. This is done through the study of several challenging and scientifically relevant missions, which are not part of the ESA science programme, and focus on the medium term enabling technology requirements.
The TRS's share the same baseline: the use of one or more small spacecraft using a suite of highly miniaturised and integrated payloads, with strongly reduced resource requirements. The purpose of this approach is to achieve the science objectives with a phased, cost efficient exploration, resulting in a reduced overall mission risk, when compared to a large "one-shot" mission.

This paper addresses one of them called JMO [1], standing for Jupiter Minisat Orbiter. This mission has to be seen as the first phase of a larger concept aiming at the Jovian system exploration, JME. JME is composed of up to three phases spaced in time by 6 years, using pairs of minisats. The science objective of these minisats is targeted towards Europa with regard to astro-biology and the presence of surface ice on this Jupiter moon, although other moons could be of interest. One of these two satellites, the Jovian Europa Orbiter (JEO) will perform in-orbit remote sensing measurements for an expected duration of 60 days, whereas the second one, the Jovian Relay Satellite (JRS) will be put in an orbit around Jupiter outside its main radiation belts, permitting to relay the JEO science data to Earth. Jupiter science observations from JRS are also seen as an added value to the mission.

The work presented here was performed within the frame of a technical assistance to ESA in support of an assessment study of the JMO.

Our presentation first addresses the **interplanetary transfer** analysis carried out in combination with propulsion systems and spaceship staging trade-offs. Chemical propulsion is considered either alone or associated with solar electrical propulsion. The constraining mission requirements are a launch by Soyuz-Fregat and a trip to Jupiter not longer than 6 years, fixed for cost efficiency purposes.

The Jovian hostile **radiation** environment is the second critical point ; in the Europa vicinity, surviving a few Mrads dose after 2 months of science experiments is indeed a real challenge. Our document shows the method for assessing the radiation levels, and the mitigation strategy.

At a distance of 5AU from Sun, and anticipating solar cell degradation as high as 38%, the use of solar electrical **power generation** rather than radio-isotope might seem unrealistic. We however demonstrate that this option is reachable within reasonable technological steps and with acceptable panel sizes.

Next, we discuss the need for a **communications** relay satellite, the choice between Ka-band and optical links, and the communications architecture designed both for commanding the two satellites and for science data download towards Earth, collected at 40kbps around Europa.

System **autonomy** is then presented. Autonomy is mandatory when a quick reaction is required whereas any signal takes more than 50 minutes to travel between Earth and spacecraft. It is also a mean to reduce the ground station work load.

Planetary protection is also an issue of prime importance as the objective is to keep Europa uncontaminated in order not to compromise further science investigations.

The resulting **spacecraft conceptual design** is described, with some details on the avionics and the propulsion system. The presentation ends with a synthesis of **requirements for new technologies,** and a tentative schedule for JMO developments.

2 INTERPLANETARY TRANSFER

2.1 Requirements and drivers

The objective is to define an interplanetary scenario that will place the two JMO minisatellites in their operational Jovian orbits using a Soyuz-Fregat launch vehicle in its updated variant, 2B, from Kourou, with an assumed launch capacity of 3000kg in geostationary transfer orbit [3]. The Orbiter (JEO) is placed in a low orbit around Europa, whereas the relay satellite (JRS) remains in an orbit around Jupiter. Such scenario has to be optimised according to the following criteria:

- interplanetary transfer duration; <6 years up to Jupiter arrival fixed as a requirement;
- radiation doses during transfer around Jupiter;
- propellant mass minimisation;
- intersatellite communications constraints.

To perform such optimisation, the following parameters were considered:

- interplanetary route with use of gravity-assist manoeuvres;
- all chemical propulsion system versus hybrid solar electrical-chemical propulsion system;
- number of modules: a dedicated propulsion stage is therefore considered in addition to the two minisatellites;
- operational orbits orbital parameters.

2.2 All chemical propulsion

Direct high thrust transfers up to Jupiter require too high ΔV for the available launch capacity and therefore are discarded. Multi gravity assists (GA) options, around Earth and Venus to provide aphelion raising, permit to save propellant mass.

The most ΔV efficient case is the VGA-EGA-EGA route, shown in Figure 1. That was the route followed by Galileo (launched in Oct. 1989) [2].

Good launch opportunities, summarized in Table 1, occur between 2010 and 2030, and are driven by the Earth-Venus synodic period,. Transfer durations in the order of 6 years imposes ΔVs' in the range 1900m/s to 2400m/s.

Launch Date	Total DV	Duration	Launch Date	Total DV	Duration
19-Jul-10	2290 m/s	6.4 yrs	8-Feb-20	2770 m/s	6.2 yrs
31-Jul-11	2380 m/s	6.2 yrs	16-May-23	2140 m/s	6.2 yrs
21-Apr-12	1890 m/s	6.4 yrs	26-Oct-24	2560 m/s	7.2 yrs
7-Oct-13	2300 m/s	6.2 yrs	13- Aug-26	2210 m/s	6.1 yrs
1-Jan-17	2180 m/s	6.0 yrs	20-Nov-29	2580 m/s	7.4 yrs
25-Jun-18	2240 m/s	9.1 yrs			

Table 1: ΔVs and durations as function of launch dates for interplanetary transfer using all chemical propulsion

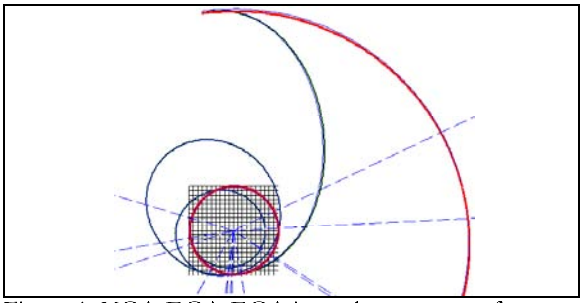


Figure 1: VGA-EGA-EGA interplanetary transfer

2.3 Hybrid electrical/chemical propulsion

Solar Electrical Propulsion (SEP) is considered in addition to chemical propulsion. Due to the power demand of such propulsion mode, the strategy is to use it only for Earth departure, where adequate power can be collected with solar arrays of acceptable sizes. SEP is combined with a Lunar GA (LGA) to provide a low energy Earth escape, and one or several EGAs' to provide aphelion raising.

After LGA, return to Earth occurs typically 15 months later after an intermediate deep space ΔV to increase Earth approach speed. After a second gravity assist at Earth, aphelion is raised considerably. With one EGA the transfer duration is 3.7 years but requires 6000m/s with a 200mN thrust per ton, whereas it is 5.7 years with 2 EGAs', and only 2200m/s with same thrust to mass ratio.

A study of ΔV sensitivity to acceleration capacity (thrust to mass), presented in Figure 2, allows trade-off of ΔV with propulsion system mass. After aphelion raising by chemical propulsion the composite spacecraft mass is <2000kg. Moreover, Europe is developing SEP Xenon thrusters providing 150mN to 200mN thrusts with Isp>4000s, for telecommunications satellites and for the Bepi Colombo mission to Mercury. Comparing the propellant mass gain of having more thrusters with the mass penalty due to SEP hardware equipment showed that one thruster is optimal. During this analysis, it has also been demonstrated that the solar array as sized for power requirements around Jupiter is sufficient for the SEP needs around Earth.

Launch opportunities occur each 13 months with ΔV varying between 2800m/s and 3300m/s, for a Xenon mass variation between 170kg and 190kg. This proves that SEP permits to have a system concept more flexible with respect to the launch date than the all chemical propulsion option.

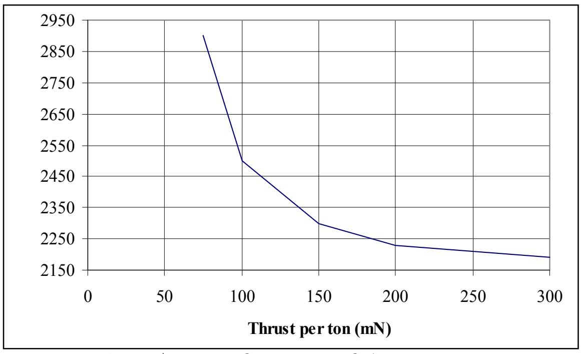


Figure 2: SEP ΔVs' as function of thrust to mass ratio in an LGA-EGA-EGA scenario

2.4 Capture at Jupiter

The capture at Jupiter is performed by chemical propulsion. An impulsive ΔV can perform a direct injection, but a Callisto or Ganymede GA followed by a pericentre ΔV is more efficient. This ΔV depends on the arrival velocity, as shown in Figure 3. GA with Io was the option for Galileo.

The resulting capture orbit is a 900000km by 20million km orbit. After capture, it has been found more mass efficient to put the two satellites on different trajectories.

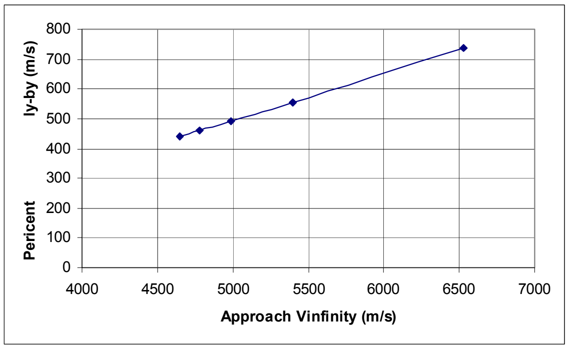


Figure 3: injection ΔV as function of approach velocity after a Ganymede GA

2.5 JEO Jupiter tour and insertion

The main issues for JEO are transfer time, ΔV and radiation. The pericentre burn required for Europa orbit insertion depends on the approach excess hyperbolic speed to the moon. Therefore, insertion ΔV is reduced if GAs' and intermediate ΔVs' can be used to achieve a low eccentricity Jupiter orbit with a similar semi-major axis as Europa. The trade-off resulted in a sequence of 4 Ganymede GAs', followed by 7 Europa GAs', for a total duration of 550days. Intermediate ΔVs' amount to a total of 350m/s.

Finally, JEO is inserted on a polar circular orbit around Europa, at 200km altitude, by means of a

ΔV equal to 920m/s. Such altitude was selected with respect to the following constraints:

- power required on the science instruments for a given observation accuracy rapidly increases with altitude;
- altitude also influences the eclipse duration, which should be limited to reduce battery charging needs;
- the higher the altitude, the quicker the orbit decay (this trend starts from ~150 km upwards; below it is not the case).

Orbit altitude control is moreover baselined as it only requires ~20m/s for 60 days. Without any orbit control, typically when its mission ends, JEO impacts the Europa surface after 60days.

2.6 JRS Jupiter tour and insertion

Similarly to JEO, the issues for JRS are transfer time, ΔV and radiation. But communications with JEO play an important role in the final orbit selection. An operational orbit resonant with Europa permits to envisage communications slots at regular time intervals and shortest distance. A 3:1 resonant orbit instead of a 2:1 resonant one was selected, as it implies higher distances to Jupiter, which means lower radiation. The orbit is equatorial with apojove at 26.3Rj (Rj=Jupiter radius=71,400km) and perijove at 12.7Rj. The orbital period is 10.6days.

Such orbit is reached by means of 4 Ganymede GAs', 1 Callisto GA and a final Ganymede GA. The required ΔV is 280m/s, and the tour duration 450days.

2.7 Staging analysis

The staging analysis is first intended to optimize the total mass by defining the number of modules composing the spacecraft, and the propulsion system assigned to each of them. Complexity and cost are other criteria taken into consideration.

Staging optimisation included the following options:

- addition of boosters for Earth aphelion raising;
- high thrust chemical or SEP for Earth escape;
- additional Carrier module to perform Jupiter insertion, and possibly to bring JRS or JEO to its operational orbit.

The trade-off conclusions are the following:

- There exist solutions compatible with launch mass capacity for both all chemical propulsion and chemical+electrical propulsions;
- the mass optimum architecture with solar electrical propulsion is 140kg lower in mass than the mass optimum architecture with all chemical propulsion;
- options with Carrier module are heavier;
- boosters are only worth using with electrical propulsion because of the Isp improvement in the apogee raising phase (320s for boosters instead of 290s for small thrusters).

3 SYSTEM CONCEPTS

3.1 System concept drivers

Table 2 gives an overview of the JMO system drivers, and the decision rationale for each of them. The main criteria for decision were the cost, the reliability and the science return. Although new technologies are mandatory to enable such mission, particular care was taken to limit their number, in order to increase the chances for realizing it within two decades from now.

	Elements to be traded	Decision rationale
Mission with one or two satellites	Need for JRS to relay science data between JEO and Earth	Two satellites required. Launch mass does not permit to embark enough power on JEO to directly transmit the required amount of data in 60 days
Inter-planetary transfer	Chemical propulsion or hybrid chemical-electrical propulsion	Chemical propulsion selected for robustness and cost reasons
Staging	Number of stages and type of propulsion system	2 stages selected for simplicity (=> robustness & cost) & launch mass Dual mode on JEO using 4x22N thrusters for deltaVs , Isp=308s
Radiation	JEO equipment tolerance	H/W tolerant to 1Mrad together with additional satellite shielding (10mm on JEO), based on a technological feasibility estimation
	Solar cell tolerance to radiations	Off-the-shelf GaAs cells have acceptable degradation levels
Power systems	Solar arrays or RTGs	Solar arrays selected because of ecological problems due to Earth fly-bys and lack of RTG availability. JEO Europa orbit local solar time=60°
Commu-nication archi-tecture	Permanent links between JEO and JRS versus dedicated slots	JRS orbit selection permitting communications at shortest distance
	Mobile or fixed antenna on both JRS and JEO	Fixed antenna selected for robustness and cost purposes
	JEO direct communi-cation with Earth for ranging and TM/TC (excluding science TM)	JEO TM/TC communication with Earth, identical to JRS
Wavelength for commu-nications	Ka-band, X-band or optical links Data rates	Ka-band at 30kbps to Earth and 2300kbps at 250000km from JEO to JRS. Optical links not bringing advantages accounting for mid-term perspectives

Table 2: JMO system concept trade-offs overview

3.2 Radiation

Radiation, caused by Jupiter electrons emission, is the most critical issue of the JMO mission.

To assess the Jovian radiation levels around Jupiter, the Divine-Garrett model was considered as the reference model for the JMO study. This model seems to be more pessimistic, and thus leads to more conservative solutions, than a new model like the Galileo Interim Radiation Electron (GIRE) that could replace it in the near future.

Results, as shown in Figures 4 & 5, demonstrate that off-the-shelf hardened devices, withstanding typically 200krad as a maximum, are not well suited for a JMO mission. The 1st enabling technology for JMO is therefore to consider that new equipments can be built which are tolerant to 1Mrad. A substantial development effort will be required for such objective.

Based on that assumption, a typical 10mm aluminium shielding is required on JEO, and 4mm on JRS. This is presented in Table 3 together with fluences computation, assuming 500µm cover glass over the solar cells, and 50% margin.

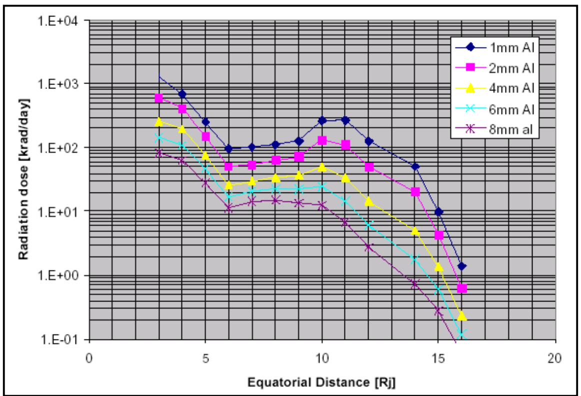


Figure 4: Daily ionizing dose as a function of distance from Jupiter for varying shielding thicknesses

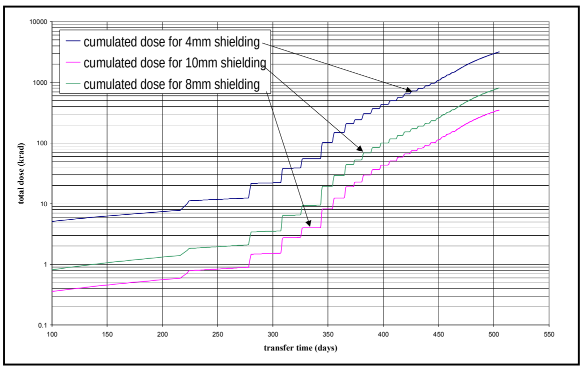


Figure 5: JEO total ionizing dose during 1.5-year Jupiter tour for varying shielding thicknesses

	Radiation dose (krad)		Fluence on cells (1MeV e-/cm²)	
	JEO	JRS	JEO	JRS
Jupiter tour	350	74	9.0e14	2.2 e13
orbit	420	450 per year	1.15e15	1.4 e14 per year

Table 3: JEO and JRS total doses and fluences

3.3 Power

An early decision not to go for radioisotopes, motivated by launch safety constraints and lack of existing hardware in Europe, has permitted to study a concept with solar power generation.

Solar cells

Due to low solar flux input (50W/m²), triple-junction GaAs cells are preferred to Si cells for their higher efficiency. Such cells however will operate in low intensity and low temperature (LILT) conditions. In Europe, the LILT technology only exists for Si cells [4], and was applied for the Rosetta probe, launched in 03/02/2004 from Kourou. The 2nd enabling technology is thus the development of GaAs LILT cells.

In addition, and according to manufacturers data, the efficiency decrease of off-the-shelf GaAs cells covered with 500µm coverglass, and for a fluence of 3e15 1MeV-equivalent electrons/cm², is 38%. That value, combined with an initial electrical power conversion efficiency of 34% at -100°C, was found to be acceptable in the system design. However, early verification of cell behaviour with respect to high fluences will be required.

Solar concentrators

Due to the low Sun flux intensity and the mass constraints, techniques enabling to collect more flux on the cells were investigated. The most efficient was found to be a concept similar to the one used for the concentrators implemented on the Boeing HS-702 telecommunications satellite. The principle of these concentrators, considered as the 3rd enabling technology, is depicted in Figure 6. Light incident on inclined flat panels mounted on both sides of the panel covered with cells is specularly reflected on that one.

The concentrators surfacic mass is assumed to be 150g/m², where solar panels featuring cells and coverglass weight 4kg/m². Without concentrator, the panel mass efficiency is 2.2W/kg, for JEO in end-of-life (EOL) conditions. The theoretical maximum with concentrators leads to 4.7W/kg. The baseline is actually to consider concentrators tilted by 60°, thus with same width as the solar panel, and a specularity ratio of 0.8 at EOL accounting for any degradation. This gives:

- JEO EOL: 3.9W/kg & 15.8 W/m^2;
- JRS EOL: 4.9W/kg & 21.2 W/m^2.

Early testings of the specularity ratio will be however required to confirm the assumption, as the impact on the system may be very high.

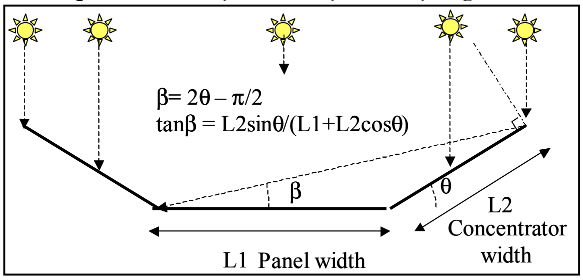


Figure 6: JMO solar flux concentrator principle

3.4 Communications

Communication architecture drivers

The trade-off objective is to determine the preferred JMO communication architecture in terms of wave-lengths (RF & optical) and associated equipments, possible communication links between JEO, JRS and Earth, and types of antenna. The trade-off drivers were the following:

- Data: data rate & volume;
- JEO-JRS relative geometry: distances, pointing, occultations;
- Antenna size limitation: Soyuz-Fregat, pointing accuracy;
- Ground station: available frequencies, G/T (for 34m ESA DSN), EIRP for TC;
- RF emission/reception auto-compatibility;
- ranging: position accuracy vs correction ΔV;
- Robustness (to avoid sat. to be 'lost in space');
- Cost: technologies, common equipments on JEO & JRS, ground operations;
- Mass: limit numbers of equipments.

Wavelength selection

Optical links present a big potential for future missions [5]. In Europe, it has been successfully tested between the Artemis geostationnary satellite and the Spot4 low Earth orbiting satellite. Presently however, such technology is not mature enough, doesn't show decisive advantages in terms of mass power and volume for the mid-term, and would add too much complexity.

RF is therefore selected, and Ka-band preferred to X-band, as it requires less power for same data rates.

Communication links

The JMO system communication links are sketched in Figure 7. Links can be permanent or temporary. A permanent communication link of JEO with JRS during science observations would impose a high gain antenna (HGA) mounted on 2 axes. This means a complex mechanism, although this exists on Rosetta. Moreover, the data rate capacity is very low in regions where the distance between the two satellites is maximum. On the other side, a permanent link on JRS would impose either to upload all science data before sending them to Earth, or to have 2 HGAs'.

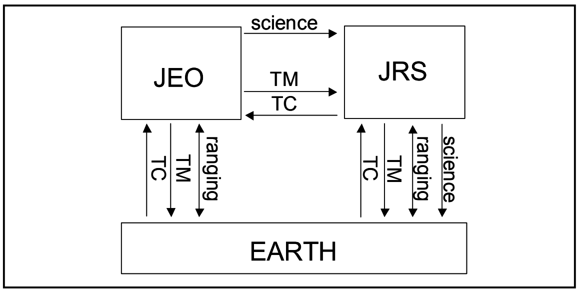


Figure 7: Communications links configuration

Having temporary slots and fixed antenna on both satellites is thus the selected option. The best strategy for this is to have a JRS orbit synchronised with Europa, and to perform communications when JRS is at its perijove, as depicted in Figure 8. JEO & JRS are baselined with identical 1.5m parabolic HGA. JEO features a 3.5W solid-state power amplifier, enabling a data rate of 2.3Mbps at 250000km. Science data collected by JEO during 10.6 days at 40kbps (required value) are thus transmitted to JRS in less than 6h. Such strategy requires the 4th enabling technology:

- high data rate Ka-band receiver on JRS;
- 30% efficiency SSPA on JEO, to optimize power resources, where the current state-of-the-art is 15% [6].

On JRS, the 1.5m HGA permits to consider emitting data towards Earth at 30kbps with 45W RF power. Assuming 8h communication windows a day with Earth, 300 days are enough to transmit the whole science data estimated at 250Gbits.

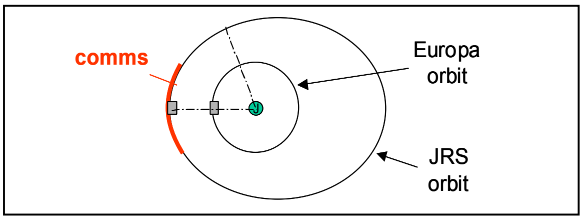


Figure 8: JEO-JRS communication slot

3.5 System Autonomy

System autonomy is mandatory due to the very long mission duration, and because it's not possible to react interactively from ground at such distances from Earth.

During interplanetary cruise, a daily beacon monitor track is performed to establish that no on-board event has been detected that requires ground interaction until the next regularly (interval in the order of two weeks) scheduled telemetry pass.

The design shall be flexible enough to autonomously handle unexpected situations onboard in the following most critical phases of the mission:

- Jupiter and Europa insertions, where a failure could lead to spacecraft loss;
- All gravity assists where inaccurate trajectories would lead to a prohibitive propellant cost;
- Europa fly-bys and orbit around Europa where JEO collision with moon should be avoided before end of mission.

Applied techniques shall be sized with the needs, and the possibilities of validating them on ground shall be guaranteed at an acceptable cost.

3.6 Planetary Protection

COSPAR rules [7] impose a probability of a Europan ocean contamination<1.10^{-4}.

After orbit insertion, it is a fact that JEO hasn't enough capacity to avoid a final Europa surface collision. Therefore, to mitigate contamination risks a two step approach is proposed:

1^{st} step: Jovian radiations are considered to clean the JEO satellite external surfaces of any biological element. Indeed, with a total time in orbit of 120days before final collision with Europa surface, 10Mrad are received behind 4mm Al.

2^{nd} step: for JEO radiation protected equipments, specific integration processes on ground in a class-100 room are required. Bioburden reduction can be performed by various means: dry heating, radiation sterilization, or Hydrogen Peroxide Gas Plasma. Before mounting on the platform, these equipments are eventually sealed in a box to avoid any Earth backward contamination.

Consequently to that approach, collision risks during fly-bys and at insertion impose a reliable and accurate autonomous navigation, with avoidance manœuvres in case of major failures.

4 SATELLITE CONCEPTUAL DESIGN

4.1 Science Payload

A preliminary assessment of strawman science instrument packages for JEO & JRS, was performed to determine requirements for new technologies together with the scientific interest potential of such a mission in terms of amount of data collected, accuracy and types of instruments that can be implemented on-board [8][9]. Having an interactive process between the science instruments and the system design in such feasibility study permits to rapidly identify the possibilities.

A highly integrated payload approach for JEO & JRS was considered to optimise the masses. Tables 4 & 5 illustrate the capacities of the built scenario, and are ready for a deeper scientific expertise. On JEO, the ground penetrating radar is assumed to operate alternatively with the other instruments.

Instruments	Mass (kg)	Power (W)		Data rate (kb/s)
Ground penetrating radar	11.5		25	28
Stereo Camera	0.6	1.2		5
Visible-Near Infrared spectrometer	2.0	2		13
Radiometer	2	1		0.1
Magnetometer	1.4	0.5		0.3
Laser Altimeter	2	2.5		3.0
Radiation Monitor	1.5	1		1.1
γ and Neutron spectrometer	3.1	1		To be defined
Digital processing unit	2.5	4.0	4.0	
Structures & Shielding	6.2			
Margins (20%)	6.6	2.6	5.8	
Total	39.4	15.8	34.8	25 to 30

Table 4: JEO preliminary science instruments package

Instruments	Mass (kg)	Power (W)	Data rate (kb/s)
Radiation Monitor	1.5	1	1.1
Plasma wave instrument	3.5	1.6	3.8
Narrow camera	1.5	1	9.1
Magnetometer	1.4	1.0	0.3
Dust detector	1	1	0.02
Digital processing unit	2.5	4.0	
Structures & Shielding	4.3		
Margins (20%)	3.1	1.9	
Total	18.8	11.5	14

Table 5: JRS preliminary science instruments package

Thanks to JRS arrival on its orbit 100 days before JEO, and to its lifetime going beyond JEO science data transmission to Earth, a Jupiter science mission can easily be envisaged. Moreover, power for science is not an issue on-board JRS since power resources can be used alternatively for communication and science.

4.2 JMO design drivers

Table 6 presents an overview of the spacecraft design trade-offs carried out in the frame of the JMO study. The propulsion system and the highly integrated avionics are further discussed in sections 4.3 & 4.4.

Function	Elements to be traded	Design selection
Command & Data Handling System	Strategy during interplanetary transfer	JRS as master and JEO as slave
	Reliability	redundancy strategy, protection to radiation, functions sharing between the different CPUs
	Mass memory	50Gbits for JEO & 256Gbits for JRS
Attitude & Orbit Control System	Equipments high integration for shielding mass optimization	Shielded Highly Integrated Avionics box including all radiation sensitive electronics
	Power optimization	Low power bus
	Autonomous navigation	Navigation camera
	Mass optimization	Attitude control with wheels (1Nms)
	JEO nadir pointing accuracy	Need for a Europa horizon sensor to be further investigated
	Recurrency optimization	Same avionics on both JEO & JRS
Propulsion System	Large DeltaV	500N main engine on JRS
	Gravity losses vs hardware mass	22N Leros thrusters on JEO, Isp=308s. No main engine.
	JEO accommodation on top JRS	
	Cost	Off-the-shelf E2000+ tanks on JRS
Thermal Control System	Highly varying fluxes from Venus vicinity to Jupiter	Standard thermal control. Need for fluid loops to be further investigated
	Limited power resources	Need for local RHUs to be further investigated
Power System	Low Solar input flux	Solar array with concentrators
	Solar cells degradation by radiations	LILT triple-junction GaAs, 500µm coverglass
	Recurrency optimization	Same solar arrays on JEO & JRS
	High fluxes in Venus vicinity	Si cells + 25% OSR on JRS solar array back side used at distance from Sun<1AU
Commu-nications System	Data rates with low power resources	Ka-band selection for science data transmission
	Reliability	fixed HGA preferred
	Recurrency optimization	Same transponder & HGA on JEO & JRS
	Possibility to use JRS HGA or JEO HGA during cruise	Impact on JRS/JEO electrical interface

Table 6: JMO design trade-offs overview

4.3 JMO propulsion system

The JRS propulsion system, shown in Figure 9, is used for Earth departure, interplanetary transfer, Jupiter capture, JRS Jupiter tour and station keeping. It is a conventional 4-tank MMH/NTO bi-propellant system, using EADS Eurostar 2000+ (Hotbird 2) tanks with a capacity of 393l each. and an EADS 500N main engine, under development in Germany.

The JEO propulsion system, shown in Figure 10, is used for JEO Jupiter tour, Europa orbit insertion and station keeping. It is a dual-mode system using N_2H_4 as fuel, and MON-3 (N_2O_4) as oxidiser. It features two 72l fuel tanks, one 85l oxidant tank, and four redunded ARC UK Ltd 22N thrusters (Leros 20H), with Isp of 308s, under development in UK.

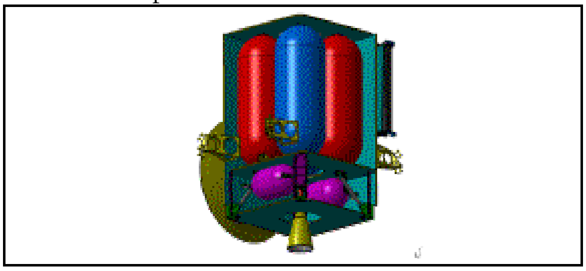


Figure 9: JRS propulsion system

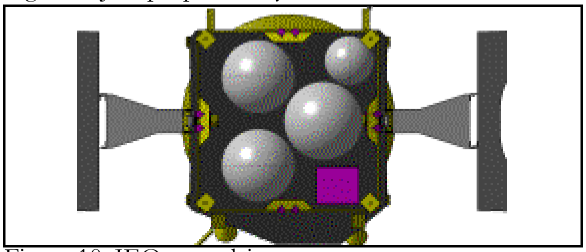


Figure 10: JEO propulsion system

4.4 JMO avionics

Having a highly integrated avionics has a double advantage for JMO: it reduces the mass and the hardware volume. Reducing the volume enables to limit the room to be shielded against radiation, and thus the shielding material mass.

The reference data handling architecture selected for both JEO and JRS is the Bepi Colombo computer. Functional enhancements are considered, such as integration of the star tracker electronics, the navigation camera electronics and the inertial measurement unit. In addition, the Power control & distribution unit board becomes part of the avionics box. A technological enhancement is also considered with the replacement of the standard 1553 bus by a low power 1553 (preferred for compatibility) or CANBus, or even Spacewire.

The avionics box functions and interfaces are summarized in Figure 11.

The estimated size of the avionics box is 800x200x250mm based on Double Europe format (233x160x20mm) boards. For JEO, assuming a wall thickness of 7mm, the aluminium shielding box weights 15.5kg.

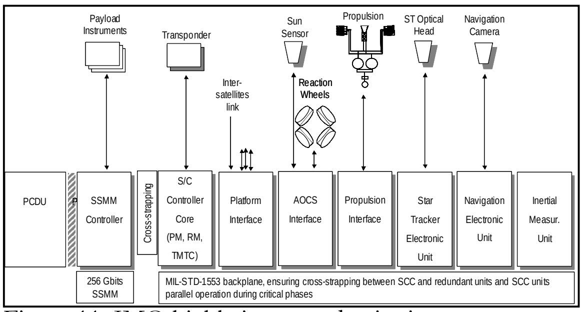


Figure 11: JMO highly integrated avionics

4.5 JMO configuration

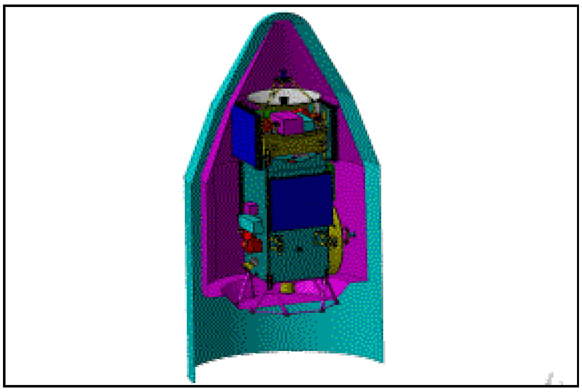


Figure 12: JMO in launch configuration with JEO mounted on top JRS

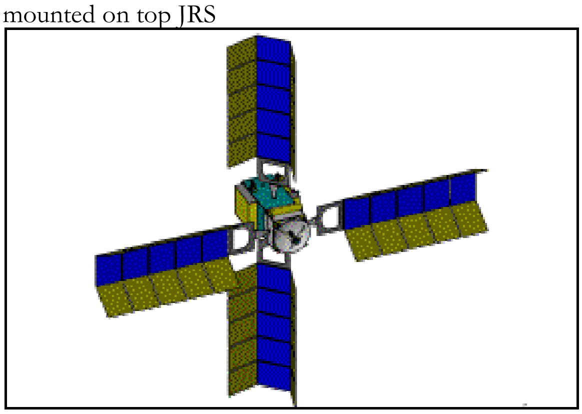


Figure 13: JMO in cruise configuration with solar panels and concentrators deployed

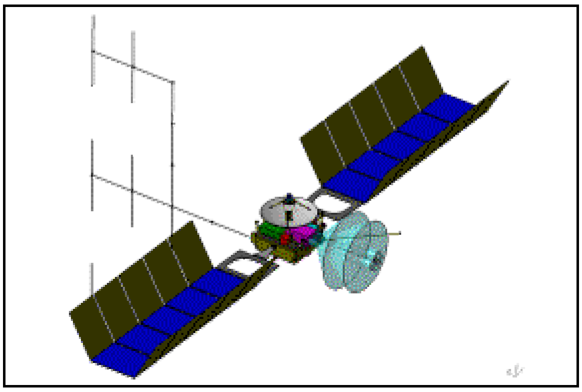


Figure 14: JEO in cruise configuration with ground penetrating radar deployed

4.6 JMO budgets

Table 7 is a summary of the satellite subsystems masses, without payload.
The JMO system mass budget, presented in Table 8, shows the science payload maximum mass capacity with maximum launch mass.

	JRS	JEO
Power	110kg	106kg
AOCS	8kg	8kg
Propulsion	135kg	40kg
CMDS	26kg	26kg
Communications	42kg	25kg
Structure	145kg	73kg
Thermal	10kg	6kg
Radiation shielding	8kg	27kg
Total	**484kg**	**311kg**

Table 7: JRS & JEO masses with maturity margins

JRS platform	580kg
JRS science instruments	14kg
JRS propellant	1679kg
JRS wet mass	**2274kg**
JEO platform	373kg
JEO science instruments	30kg
JEO propellant	254kg
JEO wet mass	**656kg**
Total Launch mass (without adapter)	**2930kg**
Launcher adapter	70kg
Launcher capacity	**3000kg**

Table 8: JMO system mass with 20% system margin

The limited power resources led to the optimisation of the consumptions. Assessment of these consumptions were based either on equipments under development or on potential improvements to occur in a short to mid-term.
Table 9 gives an overview of JEO & JRS power needs in worst cases. It appears that, due to differential cells degradations on both satellites, JEO & JRS can be designed with the same solar array of 14.7m² (excluding concentrator areas).
Most JRS payloads operate outside of communications windows with Earth. This means

that power available for communications becomes available for science.

	JRS	JEO
Power	10W	67W
AOCS	23W	23W
CMDS	29W	29W
Communications	123W	11W
Thermal	54W	28W
Harness losses	8W	5W
Payload	2W	30W
Total with 20% system margin	**359W**	**270W**

Table 9: JMO power budget with 20% system margin

5 DEVELOPMENT PLAN

The previous chapters show that the feasibility of a JMO mission depends on a limited number of new technologies summarized in Table 10.

Technology	Development activity
Electrical	
Rad-hard components	Specify, design & qualify 1Mrad tolerant components, common to payload, avionics & communications systems
Shielding material	Specify, design & qualify radiation shielding structure for electronic housing enclosures (avionics, payloads, communication system)
Rad-hard avionics box	Study & develop an integrated avionics box concept bread-board, specified to operate up to a 1Mrad dose and aiming at low total mass (electronics + radiation shielding enclosure)
Power	
GaAs cell for LILT & harsh radiation environment	Delta-qualify cells for the specified environment
Solar concentrators	Specify, design and qualify one solar panel with concentrators & deployment mechanisms (with ground test in solar simulator chamber)
RF communications	
High data rate receiver (3 Mbps)	Design & development of a bread-board transponder
High efficiency Ka SSPA (30% @ 3.5 W RF)	Design & development of a bread-board with specific components (e.g. FPGA)
Avionics	
Software architecture for high autonomy	Design and validation on numerical system simulator
Autonomous optical navigation & small correction manoeuvre scheduling	Camera Bread-board + RT system simulator with hardware in the loop. Specify, develop & validate algorithms for optical navigation within Jovian system.

Table 10: Technology development activities

Should a launch be considered in 2016, the development schedule could look like Table 11.

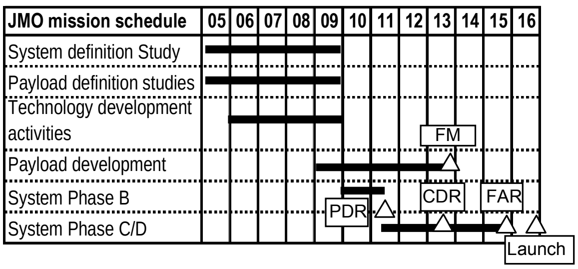


Table 11: JMO development plan

6 CONCLUSION

The feasibility of a valuable scientific mission to Jupiter/Europa with two mini-satellites, launched by Soyuz-Fregat from Kourou, and powered by solar generators is demonstrated by this technical assistance study. There is a good confidence in the final result due to:

- focus on a low cost, reliable, technically sensible mission ensuring good science return;
- a full coverage of the mission permitting to identify major system concept drivers;
- a preliminary payload assessment feeding the system with science considerations in its very early stage;
- the presentation of a coherent and sensible scenario;
- a rigorous margin philosophy guaranteeing flexibility in the scenario;
- a proposed system mixing conservative approaches and innovative solutions based on EADS-Astrium experience in scientific missions (Rosetta, Mars Express) and on its technical expertise;
- a limited number of enabling technologies.

JMO development plan remains however very challenging for the European community. A firm commitment is required on enabling technologies.

7 REFERENCES

1. Development of the JMO TRS: Part 1 of the JME, P.Falkner, ESA/ESTEC 2003, SCI-A/2003-162/TP/PF
2. Galileo: the tour guide at: http://www2.jpl.nasa.gov/galileo/tour
3. Europa TRM Mission Analysis, Technical notes 32, 33 and 34, M.Khan, S.Campagnola, & M. Croon, ESA/ESOC, Mission Analysis Office, 2003
4. LILT qualification test results of Silicon 10LITHI-ETA^{R}3 solar cells, C.Signorini, E.Fernandez, H.Fiebrich ESA/ESTEC, G.D'Accolti, Galileo Avionica, T.Gomez, L.Pazos, INTA Spasolab, G.Strobl, RWE Solar GmbH
5. Free space optical communications at JPL/NASA, Hemmati, 2003
6. Deep space one: Nasa's first deep space technology validation mission, M/D/Rayman & D.H.Lehman, JPL, IAF-97-Q.5.05

7 COSPAR Planetary protection policy, October 2002

8. Science requirement document for the Jupiter Minisat explorer, A.Atzei, SCI-A/2004.069/AA

9 JME Payload Definition Document, S.Kraft, Cosine, CR-PTRM-JME-PDD-Issue-03, Sep. 2004

55th International Astronautical Congress, Vancouver, Canada, 2004

THE INTERSTELLAR HELIOPAUSE PROBE

A. Lyngvi[1], P. Falkner[1], S. Kemble[2], M. Leipold[3] and A. Peacock[1]
[1]Science Payloads & Advanced Concepts Office, European Space Agency, ESTEC, The Netherlands
[2]EADS Astrium Limited, Stevenage, England
[3]Kayser-Threde GmbH, Munich, Germany

ABSTRACT

The Interstellar Heliopause Probe (IHP) is one of four Technology Reference Studies (TRS) introduced by the Planetary Exploration Studies Section of the Science Payload & Advanced Concepts Office (SCI-A) at ESA. The overall purpose of the TRSs is to focus the development of strategically important technologies of likely relevance to future science missions. This is accomplished through the study of several technologically demanding and scientifically interesting missions, which are currently not part of the ESA science programme. The TRS baseline uses small satellites (~ 200kg), with highly miniaturized and highly integrated payload suites. By using multiple low resource spacecraft in a phased approach, the risk and cost, compared to a single, high resource mission can be reduced.

Equipped with a Highly Integrated Payload Suite the IHP will answer scientific questions concerning the nature of the interstellar medium, how the interstellar medium affects our solar system and how the solar system impacts the interstellar medium.

This paper will present an update to the results of the studies being performed on this mission. The current mission baseline and alternative propulsion systems will be described and the spacecraft design and other enabling technologies will be discussed.

1. INTRODUCTION

Technology Reference Studies (TRS)[1] have been introduced as a tool to identify future technology needs and enable strategic technology development. TRSs are characteristically challenging missions in which critical technologies have yet to be identified. By introducing the TRSs the Science Payload & Advanced Concepts Office (SCI-A) ensures long-term technology developments within the science directorate to facilitate future science missions.

The interstellar medium is one of the frontiers of future space exploration and extreme challenges are imposed on a mission to reach the required distance for in-situ measurements of this medium. In the Interstellar Heliopause Probe (IHP) TRS a feasible mission concept is being developed, enabling a mission to investigate the outer heliosphere, the interstellar medium and the interface region between them.

Several missions to the heliopause have already been proposed. In the early 1980’s the ”Thousand Astronomical Units” (TAU) mission[2] was proposed based on a 1 MW nuclear powered electrical propulsion system. Later missions such as the Heliopause Explorer[3], and the Interstellar Probe[4,5] where studied using solar sails. All these studies have slightly different mission profiles in order to obtain the required distance from the Sun. The goal of the IHP TRS is to identify requirements for future technology developments that will enable such a mission, making it possible to reach a distance of 200 AU within 25 years transfer time.

2. SCIENTIFIC OBJECTIVES

The Heliosphere contains the plasma that originates from the Sun. This region is formed and structured by the Local Interstellar Medium (LISM), the solar wind and the relative motion of the Sun with respect to the LISM.

The heliopause separates the solar plasma from the interstellar plasma and can be considered to be the boundary between the interstellar medium and the heliosphere. The heliopause is located between the solar wind termination shock and bow shock in the LISM (Figure 1). These two shock surfaces terminate the undisturbed supersonic flows of the solar wind and LISM respectively and represent the inner and outer boundary of the heliospheric interface. In this region the IHP will make in-situ measurements to answer the questions:

- *What is the nature of the interstellar medium?*
- *How does the interstellar medium affect the solar system?*
- *How does the solar system impact the interstellar medium?*

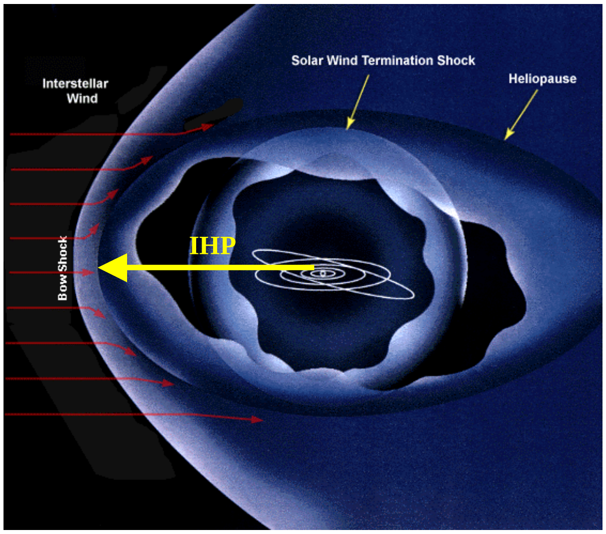


Figure 1: The heliosphere in the LISM

In order to answer these questions the IHP needs to make in-situ measurements continuously while travelling from the outer heliosphere and into the interstellar medium. Each of these regions will have different scientific interest.

2.1 The Outer Heliosphere

The Solar System was formed approximately 4.6 billion years ago. Still, the origin and the evolution of the solar system are fairly unknown. Collisions play a central role in the formation and evolution of planetary systems. The present interplanetary dust population is a result of collision processes occurring in the solar system. By studying interplanetary and interstellar dust the IHP will help to understand the origin and nature of our solar system and hence other planetary systems.

2.2 The interface region

The location of the termination shock and the heliopause are yet to be known exactly. Determining the location of these areas as they vary with solar and interstellar pressure is one of the key objectives of the IHP.

Anomalous cosmic rays are particles accelerated in the termination shock. How these particles are accelerated is not yet fully understood. Hence, getting a better understanding of this process is also an important scientific goal for the IHP.

2.3 The Interstellar Medium

The Sun is thought to be located near the edge of a low-density interstellar cloud (~0.3 cm^{-3}). In order to establish an understanding of the LISM's nature, a series of measurements should be made in this region. The IHP will facilitate the derivation of the physical properties of the LISM and investigate astrophysical processes such as acceleration by supernova shock waves, interstellar radio heating and dynamics of the interstellar medium.

3. PAYLOAD

A common building block of all the TRSs in the Planetary Exploration Studies Section of SCI-A is the Highly Integrated Payload Suite (HIPS) concept[6]. A HIPS reduces the overall resources (i.e. mass, power and volume) by sharing common structures and payload functionalities, such as power supply and processors, and by using miniaturized sensors and components, such as stacks, 3D electronics, etc.

By performing the measurements described in Table 1 the IHP will be able to meet the scientific objectives. The mass of this payload, excluding the secondary instruments, is about 22 kg and the power requirement is less than 15 W. Due to sequential operation of the instruments the continuous payload power demand could be less than 10 W.

4. PROPULSION SYSTEM

To keep total cost of the mission reasonable the IHP shall be compatible with a launch on a Souyz Fregat 2B launch vehicle from Korou. This allows for a total launch mass approaching 2000 kg to a low energy Earth escape orbit.

P/L	**Purpose**	**Mass (kg)**	**Power (W)**
Plasma Analyser	Elemental and isotopic composition of plasma and the associated energy levels at temporal composition	2.0	1
Plasma radio wave experiment	Plasma and radio waves experiment	5.5	2.5
Magnetometer	Magnetic field measurements in very low fields	3.2	2.5
Neutral and charged atom detector	Energy levels, composition, mass, angular and energy distribution of neutral atoms	0.5	1
Energetic particle detector	Energy levels of cosmic rays	1.8	1.2
Dust analyser	Energy levels, mass and composition of dust particles	1.0	0.5
UV-photometer	Hydrogen density	0.3	0.3
FIR spectral imager*	Measurement of the radial distribution of dust and the cosmic infrared background	(0.3)	(0.2)
VIS-NIR imager*	Determine the radial distribution of Small Kuiper belt objects and TNO	(1.0)	(0.5)
DPU + CPS	*Data processing and power supply*	*2*	*3.5*
Structures	*Optical bench and mounting structures*	*2*	-
System Margin (20 %)		*3.7*	*2.6*
Total		**22.0**	**14.9**

Table 1: Tentative payload for IHP. *) Instruments are only secondary and are hence currently not part of the overall mass and power estimate

To reach the interstellar medium in the shortest possible time the spacecraft will have to be launched in the direction of the heliosphere nose, which is located at 7.5° latitude and 254.5° longitude in the ecliptic coordinate frame.

Three different propulsion systems where identified as potential candidates for the IHP TRS, chemical propulsion, Nuclear Electric Propulsion (NEP) and solar sailing. Each of these propulsion systems were investigated in order to find the most feasible alternative with the given requirements and constraints.

A major factor for all propulsion system types considered is the desire to achieve the transfer to the heliopause within some maximum mission duration, typically 15-25 years.

4.1 Chemical Propulsion

The chemical propulsion system could use one or two Earth gravity assists to reach Jupiter and then employ a close solar flyby with a propulsive manoeuvre to achieve the required Solar system excess hyperbolic speed. The two Earth gravity assist route is used in conjunction with a preceding Lunar or Venus gravity assist. A Lunar-Earth-Earth-Jupiter-Sun (LEEJS) trajectory is shown in Figure 2. The solar approach would have to be as close as 4 solar radii to obtain the Delta-V required to reach the distance to the heliopause. However, even at this close distance to the Sun current chemical propulsion systems would not have sufficient specific impulse to provide more than approximately 50 kg of useful mass (i.e. mass of spacecraft excl. propulsion system).

With new developments in high thrust propulsion systems such as nuclear thermal propulsion this trajectory could become feasible. However, the thermal requirements for such a mission would be extremely challenging and hence this alternative was discarded as an option for IHP.

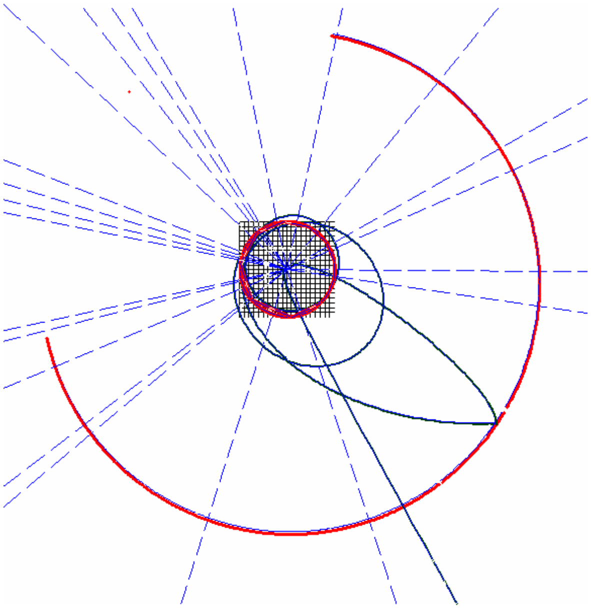


Figure 2: Chemical propulsion trajectory with close solar flyby (LEEJS)

4.2 Nuclear Electric Propulsion (NEP)

The second propulsion system identified as an option was NEP. The best launch scenario would then be an LEEJ gravity assist sequence. The number of gravity assists would be limited due to the cruise time constraint to 200AU. More gravity assists reduces the ΔV needed to achieve a high energy Jupiter crossing orbit, but requires a greater ΔV after passing Jupiter because the remaining cruise time is reduced.

An example of such a transfer is shown in Figure 3. After a close Jupiter fly-by, achieving a moderate Solar system excess hyperbolic speed, an extended low thrust phase is required to achieve the transfer in the required timescale.

The thrust required for a NEP mission is dependent on the time that the thrusters are on (thrust time). The longer the thrust time the less thrust is required. For instance a 20-year thrust sequence after JGA will require a thrust of between 53-68 mN/tonne and a 10-year thrust after JGA will require about 77-95 mN/tonne. The specific impulse that was investigated was between 5000 s and 20000 s.

The thrust and specific impulse necessary for this option would require a large amount of power (i.e. between 5 kW and 20 kW). To produce such power at distances beyond Jupiter orbit requires power systems that are currently not available. In a best-case scenario with a 20-year thrust time the required specific power for these systems would be in excess of 10 W/kg. Current Radioisotope Power Systems (RPS) are not capable of providing specific power of this magnitude. Even if high specific power RPS were developed, the required amount of radioisotope material would be too great to make NEP a viable power system solution for IHP. The only alternative remaining would be to use a nuclear reactor. If a nuclear reactor were developed, the required specific power could be possible to obtain. However the total power system mass is severely constrained due to the extremely large Delta-V required and cannot be more than approximately 500 kg for IHP. To develop a nuclear reactor capable of meeting the IHP requirements and constraints will therefore be extremely challenging, as the nuclear reactor is difficult to scale down in mass while keeping the specific power high. Therefore even a 20 kW reactor would have a mass in excess of the maximum available mass for the IHP.

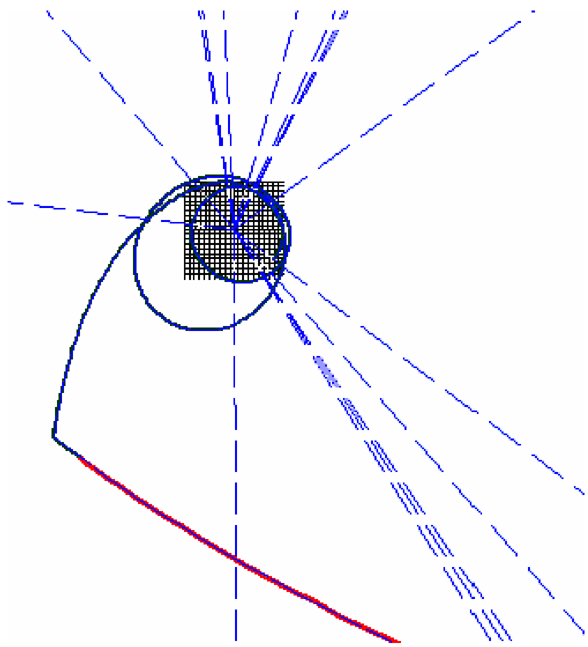


Figure 3: EEJ gravity assist trajectory using Nuclear Electric Propulsion.

Another issue with the NEP option is the thrust time. The power system would have to be even larger for thrust times of less than 20 years as a higher thrust is needed. Current electrical propulsion systems are quite far from obtaining 20-year continuous thrust lifetime and even a 10-year lifetime will require substantial development.

Based on this careful assessment the NEP option was considered to be infeasible with the current TRS constraints.

4.3 Solar sailing

The third, and currently most feasible propulsion technology given the constraints and requirements of the study is solar sailing. Hence this is the baseline propulsion system for the IHP. There are many different solar sail configurations and currently two are being investigated for IHP; a square sail and a spinning disk sail.

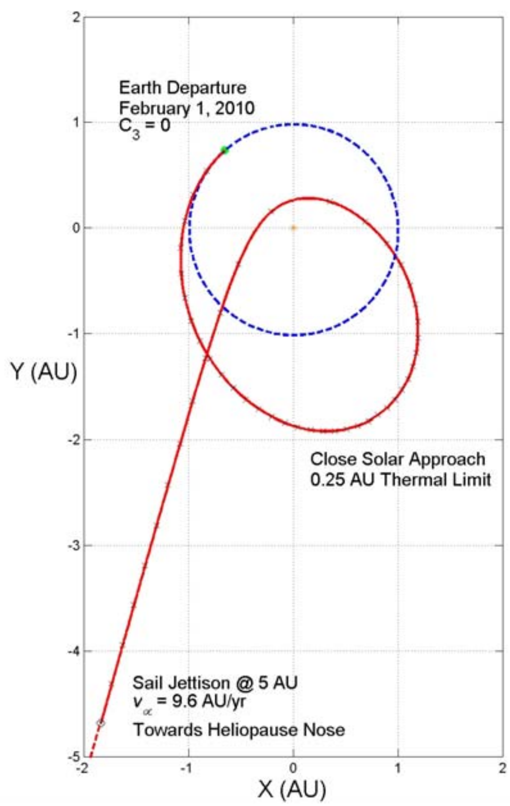


Figure 4: Trajectory for IHP with a characteristic acceleration of 1.5 mm/s^2

Solar sails utilize the momentum of photons to obtain a very low acceleration. However, since no propellant is being used the propulsion system is very effective although very large structures are needed. For the square sail scenario a sail size of about 260 m x 260 m and a sail thickness of 2 μm is needed. At 1 AU this sail size will give us a characteristic acceleration of 0.85 mm/s^2, which will greatly increase as the probe travels closer to the Sun. Hence, all the solar sail alternatives for the IHP have to capitalize on this effect by first travelling closer to the Sun to get the required acceleration to reach 200 AU in 25 years.

A spinning sail without rigidizing structure is lighter compared to the square sail option, which requires booms. Since the mass of the sail can be reduced in this configuration, a smaller sail could be used to obtain the same acceleration as the square sail. The current spinning disk sail scenario utilizes a sail with a radii of approximately 140 m and a sail thickness of 1 μm, this gives a characteristic acceleration of 1.5 mm/s^2. The trajectory for this configuration is shown in Figure 4.

5. IHP SPACECRAFT

Because most of the instruments described in Table 1 require a 4π field of view the IHP spacecraft is spinning. The subsystem mass breakdown of IHP is given in Table 2. These numbers were obtained assuming several technology developments such as within the power system, where a specific power of approximately 10 W/kg has been used.

System	**Mass (kg)**
Science instruments	22
Attitude Determination and Control	35
Telemetry, tracking and command	61
On-board data handling	12
Thermal Control	14
Power	42
Mechanisms and structure	27
Total mass	213

Table 2: Current subsystem mass breakdown

The spacecraft dry mass of the IHP is similar for both solar sail configurations. However, the sail masses differ significantly. The square sail is much heavier than the disk sail, mainly due to the mass of the boom structures and the thickness of the film (Table 3). The total solar sail mass includes the spin-up mechanisms using cold gas thrusters on thrust arm booms and the deployment mechanism, which is quite different for the two concepts. The overall mass including margin for both sails is well within the capabilities of a Souyz Fregat.

Sail configuration	**Spinning**	**Square**
IHP Spacecraft Dry Mass	213 kg	213 kg
Solar Sail System Mass	311 kg	492 kg
System margin (20 %)	104 kg	141 kg
Total launch mass	**628 kg**	**846 kg**

Table 3: Launch mass for different sail configurations

5. TECHNOLOGY DEVELOPMENT

The IHP TRS faces several challenges. However the purpose of the study is to identify these challenges and potentially develop technologies that will enable such a mission in the future.

5.1 Solar sail technology developments

The biggest challenge that a mission like the IHP will face is the development of an adequate propulsion system. The development of solar sails for the IHP will require great advances from current available technologies. Presently the largest sail deployed has been the ESA/DLR ground deployment test[7]. There have been other deployment demonstrations as well, such as a small spinning disk sail deployment[8] and the in-orbit solar sail deployment on the Progress vehicle[9].

Due to the close approach to the Sun the IHP will have very stringent thermal requirements. The minimum distance to the Sun is currently set at 0.25 AU, which implies that the solar sail will have to withstand a solar flux 16 times greater than at the Earth. The solar sail will be jettisoned at 5 AU, which means that the solar sail is used for a period of close to 5 years. Within those 5 years it is important that the sail keeps its optical properties, since the performance of the sail is directly dependent on the reflectivity of the sail material.

Solar sails will also require a reliable deployment mechanism. This will be dependent on the chosen sail configuration (i.e. disk or square sail). For the smallest sail option using a spinning disk sail the area of the sail is as large as 50 000 m^2, which will be very challenging to deploy without causing ruptures, damage to the coating, etc.

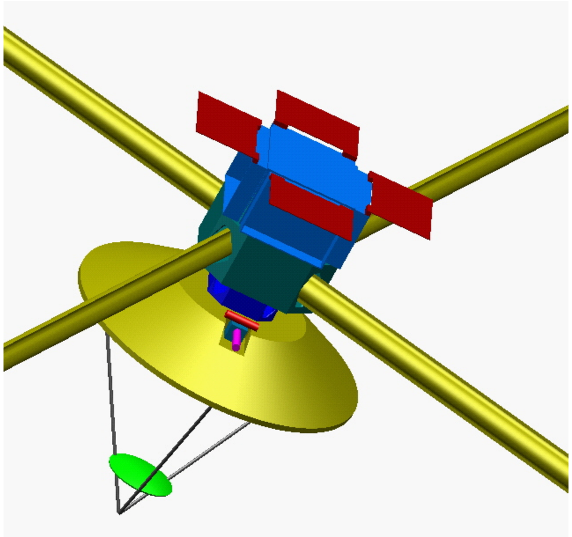


Figure 5: Potential configuration of the IHP

Designing an Attitude Determination and Control System (ADCS) for the spacecraft sailing phase will also be challenging. Several alternatives exist, such as gimballed boom between sail structure and spacecraft bus, tip vanes or thrusters on booms. A suitable alternative for this system must be identified and developed in order to enable a mission such as the IHP. This system will be dependent on whether a spinning or a three axis stabilized sail is chosen.

If the square sail configuration is used then there will be a need for further development of boom technologies. Current booms in Europe have a specific mass of ~100 g/m, this implies a large mass penalty compared to lower mass booms.

Using solar sails for a mission like IHP also requires development of a jettison mechanism that could safely jettison the sail from the spacecraft after 5 years with minimum risk for collision with the spacecraft structure.

5.2 Additional technology developments

5.2.1 Power

The power system that will be used for the IHP will most likely utilize conversion of radioisotope thermal energy. Current Radioisotope Thermoelectric Generators (RTG) such as the General Purpose Heat Source (GPHS) RTG, have a specific power of about 5.1 W/kg and conversion efficiency of 6.6 %[10]. The solar sail principle greatly benefits from general mass reductions as the acceleration is based on conversion of momentum. An improvement to the specific power of current RPS will help reduce the solar sail size of the IHP. The current estimated specific power needed for IHP is 10 W/kg. Figure 6 shows how the specific power influences the solar sail size for the spinning disk sail configuration.

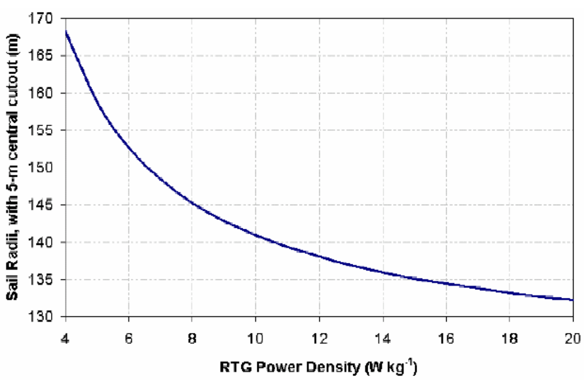


Figure 6: Solar sail size as a function of specific power

5.2.2 Communication

The communication system for the IHP will be limited to downlink an average data rate of approximately 200 bps at 200 AU. In the current IHP profile an RF communication system has been selected. However, both RF and optical communication systems are being assessed for the IHP in the ongoing study.

If optical communication is chosen, issues such as acquisition strategy will have to be solved. Current optical communication systems use a beacon strategy to communicate. Light takes 13 hours to travel a distance of 100 AU. Hence using a beacon is not a suitable acquisition strategy for the IHP. In addition to an acquisition strategy, lightweight components and long lifetime lasers will need to be developed in order to make the optical alternative feasible.

The other alternative for a communication system is RF communication. This requires a large antenna size and high power levels. This results in a high overall mass of the system (in excess of 60 kg total). The IHP will therefore greatly benefit from development of lightweight antenna structures and highly efficient power amplifiers.

5.2.3 Lifetime and autonomy

The long lifetime of the spacecraft sets strict requirements on all subsystems. Due to the long travel distance of 200 AU the lifetime of the IHP must be more than 25 years hence each of the subsystems will have to be designed for this duration.

Satisfying these long lifetime constraints will require innovative ways of making the spacecraft fault-tolerant and provide significant redundancy. Furthermore, as operations cost is traditionally a large portion of the overall mission cost a reduction of the manpower required to operate the spacecraft over such a long time period is of paramount importance. This will require, due to long transmission and response times, a large degree of onboard autonomy.

6. CONCLUSION

Technology Reference Studies have been introduced by ESA’s Science Payload & Advanced Concepts Office to identify critical technologies of likely relevance to future science missions. By studying challenging future mission concepts where enabling technologies have yet to be identified the individual TRSs can provide a guideline for future technology development activities.

The Interstellar Heliopause Probe (IHP) is one of these technology reference studies. It has identified a feasible propulsion option and identified several technologies and technology areas in which developments are required to enable such a mission profile. The ongoing study of the IHP will provide additional and consolidated requirements for these technologies, which also can benefit other missions such as outer planet missions, which are sharing very similar technologies.

7. ACKNOWLEDGEMENTS

The authors would like to acknowledge the work by Colin McInnes, Malcolm Macdonald and Gareth Hughes from University of Glasgow.

8. REFERENCES

1. A. Lyngvi, P. Falkner, A. Atzei, D. Renton, M. L. v. d. Berg and A. Peacock. Technology Reference Studies, proceedings of IAC Vancouver 2004

2. Etchegaray M.I., Preliminiary Scientific Rationale for a Voyage to Thousand Astronomical Units, JPL Publication 87-17.

3. Leipold M., Fichtner H., Heber B., Groepper P., Lascar S., Burger F., Eiden M., Niederstadt T., Sickinger C., Herbeck L., Dachwald B. and Seboldt W., Heliopause Explorer – A Sailcraft Mission to the Outer Boundaries of the Solar System, Fifth IAA International Conference on Low-Cost Planetary Missions,ESA to be published in SP-542, 2003.

4. Liewer P., Mewaldt R.A., Ayon A., Garner C., Gavit S. and Wallace, R.A., Interstellar Probe using a Solar Sail: Conceptual Design and Technology Challenges, COSPAR Colloquium Series Volume 11, Proceedings of the COSPAR Colloquium on The Outer Heliosphere: The Next Frontier, Potsdam, Germany, July 24-28, 411-420, 2000.

5. McNutt Jr R.L., Andrews G.B., McAdams J., Gold R.E., Santo A., Oursler D., Heeres K., Fraeman M. and Williams B., Low-cost interstellar probe, Acta Astronautica . Vol. 52, 267–279, 2003.

6. S. Kraft, J. Moorhouse, A.L. Mieremet, M. Collon, J. Montella, M. Beijersbergen , J. Harris, M.L. van den Berg, A. Atzei, A. Lyngvi, D. Renton, C. Erd, P. Falkner, On the study of highly integrated payload architectures for future planetary missions. Proceedings of SPIE 2004.

7. Leipold M., Eiden M., Garnerd C.E., Herbeck L., Kassing D., Niederstadt T., Krüger T., Pagel G., Rezazad M., Rozemeijer H., Seboldt W., Schöppinger C., Sickinger C. and Unckenbold W., Solar sail technology development and demonstration, Acta Astronautica, Vol. 52, 317-326, 2003.

8. Salama, M., White, C., Leland, R., 2003, “Ground Demonstration of a Spinning Solar Sail Deployment Concept,” Journal of Spacecraft and Rockets, Vol. 40, No.1, January-February 2003, pp. 9-14.

9. see http://www.space-frontier.org/Events/Znamya/

10. Hunt, M. E., High efficiency Dynamic Radioisotope Power Systems for Space Exploration – A Status Report, IEE AES Systems Magazine, Vol. 8, Issue 12, 18-23, 1993.

IAC-04-S.2.b.08

ANALYSIS OF A SOLAR SAIL MERCURY SAMPLE RETURN MISSION

Gareth W. Hughes*, Malcolm Macdonald*, Colin R. McInnes†
*Research Assistant, Dept. of Aerospace Engineering, University of Glasgow, Glasgow, UK
†Professor, Department of Mechanical Engineering, University of Strathclyde, Glasgow, UK
ghughes@aero.gla.ac.uk, m.macdonald@aero.gla.ac.uk, colin.mcinnes@strath.ac.uk

Alessandro Atzei, Peter Falkner
ESA/ESTEC, Science Payloads and Advanced Concepts Office, Noordwijk, The Netherlands
aatzei@rssd.esa.int, Peter.Falkner@esa.int

ABSTRACT

A conventional Mercury sample return mission requires significant launch mass, due to the large Δv required for the outbound and return trips, and the large mass of a planetary lander and ascent vehicle. Solar sailing can be used to reduce lander mass allocation by delivering the lander to a low, thermally safe orbit close to the terminator. In addition, the ascending node of the solar sail parking orbit plane can be artificially forced to avoid out-of-plane manoeuvres during ascent from the planetary surface. Propellant mass is not an issue for solar sails so a sample can be returned relatively easily, without resorting to lengthy, multiple gravity assists. A 275 m solar sail with an assembly loading of 5.9 g m^{-2} is used to deliver a lander, cruise stage and science payload to a forced Sun-synchronous orbit at Mercury in 2.85 years. The lander acquires samples, and conducts limited surface exploration. An ascent vehicle delivers a small cold gas rendezvous vehicle containing the samples for transfer to the solar sail. The solar sail then spirals back to Earth in 1 year. The total mission launch mass is 2353 kg, on an H2A202-4S class launch vehicle (C_3=0), with a ROM mission cost of 850 M€. Nominal launch is in April 2014 with sample return to Earth 4.4 years later. Solar sailing reduces launch mass by 60% and trip time by 40%, relative to conventional mission concepts.

INTRODUCTION

Mercury Science

Of the terrestrial planets, Mercury is the one of which we know the least, its location deep within the solar gravity well ensuring that spacecraft have been sent there infrequently. Mercury's unusual 3:2 spin-orbit resonance meant that the same side was imaged in each of the Mariner 10 flybys. Surface coverage is incomplete and the planet must be comprehensively mapped by an orbiter mission such as BepiColombo or Messenger, before a sample return mission can proceed and a landing site selected. There is no significant water or atmosphere, so that daytime temperatures can soar to 700 K, and plummet to 100 K at night, due to the slow spin period. The lack of CO_2 or H_2O in the atmosphere suggests that Mercury is either intrinsically volatile deficient, or is not out-gassing at a rate comparable to that of the Earth, and so is less geologically active.[1] Aside from the Earth, Mercury is the only terrestrial planet which is known to have an intrinsic, weak, magnetic field. This is produced either by an Earth-like magnetohydrodynamic dynamo in the core, or a remnant magnetic field in the rock, which could be evident in any surface samples returned. The high average density of 5.43 g m^{-3} could be due to the presence of Iron within the interior, perhaps generated by this Earth-like magnetohydrodynamic dynamo, consistent with electrical currents flowing in a molten core. Tectonically, unique compressive thrust faults called lobate scarps occur on a global scale, implying global compressive stresses in Mercury's distant past. Large impact basins on Mercury can also contain volcanic deposits, which suggests that there has been volcanic activity after the impact. Little is known about the surface geology, composition, and chemistry, therefore sample return would be of significant benefit. Radar reflection measurements appear to show volatile compounds, possibly water ice, at both poles, deep within the shadows of craters, but observations from Earth are difficult due to the proximity of Mercury to the Sun. The lack of any appreciable atmosphere means that very cold regions exist in polar craters, allowing radar-bright materials to remain.

Science Objectives

It is important to ascertain the surface age of Mercury to understand its geologic history. Accurate rock dating of Mercury surface samples is only possible on Earth. Due to the tenuous atmosphere, the entire descent must be via chemical propulsion. A high-latitude landing site is selected due to thermal constraints, and prior imaging of this site from the orbiter at a resolution of better than 1 metre per pixel is necessary. Even at high latitudes, landing in direct sunlight, or indeed in permanent shadow would be undesirable. A landing site within a suitable crater, in partial shade, but with some light reflected from the crater walls is preferable, with a sample drilled from a rock outcrop within the crater.[2] However, recent craters may be contaminated with material from their impactor, and should be avoided. Guided descent is employed for all but the last few metres of the descent, since the thruster plume would scorch the landing site, contaminating the surface regolith to be sampled. The stroke of the landing legs is used to absorb the remaining kinetic energy of surface impact.

Baseline science objectives for a Mercury sample return mission are therefore, to acquire a surface sample though a precision landing at a carefully selected high latitude landing site in partial shadow, within a suitably aged crater, with high resolution imaging for documentation during terminal descent. Sample pre-selection and pre-analysis will be conducted in-situ during landing site characterisation using a robotic arm and small mobility device (20 m range).[1] The primary science goal is to acquire 350 g of surface regolith. Mercury is not thought to be of direct interest to exobiology in the solar system, so planetary protection measures will be simpler than for Mars missions, more similar to lunar missions.

Solar Sailing

The extremely high Δv required for Mercury sample return can be met relatively easily by solar sails, since propellant mass is not an issue, significantly reducing launch mass. Lengthy multiple gravity assists are not required, and the launch window is always open in principle. Thermally-safe orbit precession at Mercury is possible using the continuous thrust. Solar sail performance is defined by the Characteristic Acceleration, the solar radiation pressure induced acceleration at 1 AU with the sail normal oriented along the Sun-line.[3]

PAYLOAD MODEL

A full and detailed solar sail payload has been defined and customised,[4] based loosely on an internal ESA Assessment Study,[1] with some aspects drawn from a NASA/JPL Team X report.[2] A trade-off of the optimum solar sail parking orbit at Mercury was conducted so as to minimise the Mercury Ascent Vehicle (MAV) Δv requirements. The use of an artificial Sun-Synchronous polar orbit at Mercury close to the planetary terminator,[5] can be effected to reduce the thermal loads on the orbiter through a constant precession of the line of nodes, enabling a longer orbiter stay time and much lower parking orbit. The characteristic acceleration of the sail in the parking orbit is defined by the parameters of the Sun-Synchronous orbit, and so as the acceleration is increased the Sun-Synchronous orbit can be increasingly circularised. Fig. 1 shows the effect of rendezvous orbit altitude on MAV launch mass. It is seen that ascent direct to the Sun-Synchronous orbit requires much more Δv than ascent to a circular orbit. A circular 100 km orbit was selected to minimise MAV Δv requirements, with the sail used to deliver the lander onto the 100 km orbit, after an initial 44 day science and landing site selection phase on a 100 x 7500 km forced Sun-Synchronous orbit, 10° ahead of the solar terminator. During sample acquisition, until after coplanar MAV launch, the sail rotates the circular 100 km orbit plane to rendezvous with the MAV orbit, before spiralling to escape.

The solar sail payload stack comprises a small cold-gas Sail Rendezvous Vehicle

(SRV), to conduct proximity manoeuvres when transferring the sample from the MAV to the ballistic Earth Return Vehicle (ERV) attached to the Sail Cruise Stage (SCS). The bi-propellant MAV and cold-gas SRV is mounted on the bi-propellant Mercury Descent Vehicle (MDV). The MDV has a large science platform and 0.4 m^2 Gallium Arsenide solar arrays. Fig. 2 shows the lander deployed with its landing legs extended. Tables 1-4 show the mass breakdown of the SRV, MAV, MDV, and SCS, respectively. An analysis of the spacecraft subsystems, shows a total spacecraft mass of 1905 kg, to support acquisition of 350 g of surface samples.

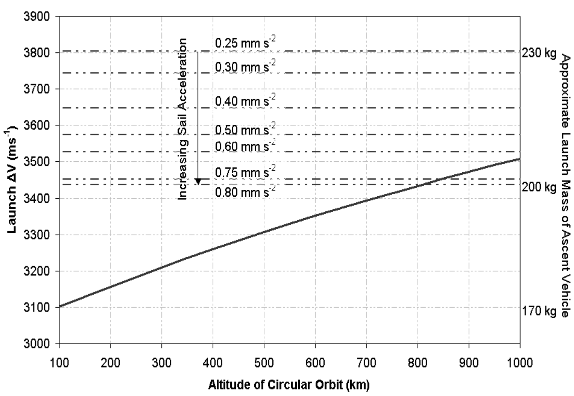


Figure 1: Mercury Ascent Vehicle rendezvous orbit trade-off (solid line: ascent to circular orbit, dashed lines: ascent to elliptical Sun-Synchronous orbit)

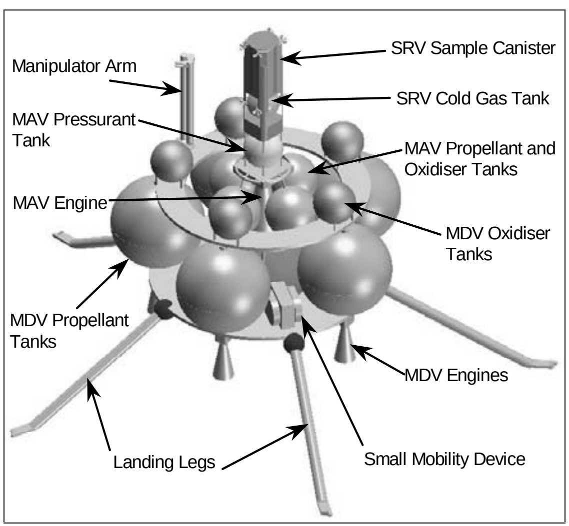


Figure 2: Mercury Sample Return lander

The SRV has a 2 kg sample container which holds the surface samples, with 50 m s^{-1} of propellant allocated for the rendezvous manoeuvre. The MAV uses a single stage DASA S3K class bi-propellant MMH/MON-3 engine, with a specific impulse of 352 s. However, volume reductions and an increase in thrust to 4 kN would be necessary. The MDV uses 5 bi-propellant MMH/MON-3 engines, delivering 6 kN each with a specific impulse of 320 s. The SCS allows for on-orbit power generation via 6.25 m^2 Gallium Arsenide solar arrays. The SCS telecommunications system comprises low and medium-gain X-band systems, a high-gain X/Ka band system, and a UHF link with the lander. The telecommunications systems have been sized to ensure adequate data return for the mission. A 28 volt, three domain, regulated power system is used. The SCS requires 332 W in Sunlight and 310 W during eclipse, met by 365 W 6.25 m^2 GaAs solar arrays, and 349 Wh Lithium-Ion batteries. The MDV requires 71 W, met through a 78 W 0.4 m^2 GaAs solar array. The 56 W MAV power requirement is attained through 53 Wh Li-Ion batteries. The SRV requires 24 W, provided by a 221 Wh Li-Ion battery over the SRV operational lifetime. The ballistic Earth Return Vehicle (ERV) uses a 41 Wh Primary Lithium battery to provide 1.7 W of power.

<u>Science Instruments</u>

The on-orbit SCS science payload includes a High Resolution Stereo Camera (10 W, 10-100 bps), Laser Altimeter (10 W, <1 bps), Infra-Red Radiometer (5 W, 100-5000 bps), X-ray Flourescence Spectrometer (10 W, 100-2000 bps), Radio Science Instruments (5 W, 10-100 bps), and associated high-capacity memory (5 W, 2-5 Gbytes). There is also an 8 kg allocation for a payload of opportunity (10 W, 5 kbps).

The lander has science instruments and manipulator hardware mounted on the MDV, which include a Sampling Device, Robotic Arm, and a small Rover vehicle. The total data rate of these instruments corresponds to 92 Mbit every 10 hours, with a total power consumption of 11.8 W.

SRV Component	Mass (kg)	Contingency (%)	Total mass (kg)
Sample container	2.0	-	2.0
SRV Payload Mass	**2.0**	**-**	**2.0**
Attitude control	3.1	10	3.4
Command & data	0.5	10	0.6
Power	2.0	10	2.2
Mechanisms	0.1	10	0.1
Telecomms	1.1	10	1.2
Thermal	1.0	10	1.1
Structure	2.0	10	2.2
SRV Bus Mass	**9.8**	**10**	**10.9**
Thrusters	0.2	15	0.23
Valves, pipes	0.1	15	0.1
Propellant tank	0.1	15	0.1
Propulsion Mass (Dry)	**0.4**	**15**	**0.43**
SRV Dry Mass	**12.2**		**13.3**
System contingency	-	1	0.1
Total SRV Dry Mass			**13.4**
Propellant for rendezvous	1.0	15	1.1
Total SRV Mass (Wet)			**14.5**

Table 1: Sail Rendezvous Vehicle (SRV) system sheet mass breakdown

MAV Component	Mass (kg)	Contingency (%)	Total mass (kg)
SRV	14.5	-	14.5
MAV Payload Mass	**14.5**	**-**	**14.5**
Attitude control	4.5	10	4.9
Command & data	2.5	10	2.7
Power	2.3	10	2.5
Mechanisms	0.5	10	0.6
Telecomms	0.0	10	0.0
Thermal	2.0	10	2.2
Structure	5.2	10	5.7
MAV Bus Mass	**17.0**	**10**	**18.6**
Thruster	15.0	15	17.3
Valves, pipes	2.9	15	3.3
Propellant tank	9.5	15	10.9
Propulsion Mass (Dry)	**27.4**		**31.5**
MAV Dry Mass	**58.9**		**64.6**
System contingency	-	1	0.65
Total MAV Dry mass			**65.3**
Propellant for Δv_1	0.5	15	0.6
Propellant for Δv_2	94.8	15	109.0
Total Propellant Mass	**95.29**	**15**	**109.6**
Total MAV Mass (Wet)			**174.9**

Table 2: Mercury Ascent Vehicle (MAV) system sheet mass breakdown

MDV Component	Mass (kg)	Contingency (%)	Total mass (kg)
MAV	174.9	-	174.9
Surface instruments	2.9	-	2.9
MDV Payload Mass	**177.8**		**177.8**
Attitude control	15.0	10	16.5
Command & data	4.0	10	4.4
Power	8.8	10	9.7
Mechanisms	22.0	10	24.2
Telecomms	0.0	10	0.0
Thermal	3.0	10	3.3
Structure	83.0	10	91.3
MDV Bus Mass	**135.8**	**10**	**149.4**
Thrusters (5 of 6kN)	50.0	15	57.5
Valves, pipes	8.3	15	9.5
Propellant Tanks	83.0	15	95.5
Propulsion Mass (Dry)	**141.3**	**15**	**162.5**
MDV Dry Mass	**454.9**		**489.7**
System contingency	-	1	4.9
Total MDV Dry Mass			**494.6**
Propellant for Δv_1	4.0	15	4.6
Propellant for Δv_2	830.8	15	955.4
Total Propellant Mass	**834.8**	**15**	**960.0**
Total MDV Mass (Wet)			**1454.6**

Table 3: Mercury Descent Vehicle (MDV) system sheet mass breakdown

SCS Component	Mass (kg)	Contingency (%)	Total mass (kg)
Lander (SRV/MAV/MDV)	1454.6	-	1454.6
Science payload	31.6	-	31.6
ERV	16.5	5	17.3
SCS Payload Mass	**1502.7**		**1503.5**
Attitude control	14.1	10	15.5
Command & data	10.0	10	11.0
Power	40.2	10	44.2
Mechanisms	161.0	10	177.1
Telecomms	24.6	10	27.1
Thermal	50.0	10	55.0
Structure	65.4	10	71.9
SCS Bus Mass	**365.3**	**10**	**401.8**
Total Sail Payload Mass			**1905.3**

Table 4: Sail Cruise Stage (SCS) system sheet mass breakdown

SOLAR SAIL SIZING

A square solar sail is envisaged, using tip-vanes for attitude control, sized to provide adequate slew rates for the planet-centred mission phases. The spacecraft (sail payload) is mounted centrally, within the plane of the solar sail, so that both faces of the core structure are free to be used as attachment points for the lander, and Earth return capsule. Fig. 3 shows approximate trip times from Earth to Mercury, generated using methods described in the Trajectory Analysis section. An outbound trip time of 2-3 years is desirable to be competitive with SEP and Chemical Mercury trip times. This is enabled by a characteristic acceleration of 0.25 mm s^{-2}. The chosen sail conceptual design used in this paper is based on the AEC-ABLE Scaleable Solar Sail Subsystem (S^4), since it can be extrapolated to large sail dimensions.[6] This design is based on Coilable booms, and the boom linear density as a function of length can be combined with NASA/LaRC/SRS 2 μm or 5 μm CP1 film to obtain the sail assembly loading as a function of sail side length, shown in Fig. 4. It is assumed that conventional coatings are used, with Aluminium (85% reflectivity) on the frontside and Chromium (64% emissivity) on the backside. Fig. 4 also shows the necessary sail assembly loading as a function of sail side length, for delivery of a 1905 kg spacecraft to Mercury with a characteristic acceleration of 0.25 mm s^{-2}.

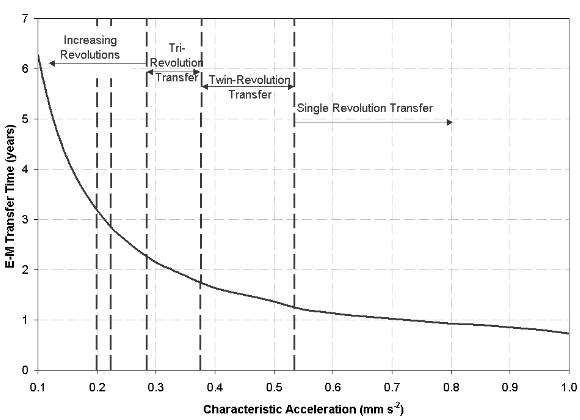


Figure 3: Approximate Earth-Mercury transfer time

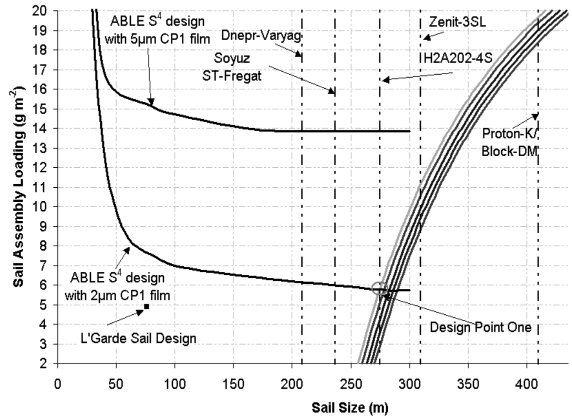


Figure 4: 0.25 mm s^{-2} solar sail design space (sail payload contours represent increasing parking orbit radius, with baseline 100 km orbit leftmost)

It can be seen that the intersection of the 2 μm CP1 ABLE S^4 sail design curve with the 0.25 mm s^{-2}, 100 km orbit payload curve yields the sail design point, with an assembly loading of 5.9 g m^{-2} and sail dimensions of 275 x 275 m. The design point and resultant characteristic accelerations during different points in the mission, as the lander is deployed and sample is returned, are shown in Table 5. It is important to stress that for a specific solar sail, the acceleration will increase as the solar sail payload mass is reduced, through the jettison of used modules.

Parameter	**Value**
Sail Assembly loading (@ 40% contingency)	5.9 g m^{-2}
Sail side length	275 m
Sail area (@ 2 μm thickness)	75625 m^2
Boom length	194 m
Sail reflective efficiency	0.85
Characteristic Acceleration (Earth departure)	0.25 mm s^{-2}
Characteristic Acceleration (Sample acquisition)	0.7367 mm s^{-2}
Characteristic Acceleration (Mercury departure)	0.7839 mm s^{-2}

Table 5: Solar sail specifications and resultant characteristic acceleration during each phase

A 275 m sail with an assembly loading of 5.9 g m^{-2} has a mass of 448 kg, with a mass budget as shown in Table 6. A linear boom density of 70 g m^{-1} is required with 0.94 m diameter to maintain a factor of safety against buckling. The total launch mass is therefore 2353 kg, which enables the use of an H2A202-4S class launch vehicle to escape velocity. The spacecraft stack with stowed sail is depicted within the H2A fairing in Fig. 5.

Component	Mass (kg)
Total payload mass	**1905**
2μm CP1 film (@ 2.86 g m^{-2})	216
0.1μm Al coating (@ 0.54 g m^{-2})	41
Bonding (@ 10% coated mass)	26
Sail booms (ABLE 0.94m booms @ 70 g m^{-1})	54
Mechanical systems (@ 40% contingency)	111
Total sail assembly mass	**448**
Total mission launch mass	**2353**
H2A202-4S capacity to C_3 = 0	**2600**
Launch mass margin	247 kg (9.5 %)

Table 6: Solar sail design point data set

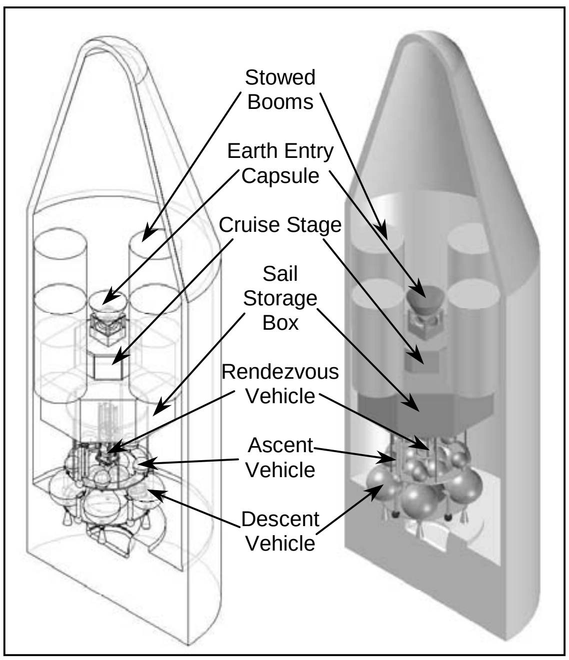


Figure 5: Payload stack in H2A 202-4S fairing

COST ANALYSIS

The spacecraft has been costed using parametric Cost Estimating Relationships (CERs).[7] Conservative margins have been added, and the cost of specialist components, such as bi-propellant engines, have been taken from NASA/JPL Team X estimates.[2] Project management and integration and support costs are also estimated using Ref. 7. The most difficult system to cost is that of the solar sail, since a sail is yet to fly, let alone one of 275 m dimension. A crude estimate leads to a ROM cost of 28.4 M€, but it should be noted the cost of the sail is small in comparison with the spacecraft itself. In addition, the reduction in launch cost compared with conventional concepts more than makes up for sail cost.

Conservative cost margins of 30% have been added to give the mission cost breakdown shown in Table 7. The total solar sail Mercury sample return mission ROM cost is therefore of order 850 M€. We note that, although the launch cost is fairly low, the predominant cost component is the spacecraft itself, which is mostly independent of the primary propulsion method used. Traditionally, solar sailing is seen to be superior to chemical propulsion or SEP, if it can reduce launch mass and cost, but for a sample return mission, the sail must *significantly* reduce launch mass, for there to be any appreciable reduction in overall mission cost.

Component	Cost (FY03M€)	Margin (%)	Total Cost (FY03M€)
SRV	27.8	30	36.1
MAV	58.8	30	76.4
MDV	88.3	30	114.8
SCS	89.1	30	115.8
SOLAR SAIL	28.4	30	36.9
EEV	4.2	30	5.5
Spacecraft Cost	**296.6**	**-**	**385.5**
IA&T	94.9	30	123.4
Program Level	156.3	30	203.2
GSE	19.6	30	25.5
LOOS	18.1	30	23.5
Launch Cost (H2A)	83.9	10	92.3
Associated Costs	**372.8**	**-**	**467.9**
Total Mission Costs	669.4	-	**853.4**

Table 7: Cost breakdown

TRAJECTORY ANALYSIS

The required Δv for direct ballistic transfer to a low Mercury parking orbit is of order 13 km s^{-1}. Chemical propulsion and Solar Electric Propulsion (SEP) both require a prolonged sequence of gravity assists to reduce launch mass. Mercury sample return from deep within the solar gravity well is one of the most energetically demanding mission concepts imaginable. However, propellant mass is not an issue here and the sail can spiral directly to the planet, making best use of the inverse square increase in Solar Radiation Pressure (SRP) at lower heliocentric radii. Many authors have recognised the benefit of solar sailing to reach Mercury, but this paper provides new data sets by considering both launch windows, and return trajectories.

Heliocentric trajectories have been optimised using the constrained parameter optimisation algorithm, NPSOL, based on Sequential Quadratic Programming (SQP).[8,9] Engineering insight coupled with 'incremental feedback' methods were used to obtain initial guesses for optimisation. Planet centred manoeuvres are modelled using a set of blended analytical control laws.[10] Mercury capture and escape trajectories have been generated mainly using a control law which maximises the rate of change of orbit energy. Many control laws are blended for Mercury-centred transfer manoeuvres.

Launch windows

Fig. 6 shows the Earth departure date scan for the selected characteristic acceleration of 0.25 mm s^{-2}, over a 3 year period. Each point on the curve represents an optimisation at that launch date. It is seen that the minimum time launch opportunities occur once every year. Solar sailing is not restricted to launch windows, but it is clear that a saving of 300 days can be achieved depending on launch date. The discontinuities posed problems when incrementing the launch date to find initial guesses for other launch dates. These discontinuities are due to the spacecraft 'just missing' the target and having to execute another revolution of the Sun to reach Mercury. To determine the optimal launch date, consideration must also be given to the variation of the capture and escape times along Mercury's orbit, and the return Mercury-Earth phase. Since Mercury has an eccentricity of 0.2056, then the available SRP will vary over a Mercury year.[11] Approximate capture and escape times are shown in Fig. 7, for the accelerations specified in Table 5.

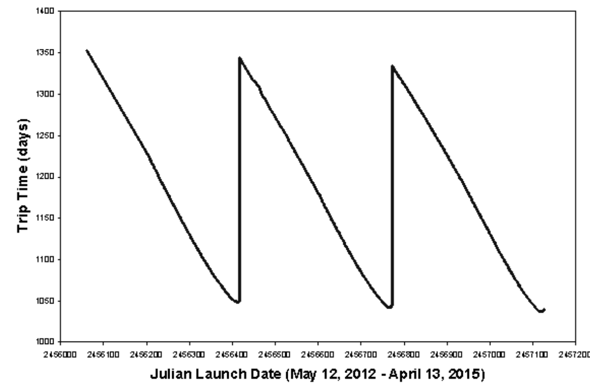


Figure 6: Earth-Mercury departure date scan

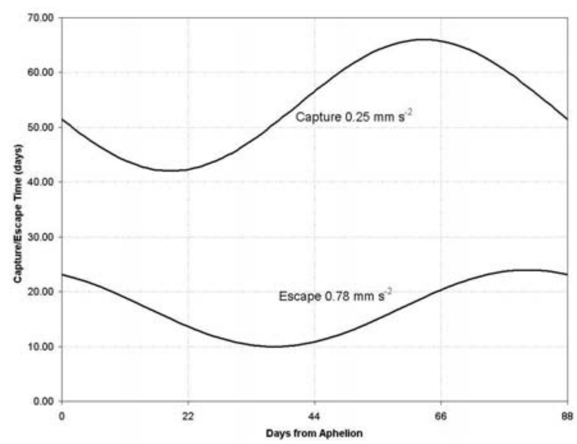


Figure 7: Mercury capture/escape time variation

With an orbiter stay time of order 40 days, Figs. 6 and 7 can be used to ascertain that the return scan was only required across a 2 year range (small variation). The 4 curves were then mapped together to determine the overall mission duration as a function of Earth departure date. This is shown in Fig. 8, where it is clear that the long duration outbound spiral dominates the total mission duration. The launch opportunity selected was that on April 19, 2014.

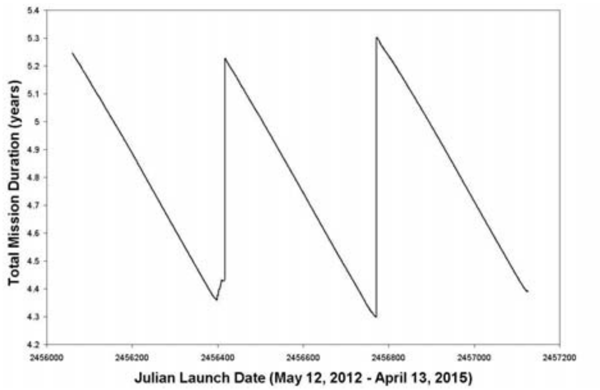


Figure 8: Total mission duration launch opportunities

Earth-Mercury Phase

The outbound trajectory is shown in Fig. 9, departing Earth with C_3 of zero on April 19, 2014. Mercury arrival is on February 24, 2017, 2.85 years later, after 5 ¼ revolutions. The optimal cone and clock control angles are shown in Fig. 10. Even at a relatively coarse control resolution of 50 linear interpolation segments, the profiles are smooth and oscillatory.

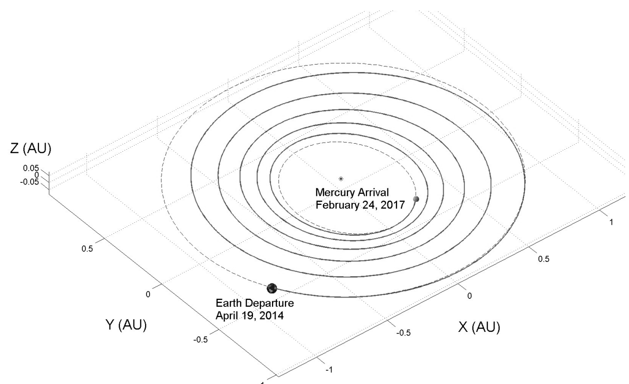


Figure 9: Earth-Mercury trajectory

The reduction in heliocentric radius and subsequent increase in sail film temperature is depicted in Fig. 11. Equilibrium sail film temperature is modelled using a black body approximation, assuming temperature changes take place instantaneously, since the micron-scale thickness of the film ensures that the thermal inertia is effectively zero. Aluminium/Chromium coatings are assumed as was discussed previously. The temperature is a function of both the radius and the sail attitude, with a maximum value of 443.7 K. Even face on to the Sun at Mercury perihelion, the worst-case temperature would be 494.5 K, still less than the predicted 520 K upper limit of polyimide films.

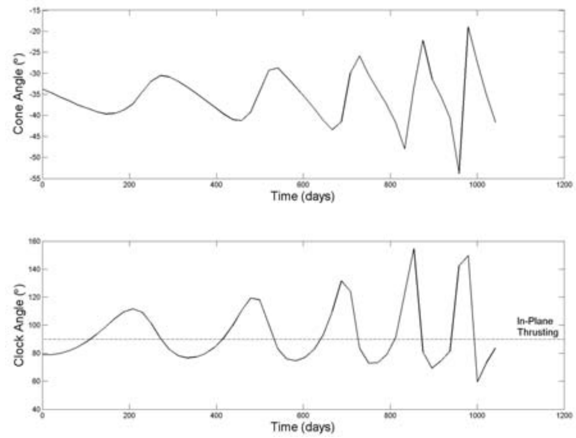


Figure 10: Earth-Mercury control angle profile

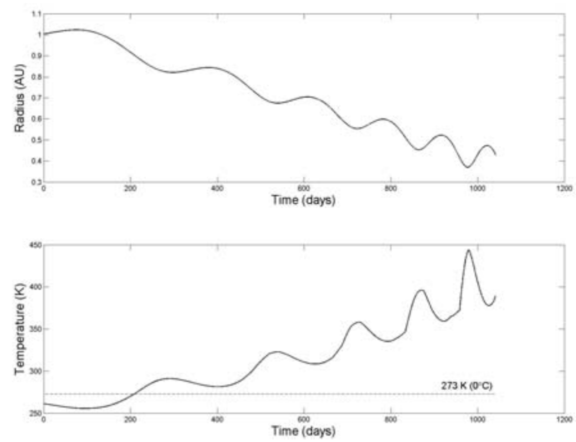


Figure 11: Earth-Mercury heliocentric radius and sail film temperature

Mercury Centred Manoeuvres

It has been assumed that the sail arrives at Mercury with zero hyperbolic excess velocity. The transition from heliocentric to planet-centred motion has not been patched. However, it is assumed that the sail can be used to correct for approach dispersion and can target the correct B-plane for capture. As has been prescribed, capture is into a 100 km x 7500 km Sun-Synchronous polar orbit, 10° ahead of the terminator, before subsequent manoeuvring into the 100 km parking orbit. This capture spiral takes 28 days and is shown in Fig. 12, arriving on orbit on March 24, 2017.

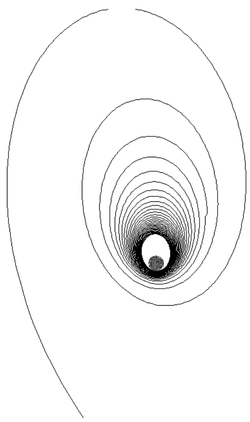


Figure 12: Mercury capture spiral into 100 km x 7500 km Sun-Synchronous polar orbit

131 days will be available for orbital science operations, surface observation and final manoeuvring to the lander descent orbit. This orbiter stay-time is also a requirement due to the thermal environment on the surface. The thermally-benign, Sun-Synchronous orbit (10° ahead of terminator) is forced for 44 days until the orbit is in the correct orientation for the landing site. The sail then waits in this orbit for 37 days. Next, a 50 day manoeuvre transfers the spacecraft to the 100 km polar orbit, where the lander begins its descent on August 3, 2017. Once on the surface, the lander carries out 4 days of sample acquisition and landing site documentation operations. The solar sail is used to rotate the orbit plane to account for Mercury landing site rotation, so that the MAV ascends in a coplanar manoeuvre. The orbit plane cannot be rotated as fast as Mercury spins, so the MAV will need to wait in the 100 km orbit (thermally-safe) until solar sail rendezvous with the MAV. Final proximity manoeuvring is accomplished with the SRV, thereby relaxing MAV launch accuracy. Rotation of the orbit plane to match that of the landing site is depicted in Fig. 13. After sample transfer to the Earth Return Vehicle attached to the sail, the solar sail spirals to escape. A method which maximises the rate of change of orbit energy while maintaining a positive altitude of periapsis is illustrated in Fig. 14. The escape spiral is initiated on August 18, 2017, with escape reached in 16 days.

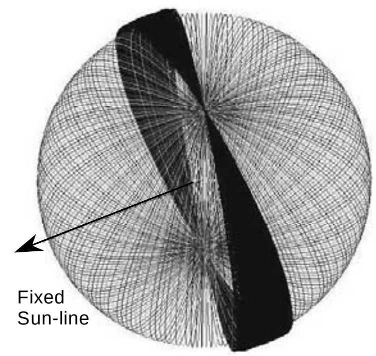


Figure 13: Rotation of 100 km polar orbit plane to match coplanar MAV ascent trajectory

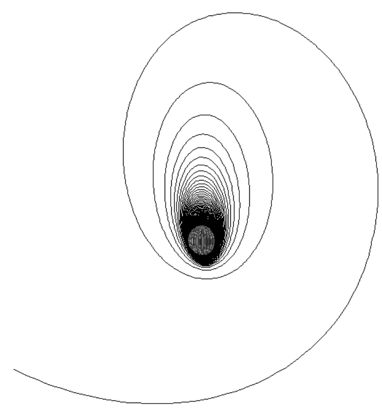


Figure 14: Mercury escape spiral from 100 km circular polar orbit

<u>Mercury-Earth Phase</u>

Return heliocentric spiralling commences after Mercury escape on September 3, 2017. The trip time is 369 days, with arrival back at the Earth with zero hyperbolic excess on September 8, 2018. Fig. 15 shows the 2 revolution trajectory, which is faster because the sail characteristic acceleration has increased to 0.78 mm s^{-2}. The cone and clock angle control profile is shown in Fig. 16. Finally, the ERV spins up and is separated to perform a ballistic entry for sample delivery to Earth. The total mission duration is 4.39 years.

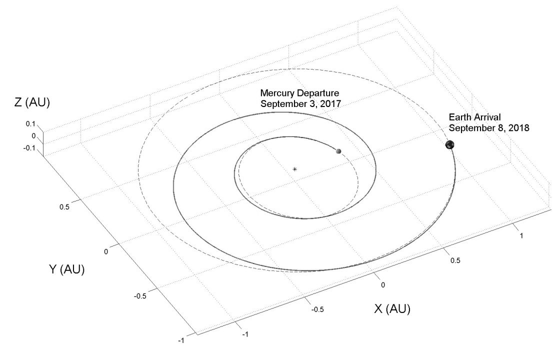


Figure 15: Mercury-Earth trajectory

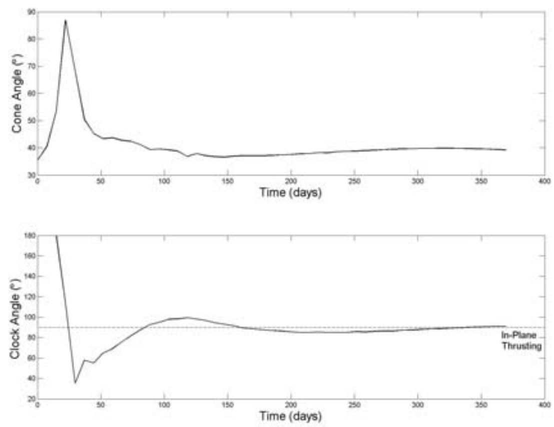


Figure 16: Mercury-Earth control angle profile

Alternative trajectory Options

Use of a positive launch C_3 against the Earth's velocity would be highly advantageous for reaching close solar orbits such as that of Mercury. The initial eccentricity for the inward spiral can be easily circularised by the increased solar radiation pressure closer to the Sun. Fig. 17, shows the effect of using excess launch energy to reduce the trip time to Mercury orbit. It can be seen that the effect is greater for lower accelerations, since the trip time is longer and there are more revolutions for C_3=0. The use of a Zenit 3-SL over an H2A, would allow for a C_3 = 8 km^2 s^{-2}, which would reduce the outbound trip time by 260 days, for the same launcher cost.

Fig. 18 shows that the inclusion of a Venus gravity assist could reduce the outbound trip time by 140 days (see Ref. 8), but gravity assists are not essential for solar sails since propellant mass is not an issue.

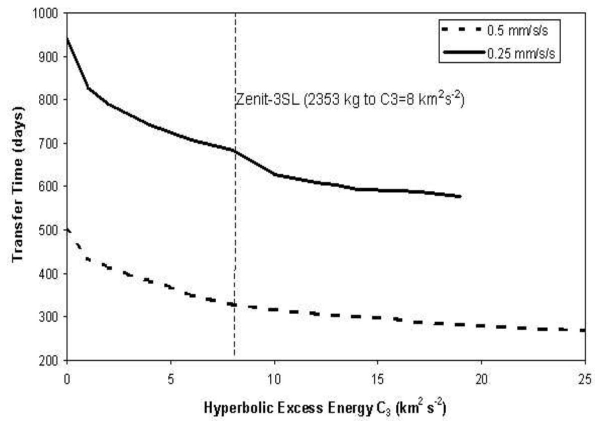


Figure 17: Effect of hyperbolic excess energy at launch

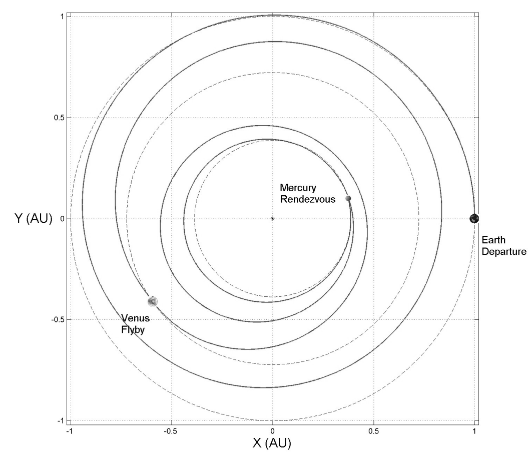


Figure 18: Venus gravity assist

MISSION EVALUATION AND CONCLUSION

Other possible mission architectures were considered in the course of this work.[4] In addition to the baseline all-sail concept, the use of the sail to spiral to Earth escape to reduce launch energy requirements was considered, a multi-mission concept, and a chemical/sail hybrid mission was briefly investigated. A chemical outbound ballistic transfer to Venus, with a small solar sail deployed for return, is attractive.[12] However, the outbound gravity assisted trajectory to Mercury would dominate the mission duration of almost 9 years, even though a smaller, cheaper solar sail could be used for the return leg. An Ariane 5 launch would be required in this case.

To summarise the Solar Sail MeSR concept, a 275 m side square solar sail is used to transport a 1905 kg payload to 100 km polar orbit at Mercury, and return a sample to Earth in 4.4 years. The 448 kg, 5.9 g m^{-2} solar sail uses AEC-ABLE booms and 2 μm CP1 film, with conventional coatings. The launch mass of 2353 kg is lifted using an H2A202-4S (C_3=0, or Zenit-3 SL to C_3=8). The total mission ROM cost is estimated to be 850 M€.

The mission concept has been compared with other propulsion options.[1,2] The 5775 kg launch mass of the NASA/JPL Team X SEP concept requires an Atlas V 551 launcher, for a 6.9 year mission, costing of order 1034 M€.[2] An ESA Chemical/SEP concept has a 6500 kg launch mass on an Ariane 5E, for a mission duration of 7.2 years.[1] No ROM cost is given for this, but it is expected to be in the same order of the NASA cost. Therefore, it is clear that a solar sail MeSR mission can reduce the total mission duration by 40%, and reduce launch mass by 60%, with a reduction in ROM cost of at least 180 M€.

Finally, this analysis assumes the feasibility of large sail structures, their deployment, and attitude control using tip-vanes. There is limited experience of large gossamer structures at present. Therefore, it is imperative that near-term demonstration missions take place, and a rigorous technology development programme is pursued, before a solar sail mission to Mercury can be realised.

ACKNOWLEDGEMENT

This work was performed under ESA/ESTEC contract 16534/02/NL/NR – Technical Assistance in the Study of Science Payloads Transported through Solar Sailing.

REFERENCES

1. Scoon, G., Lebreton, J-P., Coradini, M., et al, “Mercury Sample Return”, Assessment Study Report, ESA Publication SCI(99)1, 1999.
2. Oberto, B., *et al*, “Mercury Sample Return 2002-01,” Team X Advanced Projects Design Team Report, Jet Propulsion Laboratory, Pasadena, California, 2002.
3. McInnes, C. R., *Solar Sailing: Technology, Dynamics and Mission Applications*, Springer-Praxis Series in Space Science and Technology, Springer-Verlag, Berlin, 1999.
4. McInnes, C. R., Hughes, G. W., and Macdonald, M., “Technical Note 3 – Mercury Sample Return,” ESTEC 16534/02/NL/NR, ESA/ESTEC Contract Report, University of Glasgow, 2003.
5. Leipold, M.E., Wagner, O., “Mercury Sun-Synchronous Polar Orbits Using Solar Sail Propulsion”, J. Guidance, Control and Dynamics, Vol. 19, No. 6, pp 1337-1341, 1996.
6. Murphy, D.M., and Murphey, T.W., “Scalable Solar Sail Subsystem Design Considerations,” AIAA 2002-1703, 43rd Structures, Structural Dynamics and Materials Conference, Denver, Colorado, Apr. 22-25, 2002.
7. Wertz, J. R., and Larson, W. J., *Space Mission Analysis and Design*, Kluwer Academic Publishers, pp 795-802, ISBN 0-7923-5901-1, 1999.
8. Hughes, G. W., and McInnes, C. R., “Mercury Sample Return Missions Using Solar Sail Propulsion,” IAC-02-W-2.08, 53rd International Astronautical Congress, Houston, Texas, Oct. 10-19, 2002.
9. Hughes, G. W., and McInnes, C. R., “Small-Body Encounters Using Solar Sail Propulsion,” *Journal of Spacecraft and Rockets*, Vol. 41, No.1, pp. 140-150, Jan.-Feb., 2004.
10. Macdonald M., and McInnes C. R., “Analytic Control Laws for Near-Optimal Geocentric Solar Sail Transfers,” AAS 01-472, Advances in the Astronautical Sciences, Vol. 109, No. 3, pp. 2393-2413, 2001.
11. Macdonald M., and McInnes, C. R., “Seasonal Efficiencies of Solar Sailing in Planetary Orbit,” IAC-02-S.6.01, 53rd International Astronautical Congress, Houston, Texas, Oct. 10-19, 2002.
12. Hughes, G. W., Macdonald, M., McInnes, C. R., Atzei, A., and Falkner, P., “Terrestrial Planet Sample Return Missions Using Solar Sail Propulsion,” 5th IAA International Conference on Low-Cost Planetary Missions, ESA/ESTEC, The Netherlands, Sept. 24-26, 2003.

2nd international Planetary Probe Workshop

NASA Ames Conference Centre, USA,

23-26 August 2004

SCI-AP

European Space Agency
Agence spatiale européenne

Science Payload and Advanced Concepts Office (SCI-A)
Planetary Exploration Studies Section (SCI-AP)
2200 AG Noordwijk, Postbus 299
The Netherlands

ESA VENUS ENTRY PROBE STUDY

M.L. van den Berg [(1)], P. Falkner[(1)], A. Phipps[(2)], J.C. Underwood[(3)], J.S. Lingard[(3)], J. Moorhouse[(4)], S. Kraft[(4)], A. Peacock[(1)]

[(1)] Science Payload & Advanced Concepts Office, European Space Agency, ESTEC, P.O. Box 299, 2200 AG Noordwijk, The Netherlands, E-mail: MvdBerg@rssd.esa.int
[(2)] Surrey Satellite Technology Ltd., Guildford, Surrey GU2 7XH, United Kingdom
[(3)] Vorticity Ltd., Chalgrove, Oxfordshire OX44 7RW, United Kingdom
[(4)] Cosine Research B.V., 2333 CA Leiden, The Netherlands

ABSTRACT

The Venus Entry Probe is one of ESA's Technology Reference Studies (TRS). The purpose of the Technology Reference Studies is to provide a focus for the development of strategically important technologies that are of likely relevance for future scientific missions. The aim of the Venus Entry Probe TRS is to study approaches for low cost in-situ exploration of Venus and other planetary bodies with a significant atmosphere. In this paper, the mission objectives and an outline of the mission concept of the Venus Entry Probe TRS are presented.

1. INTRODUCTION

The Venus Entry Probe is an ESA Technology Reference Study (TRS) [1]. Technology reference studies are model science-driven mission studies that are, although not part of the ESA science programme, able to provide a focus for future technology requirements. This is accomplished through the study of several technologically demanding and scientifically meaningful mission concepts, which have been strategically chosen to address diverse technological issues.

Key technological objectives for future planetary exploration include the use of small orbiters and in-situ probes with highly miniaturized and highly integrated payload suites. The low resource, and therefore low cost, spacecraft allow for a phased strategic approach to planetary exploration, thus reducing mission risks compared to a single heavy resource mission.

2. VENUS EXPLORATION IN CONTEXT

More than twenty missions have been flown to Venus so far, including fly-bys, orbiters, and in-situ probes. These past missions have provided a basic description of the planet, its atmosphere and ionosphere as well as a complete mapping of the surface by radar. The upcoming comprehensive planetary orbiters, ESA's Venus Express (launch 2005)[2] and Planet-C from ISAS (launch 2007)[3], will further enrich our knowledge of the planet. These satellite observatories will perform an extensive survey of the atmosphere and the plasma environment, thus practically completing the global exploration of Venus from orbit. For the next phase, detailed in-situ exploration will be required, expanding upon the very successful Venera atmospheric and landing probes (1967 - 1981), the Pioneer Venus 2 probes (1978), and the VEGA balloons (1985).

3. MISSION OBJECTIVES

The objective of the Venus Entry Probe Technology Reference Study is to establish a feasible mission profile for a low-cost in-situ exploration of the atmosphere of Venus. An extensive literature survey has been performed in order to identify a typical set of scientific objectives for such a mission. From this survey, the following set of key issues has been derived (with references to review articles):

[SR1] *Origin and evolution of the atmosphere*

A major question is to understand why and how the atmosphere has evolved so differently compared to Earth. This can only be investigated by in-situ measurements of the isotopic ratios of the noble gases [4, 5].

[SR2] *Composition and chemistry of the lower atmosphere*

Accurate measurements of minor atmospheric constituents, particularly water vapour, sulphur dioxide and other sulphur compounds, will improve our knowledge of the runaway greenhouse effect on Venus, atmospheric chemical processes and atmosphere-surface chemistry, and will address the issue of the possible existence of volcanism [4, 5].

[SR3] *Atmospheric dynamics*

Venus has a very complicated atmospheric dynamical system. The driving force behind the zonal supperrotation, the dynamics of the polar vortices and the meridional circulation as well as the cause of temporal and spatial variations of the cloud layer opacity are all rather poorly understood [4, 5, 6].

[SR4] *Aerosols in the cloud layers*

Measurements of the size distribution, temporal and spatial variability as well as the chemical composition of the cloud particles is of interest for better understanding the thermal balance as well as the atmospheric chemistry [4]. Furthermore, it has been suggested that the unidentified large (~ 7 µm diameter) cloud particles might contain microbial life [7, 8].

[SR5] *Geology and tectonics*

Key outstanding questions on the surface of Venus are the mineralogy, the history of resurfacing as well as of volcanism [5]. Resolving the global tectonic structure and (improved) topographical mapping will improve our understanding on these issues.

4. MISSION DESIGN

4.1 Mission requirements

In order to address the science objectives, the following mission requirements have been imposed on the Venus Entry Probe TRS:

[MR1] In-situ scientific exploration at an altitude between 40 and 57 km at all longitudes by means of an aerobot [SR1-4].

[MR2] Vertical profiles of a few physical properties of the lower atmosphere at varying locations across the planet by means of atmospheric microprobes [SR3].

[MR3] Remote atmospheric sensing to provide a regional and global context of the in-situ atmospheric measurements (also concurrent with the aerobot operational phase) [SR2-4].

[MR4] Remote sensing of the polar vortices with a large field of view and a repeat frequency less than 5 hours [SR3].

[MR5] Remote sensing of the Venus atmosphere at all longitudes and latitudes [SR2-4].

[MR6] Remote sensing of the Venus surface by means of a ground penetrating radar and radar altimeter [SR5].

4.2 Mission concept

The mission concept that is able to fulfil all requirements consists of a pair of small-sats and an aerobot, which drops active ballast probes. A two-satellite configuration is required in order to commence

Table 1. Mission baseline scenario.

S/C Module	Measurements	Strawman payload	Requirements
Venus Polar Orbiter (VPO)	- Atmospheric composition - Atmospheric dynamics - Atmospheric structure	- Microwave sounder - Visible-NIR imaging spectrometer - UV spectrometer - IR radiometer	- Large FOV - Resolution ~ 5 km - Operational before aerobot deployment - Aerobot communications
Venus Elliptical Orbiter (VEO)	- Subsurface sounding - Topographical mapping	- Ground penetrating radar - Radar altimeter - Entry probe	- Low periapse (radar) - Entry probe deployment - Data relay to Earth
Aerobot	- Isotopic ratios noble gases - Minor gas constituents - Aerosol analysis - Pressure, temperature etc. - Tracking and localization of microprobes	- Gas chromatograph /Mass spectrometer with aerosol inlet - Nephelometer - IR radiometer - Meteorological package - Radar altimeter	- Long duration (different longitudes) - Microprobe deployment - Altitude 40 - 57 km (aerosols)
Atmospheric microprobes	- Pressure, temperature - Light level (up and down) - Wind velocity	- P/L fully integrated with probe	- Operational down to 10 km or less

the remote sensing atmospheric investigations prior to the aerobot deployment (MR3).

One satellite will be in a polar Venus orbit. The Venus Polar Orbiter (VPO) contains a remote sensing payload suite primarily dedicated to support the in-situ atmospheric measurements by the aerobot and to address the global atmospheric science objectives. The second satellite enters a highly elliptical orbit, deploys the aerobot and subsequently operates as a data relay satellite, while it also performs limited science investigations of the ionosphere and the surface (after lowering the apoapse).

The aerobot consists of a long-duration balloon, which will analyse the scientifically interesting Venusian middle cloud layer. During flight, the balloon deploys a swarm of active ballast probes, which determine vertical profiles of pressure, temperature, flux levels and wind velocity in the lower atmosphere.

The concept of a long-duration balloon with ballast probes is not new and has been proposed before, see e.g. [9, 10, 11]. The focus of the Venus Entry Probe study is to identify the critical technologies associated with such a concept with the aim to successfully support the technology development of a miniaturized aerobot system with atmospheric microprobes.

Table 1 gives an overview of the mission baseline scenario, including a strawman payload suite. Because atmospheric science investigations (large field of view and high polar revisit frequency, see MR4) and surface radar investigations (low periapse) pose different requirements on the operational orbit, the Venus Polar Orbiter will carry the atmospheric remote sensing instrumentation and the Venus Elliptical Orbiter the radar instrumentation. The tentative operational orbits for both spacecraft are listed in Table 2.

4.3 Launch and transfer to Venus

A Soyuz-Fregat 2-1B launch from Kourou has been selected as the baseline for the Venus Entry Probe TRS because it is a cost-efficient and highly reliable launch vehicle. The mass capability for direct escape to Venus is about 1400 kg. The Earth departure phase can be optimized by launching the Soyuz-Fregat into a highly elliptical Earth orbit, with the spacecraft providing the delta-V for Earth escape [12].

A standard high thrust heliocentric transfer from Earth to Venus is envisaged, because this is the most cost-efficient and flexible option for a mission to Venus. The launch opportunities are primarily driven by the Earth-Venus synodic period of 1.6 years. The 3.4º inclination of Venus' orbit to ecliptic causes a variation in the Earth-Venus distance, so that the delta-V requirements vary at successive optimum launch windows.

The typical transfer time for a half solar revolution transfer is between 120 and 160 days, with a delta-V requirement for Venus orbit insertion (250 km × 66,000 km) typically less than 1.4 km/s [12]. Depending on Earth departure strategy and planetary geometry, a high-thrust chemical propulsion system can typically bring into Venus orbit a spacecraft mass between 900 kg and 1150 kg.

The VPO and VEO spacecraft can travel as a composite or individually. As the composite configuration is more mass and cost-efficient (mission operations), this is currently selected as the baseline, with the VEO providing the propellant for departure and Venus orbit insertion.

4.4 Venus Polar Orbiter spacecraft

The 3-axis stabilized Venus Polar Orbiter spacecraft is based on a thrust tube structural concept, because of its low mass and simplicity of design. The propulsion system consists of a conventional dual mode bipropellant system, using Hydrazine and Nitrogen Tetroxide for high thrust manoeuvres and Hydrazine monopropellant thrusters for low thrust.

Table 2. Operational orbits for the Venus Polar Orbiter and the Venus Elliptical Orbiter spacecraft.

	VPO	VEO
Periapse (km)	2000	250
Apoapse (km)	6000	7500 – 20000
Period (hr)	3.1	3.1 – 6.2
Inclination	~ 90˚	~ 75 - 90˚

Table 3. Venus Polar Orbiter mass budget.

Item	Mass(kg)
Science instruments	30
Communications	22
Structure	51
Propulsion	63
ACS	10
OBDH	4
Power	21
Thermal control	14
Subtotal	**215**
System margin (20%)	43
Total dry mass	**258**

Table 3 shows the top level mass budget for the Venus Polar Orbiter. A mass budget of 30 kg (including margins) has been allocated for the remote sensing atmospheric science instruments (see Table 1). The payload instruments will be integrated into a highly integrated payload suite. By merging individual instruments onto one platform and sharing resources on a system architecture level, considerable mass and power reductions can be achieved without sacrificing the scientific performance. The science data obtained by the Venus Polar Orbiter will be relayed to the Venus Elliptical Orbiter through an X-band link.

4.5 Venus Elliptical Orbiter spacecraft

Table 4 lists the top-level mass budget for the Venus Elliptical Orbiter. For cost reduction purposes, the commonality of platform and subsystems between the VPO and VEO will be exploited as much as possible. As a consequence, the VEO spacecraft also uses a similar thrust tube concept and a dual mode propulsion system.

The Venus Elliptical Orbiter will stay in a highly elliptical orbit until deployment of the entry probe, which is initiated after the VPO has reached its final orbit and the instrument calibration phase has been completed. During this first phase, the VEO primarily acts as a relay station to Earth for data from the VPO as well as from the aerobot, possibly via the VPO. The Ka-band has been selected for communications to Earth, whereas X-band communication is the baseline for the inter-satellite communications.

After the operational phase of the aerobot has ended, the VEO will progress to its final low elliptical orbit (250 km × 7,500 – 20,000 km) in order to start the detailed (sub)surface radar investigations. The current preliminary mass budgets allow for an apoapse of 20,000 km using chemical propulsion. Further work is in progress to assess whether aerobraking or spacecraft mass reduction are viable routes towards a lower apoapse, and consequently a larger surface coverage. A mass of 20 kg has been reserved for a ground penetrating radar and a radar altimeter. Possibly a wide field camera will be included as well.

Table 4. Venus Elliptical Orbiter mass budget.

Item	**Mass (kg)**
Science instruments	20
Entry Probe	85
Communications	32
Structure	65
Propulsion	135
ACS	10
OBDH	4
Power	19
Thermal control	5
Subtotal	**375**
System margin (20%)	75
Total dry mass	**450**

4.6 Entry vehicle

Fig. 1 shows a conceptual drawing of the entry vehicle. The aeroshell has a 45º sphere-cone geometry, which provides a good packaging shape and aerodynamical stability. Most of the volume of the entry probe is taken up by the spherical gas storage tank, which is surrounded by the ring-shaped gondola. For storage of the balloon inflation gas, a conventional gas tank has been baselined, though alternatives such as cold gas generators or chemical storage of hydrogen are being considered.

In Table 5 the tentative top-level mass budget for the Venus entry vehicle is summarized. Because the design study is still in an early phase, a 25% design maturity margin has been added.

4.6.1 Probe release

The entry probe will be released from the VEO spacecraft, while it is in a highly elliptical orbit with an orbital period in excess of 24 hours. To keep the entry vehicle design simple, the VEO spacecraft will provide the required velocity and orientation for the probe entry. After release of the probe, the spacecraft will perform a re-orbit burn.

Deployment from orbit has been chosen as the baseline because direct entry from the interplanetary transfer hyperbola would require a complicated interplanetary transfer trajectory or orbit insertion scenario in order to fulfil the requirement of starting the remote sensing atmospheric science investigations with the VPO prior to aerobot deployment (MR3).

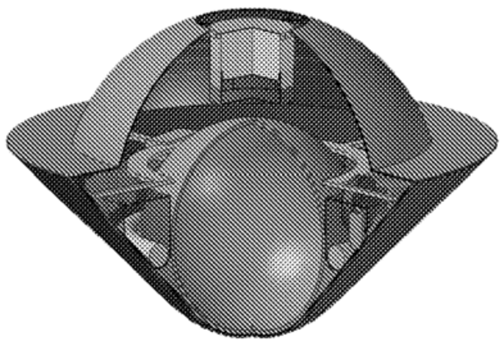


Fig. 1. Conceptual drawing of the entry probe.

4.6.2 Entry, descent and deployment

The probe will enter the dense Venus atmosphere with a velocity of 9.8 km/s and a flight path angle between 30˚ and 40˚, as this scenario yields the best overall system mass. A steep entry angle will cause the probe to penetrate deep within the atmosphere quickly, leading to high accelerations and heat fluxes. However, since the deceleration to subsonic velocities occurs very quickly, the total absorbed heat is relatively low. Additionally, the short entry duration enables a quick release of the aeroshell (~20 seconds), thus minimizing the time for the absorbed heat to soak through the heat shield.

The heat shield material consists of Carbon-Phenolic, which is capable of withstanding very high heat fluxes (~ 300 MW/m^2), much higher than the peak heat flux of ~20 MW/m^2 for a 40˚ entry angle. The maximum entry flight path angle is set by the 200 g acceleration capability of the payload.

The deployment sequence is depicted in figure 2. The 45˚ sphere-cone is designed to be stable in the hypersonic and supersonic regimes, so that no active control is required. Just above Mach 1.5, a disk-gap-band or a ribbon parachute will be deployed by a pyrotechnic mortar. The parachute stabilizes the probe as it decelerates through the transonic regime. The front aeroshell will be released a few seconds after parachute deployment when the subsonic regime has been reached. To prevent heating from the back cover, the rear aeroshell will be distanced from the aerobot by a tether. At a velocity of ~20 m/s and altitude of ~55 km, the balloon will be deployed. The parachute and rear aeroshell are released and the inflation of the balloon is started. The parachute will be designed with a small amount of glide to ensure lateral separation between the parachute and the balloon. The inflation time of the balloon is a trade between the minimum altitude and the aerodynamic loads on the balloon. Currently, an inflation duration of 20 seconds and a minimum altitude of 54 km is foreseen. The gas storage system will be released after inflation of the balloon, and the aerobot will gradually rise to cruise altitude.

Table 5. Venus entry vehicle mass budget.

Item	Mass (kg)
Gondola in-situ science instruments	4.0
Atmospheric microprobe system	4.0
Aerobot-VEO communications	1.6
Gondola structure and separation system	6.9
Gondola OBDH	0.6
Gondola power	5.6
Gondola environment	0.3
Subtotal (Gondola)	**23.0**
Balloon (including gas, envelope and deployment system)	5.4
Gas storage system	14.9
Entry and descent system	24.8
Total mass entry vehicle	**68.1**
Design maturity margin (25%)	17.0
Mass Entry Vehicle (with margin)	**85.1**

4.6.3 Aerobot

The balloon will stabilize at an altitude of 55 km. At this altitude all the scientific issues outlined in section 3 can be addressed, while the environment is relatively benign (30 °C and 0.5 bars [13]).

The goal for the aerobot operational mission duration is to travel at least twice around Venus. Taking the average speed of 67.5 m/s from the VEGA balloons that flew at a similar altitude [14], one obtains a minimum flight duration of 14 days.

A light gas balloon with slight overpressure is considered the most suitable candidate for the Venus aerobot, because such a balloon complies best with the operational requirements for a long duration mission. As the gas leaks out of the super pressure balloon, the float altitude will increase until there is insufficient gas for positive buoyancy (and the balloon sinks to the surface). A carefully selected microprobe drop scenario could partially compensate for the loss of balloon gas and thus maximize the operational lifetime. Gas release mechanisms and gas replenishment systems are also being considered in order to compensate for

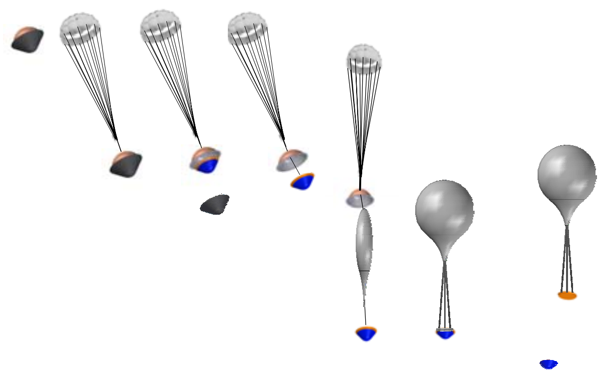


Fig. 2. A schematic of the Venus aerobot deployment.

temperature changes in the balloon gas due to gradients in solar radiation at the day/night and night/day terminators.

Hydrogen has been selected as the baseline for the balloon inflation gas, with helium as a backup option. Though the mass of gas storage systems for hydrogen and helium are similar, the main advantage of hydrogen is that it generally has a lower gas leakage rate compared to helium, which is a monatomic gas. The main disadvantage of using hydrogen is its hazardousness.

The balloon envelope material should have an extremely low leakage rate, possibly requiring welded seams. Additionally, the deployment will have to be carried out in a controlled manner to avoid the slightest damage to the envelope.

4.6.4 Gondola

Figure 3 shows a conceptual drawing of the gondola layout. A strawman payload suite has been defined, which can fulfil the mission objectives. It consists of a gas chromatograph/gas spectrometer (with aerosol inlet), a nephelometer, solar and IR flux radiometers, a meteorological package, a radar altimeter and the atmospheric microprobe system. An assessment study is currently in progress to integrate all instruments, except the atmospheric microprobe system, into two highly integrated payload suites with a total mass of 4 kg and an average power consumption of 5 W.

Electrical power will be provided by amorphous-silicon solar cells, which are mounted on the gondola surfaces, yielding sufficient power during the day.

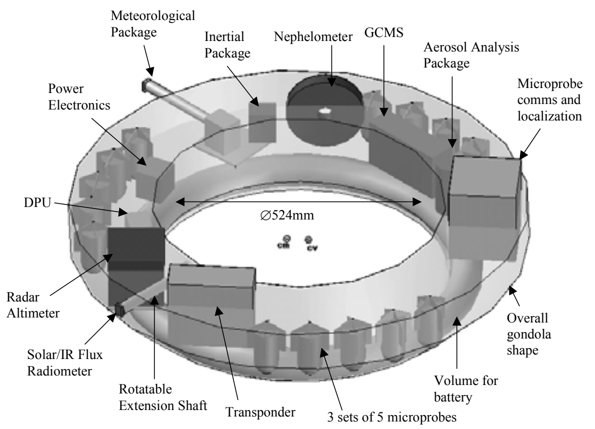


Fig. 3. The layout of the gondola.

During the night, primary or secondary batteries will be used.

Currently, Lithium-thionylchloride primary batteries have been selected as the baseline, as this is the most mass-efficient solution due to their high energy density (~590 Wh/kg). The important drawback of soluble cathode lithium cells is that they are less safe than the more common solid electrolyte Lithium cells. As an alternative, Li-polymer secondary batteries are considered which can be recharged during the day. As the energy density is significantly lower (~170 Wh/kg), the mass penalty for using rechargeable batteries is about 4 kg.

In order to save mass, the payload and communication duty cycles will be substantially lower during the night, resulting in an average night-time power consumption of 5 W, compared to 11 W during the day.

4.6.5 Atmospheric microprobes

The fifteen atmospheric microprobes on board of the aerobot serve a twofold purpose:

- Perform scientific meaningful measurements
- Drop ballast in order to increase the operational lifetime of the aerobot

The atmospheric microprobes measure in-situ vertical profiles of selected properties of the lower atmosphere from the aerobot float altitude down to at least 10 km altitude. Due to the stringent mass limitations of the aerobot, they should be as low-weight as possible. This limits the choice of sensors that can be carried with the microprobes. Currently, the following measurements are foreseen: pressure, temperature, and solar flux levels. The horizontal wind velocity will be deduced from the trajectory of the microprobes. This set of measurements, performed at different longitudes, will provide new insights in the atmospheric dynamics and the heat balance on Venus (see MR2).

In order to investigate both the local weather patterns on Venus as well the global atmospheric dynamics, the 15 microprobes will be dropped in 5 separate drop campaigns, spaced equally over the mission lifetime. The three probes in a drop campaign will be released with an interval of 5 minutes.

Localization and communication of the small microprobes is a challenging task and is therefore subject of a separate technology development activity [15]. A preliminary assessment by Qinetiq indicated a mass of 1.4 kg for the communication and localization system and 104 g for a fully functional microprobe, assuming a 5-year technology development horizon.

5. SUMMARY

The Technology Reference Studies are a tool to identify enabling technologies and to provide a reference for mid-term technology developments that are of relevance for potential future scientific missions. Early development of strategic technologies will reduce mission costs and shorten the mission implementation time. As the enabling technologies mature and mission costs reduce, the scientific community will benefit by an increased capability to perform major science missions possible at an increased frequency.

The Venus Entry Probe Technology Reference Study concentrates on in-situ exploration of Venus and other planetary bodies with a significant atmosphere. The mission profile provides a reference for the development of enabling technologies in the field of atmospheric entry systems, micro-aerobots, atmospheric microprobes and highly integrated miniaturized payload suites.

ACKNOWLEDGEMENTS

The support of the study team at Surrey Satellite Technology Ltd., in particular Adrian Woodroffe, Alex Cropp, Nadeem Ghafoor and Jeff Ward, is gratefully acknowledged. Julian Harris from Swiss Space Technology is acknowledged for assessing the miniaturization of the electronics for the payload instruments.

REFERENCES

1. See http://sci.esa.int/science-e/www/object/index.cfm?fobjectid=33170.

2. *Venus Express*, Mission Definition Report, *ESA-SCI(2001)6*, 2001. Available at www.rssd.esa.int/SB-general/Missions.html.

3. Oyama K.-I., Imamura T. and Abe T., Feasibility study for Venus atmosphere mission, *Advances in Space Research*, Vol. 29, 265-271, 2002.

4. Moroz V. I., Studies of the Atmosphere of Venus by Means of Spacecraft: Solved and Unsolved Problems, *Advances in Space Research,* Vol. 29, 215-225, 2002.

5. Titov D.V. et al. Missions to Venus, *Proc. ESLAB 36 Symposium,* ESA SP-514, 13-20, 2002.

6. Taylor F.W., Some fundamental questions concerning the circulation of the atmosphere of Venus, *Adv. Space Res.,* Vol. 29, 227-231, 2002.

7. Cockell C.S., Life on Venus, *Planetary and Space Science*, Vol. 47, 1487-1501, 1999.

8. Schulze-Makuch D. and Irwin L.N., Reassessing the possibility of life on Venus: Proposal for an astrobiology mission, *Astrobiology*, Vol. 2, 197-202, 2002.

9. Blamont J., The exploration of the atmosphere of Venus by balloons, *Advances in Space Research*, Vol. 5, 99-106, 1985.

10. Kerzhanovich, V., et al. Venus Aerobot Multisonde Mission: Atmospheric relay for imaging the surface of Venus, *IEEE Aerospace Conference Proceedings*, Vol. 7, 485-491, 2000.

11. Klaasen K.P. and Greeley R., VEVA Discovery mission to Venus: exploration of volcanoes and atmosphere," *Acta Astronautica,* Vol. 52, 151-158, 2003.

12. S. Kemble, S., Taylor, M.J., Warren, C. and Eckersley, S., Study of the Venus microsat in-situ explorer, TRM/IP/TN1, EADS Astrium Ltd., 2003.

13. Seiff A., et al. Models of the structure of the atmosphere of Venus from the surface to 100 km, *Advances in Space Research,* Vol. 5, 3-58, 1985.

14. Andreev R.A., et al. Mean zonal winds on Venus from Doppler tracking of the Vega balloons, *Sov. Astron. Lett.,* Vol. 12, 17-19, 1986.

15. Microprobe localization and communication prototype system under development by Qinetiq (ESA TRP contract 17946/03/NL/PA).

11th SPIE International Symposium on Remote Sensing

Gran Canaria, Spain,

13-16 September

SCI-AP

European Space Agency
Agence spatiale européenne

Science Payload and Advanced Concepts Office (SCI-A)
Planetary Exploration Studies Section (SCI-AP)
2200 AG Noordwijk, Postbus 299
The Netherlands

On the study of highly integrated payload architectures for future planetary missions

S. Kraft*, J. Moorhouse, A.L. Mieremet, M. Collon, J. Montella, M. Beijersbergen (1) J. Harris (2)
M.L. van den Berg, A. Atzei, A. Lyngvi, D. Renton, C. Erd, P. Falkner (3)

(1) Cosine Research B.V., Leiden, The Netherlands
(2) Swiss Space Technology, Champery, Switzerland
(3) ESA-ESTEC, Science Payload and Advanced Concepts Office, Noordwijk, The Netherlands

ABSTRACT

Future planetary missions will require advanced, smart, low resource payloads and satellites to enable the exploration of our solar system in a more frequent, timely and multi-mission manner. A viable route towards low resource science instrumentation is the concept of Highly Integrated Payload Suites (HIPS), which was introduced during the re-assessment of the payload of the BepiColombo (BC) Mercury Planetary Orbiter (MPO). Considerable mass and power savings were demonstrated throughout the instrumentation by improved definition of the instrument design, a higher level of integration, and identification of resource drivers. The higher integration and associated synergy effects permitted optimisation of the payload performance at minimum investment while still meeting the demanding science requirements. For the specific example of the BepiColombo MPO, the mass reduction by designing the instruments towards a Highly Integrated Payload Suite was found to be about 60%. This has endorsed the acceptance of a number of additional instruments as core payload of the BC MPO thereby enhancing the scientific return. This promising strategic approach and concept is now applied to a set of planetary mission studies for future exploration of the solar system. Innovative technologies, miniaturised electronics and advanced remote sensing technologies are the baseline for a generic approach to payload integration, which is here investigated also in the context of largely differing mission requirements. A review of the approach and the implications to the generic concept as found from the applications to the mission studies are presented.

1. INTRODUCTION

For most recent European scientific missions, such as ROSETTTA, Mars Express, SOHO, and Herschel/Planck, individual instruments were developed usually on the basis of the heritage of instruments from former missions. In principle, this concept reduces development times and development costs to a minimum whilst allowing instrument capability and performance to mature through actual flight performance assessment. On the other hand, only a limited evolution through new technologies can be supported, and these have both cost and technical difficulties, which need to be solved in the usually tight schedules associated with payload (P/L) and spacecraft (S/C) developments. Additionally, the current approach of building a payload suite out of separate instruments is, in general, not the most mass-efficient approach. As an alternative, it might be possible to achieve drastic mass reductions, ultimately enabling the use of small spacecraft for planetary exploration, if both P/L and S/C are assessed on system level at the very beginning in the assessment or concept phase, before the start of the instrument design phase. The idea of a small S/C is not new and was addressed earlier [1,2,3] and several missions were initiated or developed in order to demonstrate the feasibility of the small satellite concept. A historical overview of the developments in the United States is given in [4]. Here the costs of microSats are compared to the costs of larger satellites with higher performance. Of course one obvious advantage of small satellites is that within the same budget, more satellites can be launched, which reduces the total risk of failure (at all mission levels). The main advantages of small satellites are:

I. Reduced mission preparation time
II. Smaller effective project & industrial teams
III. Easier interface reduction and standardisation
IV. System level aspects are addressed in a timely and multi-mission manner

* Corresponding author: Stefan.Kraft@cosine.nl, Tel. +31 71 528 4968

V. Reduced number of different components (space qualification facilitated)
VI. Reduced launch costs
VII. More frequent and faster launch possibilities (more recent technologies can be employed)

Although there is a general consensus on the potential for resource reduction through sharing and miniaturisation, there is still a debate about the effectiveness, and the associated risk, if new technologies need to be employed. The benefits from both mission and S/C point of view have been discussed in [5]. For example the Clementine mission to the Moon was built within 22 months according to a microSat concept and has cost only 2/3 of a conventional mission, although it has a rather complex payload [4]. It is also well known that the integration, testing and documentation of missions with payloads comprising discrete separate instruments is tremendous and that interface definition can take years; in fact the mass of the interface control documents exceeds sometimes that of the spacecraft. Since a change in this P/L concept influences the whole chain involving P/L and S/C development including technology issues as well as P/L procurement approaches, it is also highly desirable to understand the impacts of such a new approach. For this reason these aspects of such a system level P/L concept are studied by deriving a preliminary architecture of a Highly Integrated Payload Suite (HIPS) for the BepiColombo Mercury Planetary Orbiter (MPO) with a view to establishing the development, assembly and verification tasks required. This MPO payload serves as a typical example, which could be designed either in a classical manner or using a highly integrated (HIPS) approach and it is used here to mature the resource estimations of the payload of the other mission studies.

2. PAYLOADS OF PLANETARY TECHNOLOGY REFERENCE STUDIES

Technology Reference Studies are mission studies, that are not part of the ESA science program, but which have the purpose to identify the technical development requirements for potential future scientific missions. For planetary exploration, the primary objective is to explore ways to decrease cost and risk by studying the feasibility of small satellite missions, which would allow a phased and systematic approach to the exploration of the planetary bodies of the solar system. The studies were selected to address a wide range of challenging technologies for future exploration of the solar system. The following TRSs are currently under study:

1. **Jovian Minisat Explorer** – a mission to Jupiter's moon Europa
2. **Venus Entry Probe** – an Aerobot for in-situ exploration of the Venus atmosphere
3. **Interstellar Heliopause Probe** – a probe into the interstellar medium towards the bow shock
4. **Deimos Sample Return** – a zero gravity landing manoeuvre to bring back 1 kg from the moon of Mars
5. **MiniMarsExpress** – small sat mission comparative to Mars Express

This paper describes the aims of these missions with a particular view to the payload requirements and the identification of the pro and cons of the HIPS concept. More details on the complete mission scenario, including S/C, launch, cruise, communication, orbit and their feasibility, can be found in ref. [8,9,10,11]. Similarities of the payload requirements are investigated so as to derive a road map of technology developments which are required to enable the presented mission concepts, where all spacecrafts are to be launched as a single or double composite on-board a Soyuz-Fregat SF-2B launch from French Guyana.

Parallel to these investigations, the HIPS concept and the related instrumentation for the BepiColombo mission is being studied further, thereby serving as a reference to prepare a realistic architecture of the P/L and to be able to compare HIPS to the conventionally implemented and distributed P/L. The status of the design case is beyond the scope of this paper and will be presented elsewhere.

2.1 Jovian Minisat Explorer (JME)

JME consists of two satellites, one of which is used as a relay station for data transmission and the observation of the Jovian system. The second orbiter shall map the moon Europa in a circular orbit at a distance of 200 km. The payload on the Jovian Relay Satellite (JRS), and especially on the Jovian Europa Orbiter (JEO) is constrained by the extreme radiation environment close to Jupiter (up to 5 Mrad after 4 mm Al). Since the instruments face a rather harsh radiation environment, it is recommended to apply radiation hard electronics and to shield sensitive components accordingly.

The main purpose of the JRS payload is the observation of the planet Jupiter and its surroundings during two years, provided the lifetime of the satellite and its payload is long enough. After the payload assessment the following instruments have been envisaged for **JRS:**

Table 1 Resource allocations and purpose of the JRS payload.

Instrument	Purpose	Mass (kg)	Power (W)	Data (kbit/s)
Jupiter Radiation Environment Monitor **(JuREM)**	Field mapping of the electron and proton activity and its distribution around Jupiter	1.5	1.70	1.1
Jupiter Plasma Wave Instrument **(JuPWI)**	Plasma wave environment, solar wind interaction with Jovian ionosphere	3.5	1.60	3.75
Jupiter Narrow Angle Camera **(JuNaCam)**	Imaging and spectroscopy of the surface with 10 different colours.	1.5	1.00	9.1
Jupiter Magnetometer **(JuMAG)**	Investigation of the Jovian magnetic field	1.15	0.95	0.25
Jupiter Dust Detector **(JuDustor)**	Measurement of dust present in the Jovian system	1	1.00	0.02
DPU + CPS	Data processing and power supply	2	3.25	-
Shielding (20%)	Shielding of the components	2.13	-	-
Structures	Optical bench and mounting structures	2	-	-
Margin (20%)		2.9	1.9	-
Total		**17.7**	**11.4**	**14.2**

It is intended that the payload shall be embedded in the satellite structure as much as possible. For the payload of JRS, this requirement is slightly relaxed compared to the Europa Orbiter, since the orbit is between 12.7 R_J and 27 R_J. The required effective shielding is only about 5 mm Al equivalent. Nevertheless, the assessment of the available resource revealed that less than 20 kg is available for the JRS payload, which is quite limited for the five instruments. Even more demanding than the low mass requirement is the low power consumption, which is imposed by the low solar flux at the large distance of the Jovian system from the Sun (~5 AU). Analysis has shown that a HIPS approach is the only viable - although still challenging- solution for the selected payload. The mass saving in electronics and the related support structures enables the installation of a payload fulfilling the required performance. One example for resource reduction is the installation of a filter wheel in front of the sensor of JuNaCam instead of in front of the aperture. This allows for a much smaller wheel, compared to a wheel in front of the much larger aperture.

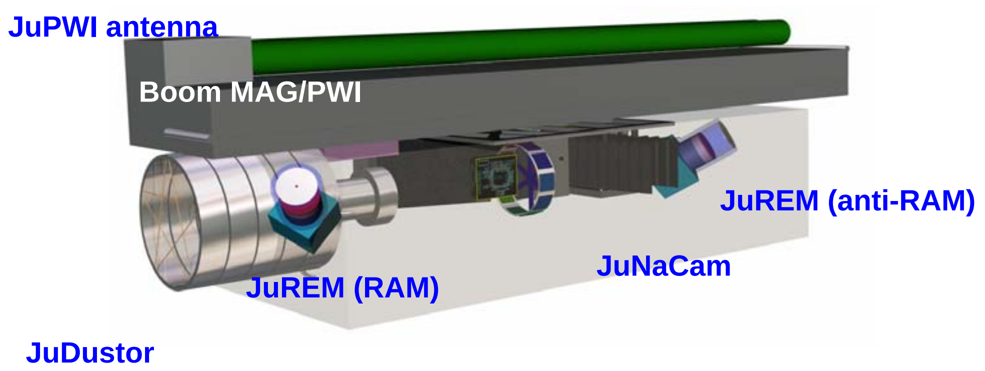


Figure 1 Visualisation of the payload suite. The instruments do not have any demanding requirements on pointing, co-alignment, or thermal requirements and can easily be operated by a central DPU.

The core science of the mission is addressed by the Jovian Europa Orbiter. The main purpose of its payload is the observation of Jupiter's moon Europa during a relatively short period of 60 days. The instruments face a rather harsh radiation environment (5 MRad), requiring a combination of radiation hardened electronics and external shielding to protect sensitive components accordingly. Also here the payload shall be embedded in the satellite structure as much as possible. The following instruments are envisaged for **JEO**:

Table 2 Resource allocations and purpose of the JEO payload.

Instrument	Purpose	Mass (kg)	Power (W)	Data (kbit/s)
Europa Ground Penetrating Radar **(EuGPR)**	Mapping of the surface and subsurface properties of Europa down to ~20km depth	9.6	20	1.5
Europa Stereo Camera **(EuS-Cam)**	Stereographic imaging of the surface to derive full topography map	0.6	1.2	5
Europa Visible Near IR Mapping Spectrometer **(EuVN-IMS)**	Imaging and spectroscopy of the surface at a spatial and spectral resolution of up to 30m/px and 30 nm resp.	2	2	13
Europa Radiometer **(EuRad)**	Determination of the temperature profiles of Europa in particular at the equator	2	1	0.1
Europa Laser Altimeter **(EuLAT)**	Topography of the surface and measurement of tidal effects	2	2.5	3
Europa Magnetometer **(EuMAG)**	Investigation of the presence of a magnetic field of Europa and its interaction with Jupiter	1.4	0.5	0.25
Europa UV Spectrometer **(EuUVS)**	Mapping of interaction of the ionosphere of Jupiter with Europa	0.5	0.5	TBD
Europa Gamma-ray Spectrometer **(EuGS)**	Investigation of the elemental surface composition	3	1	TBD
Europa Radiation Environment Monitor **(EuREM)**	Field mapping of the electron and proton activity and its distribution around Europa	1.5	1	1.1
DPU + CPS	Data processing and power supply	2.5	4	-
Structures	Optical bench and mounting structures	2	-	-
Shielding (20%)	Shielding of the components	5.4	-	-
Margin (20%)		6.5	6.8	-
Total		**39**	**40.5**	**24**

Implementation of ground penetrating radar is particularly demanding. Further savings may be achieved by a light-weight antenna technology. The instrumentation relies on a micro-laser altimeter, a camera with a visible-NIR sensor with broad spectral range and low power requirements throughout, thereby asking for highly miniaturised and integrated electronics.

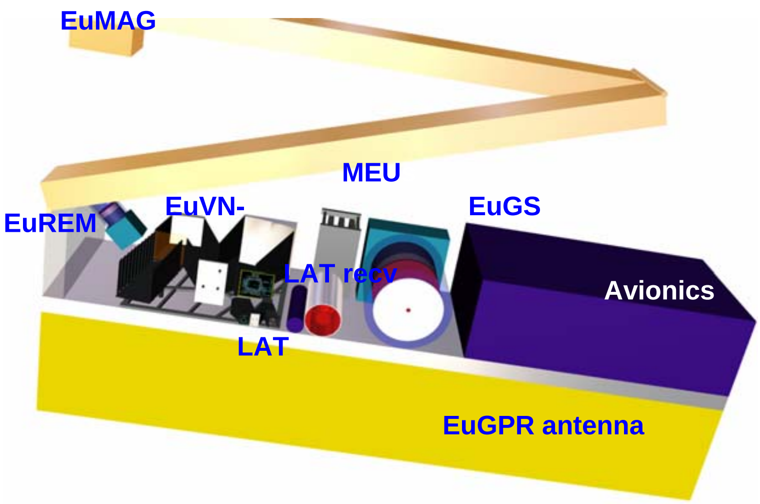
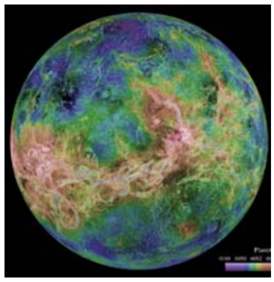


Figure 2 Conceptual layout of the JEO payload. The accommodation is preliminary and will be changed.

2.2 Venus Entry Probe (VEP)

The VEP mission study is designed to undertake the following science investigations:



1. The origin and evolution of the atmosphere by measuring the abundance and isotopic ratios of noble gases
2. Composition and chemistry of the lower atmosphere by determining the minor (<1%) constituents
3. Atmospheric dynamics by accurate measurements of vertical profiles of pressure, temperature and wind velocity
4. Aerosols in cloud layers by measuring the size distribution and temporal and spatial variability of the number density as well as chemical composition
5. Surface and subsurface investigations

These objectives can be summarized as the overall aim to fully understand the atmosphere of Venus in all its aspects and to explore the Venus surface and tectonic structure. The mission scenario that is able to fulfil these objectives consists of two small satellites: the Venus Elliptical Orbiter (VEO) and the Venus Polar Orbiter (VPO) and an Aerobot. The VPO, with the bulk of the atmospheric remote sensing payload, will operate in a polar orbit with altitude at perigee and apogee of about 2000 and 6000 km respectively. This orbit is selected for the study of atmospheric dynamics requiring high spatial and temporal resolution (the orbital period is about 3 hours).

The VEO primarily acts as a data relay station, but will also carry payload more suited to a highly elliptical orbit. The Aerobot will operate at an altitude of approximately 55 km within the Venusian middle cloud layer to derive *in situ* information. The Aerobot design is driven, in particular, by the need to operate in the harsh atmospheric environment of Venus and by a very tight mass budget. During flight, the Aerobot will release small probes which provide height profiles of pressure, temperature, solar flux levels and wind speed.

The VEO operates the following instruments:

Table 3 Resource allocations and purpose of the VEO payload.

Instrument	Purpose	Mass (kg)	Power (W)	Data (kbit/s)
Venus Surface & Subsurface Radar **(VSSR)**	Surface and subsurface study with high resolution.	12	40	14
UV/ visible camera	UV-CAM2 / tracking of UV features of cloud layers.	1	1	
DPU + CPS	Data processing and power supply	2	2	-
Margin (20%)		3	8.6	
Total		**18**	**51.6**	**21**

The VEO carries the radar instrumentation for (sub)surface investigations, which has a limited operational altitude, and a UV/visible camera for obtaining images of the complete globe at far distances. Though the topology has been completely and accurately mapped, the subsurface has never before been sounded.

The payload selection for **VPO** is driven by the penetration characteristics of radiation through the atmosphere. TIR and UV radiation can only provide information on the upper part of the atmosphere and part of the cloud layer. Through NIR radiation, it is possible to observe down to the ground in several NIR window regions. Imaging of the lower atmosphere therefore relies on several of these NIR spectral windows; different spectral channels may probe different atmospheric layers. NIR radiation is also suited to the study of dynamics by monitoring the motion of the cloud layers: while the lower atmosphere is sounded spectrally, cloud opacity can be spatially resolved because the clouds are highly, but conservatively, scattering. The microwave instrument has the attractive features of being able to measure temperature down to around 50 km and to resolve individual spectral lines from which Doppler shifts and hence velocities may be inferred.

Table 4 Resource allocations and purpose of the VPO payload.

Instrument	Purpose	Mass (kg)	Power (W)	Data (kbit/s)
Venus Ultraviolet Spectrometer **(VUVS)**	Spectroscopy of H_2O, SO_2, COS, CO, noble gases and unknown UV absorbers; study and mapping of night glow emissions as dynamics tracers; EUV spectroscopy.	4	4	10
Venus UV-Camera **(VUVCam)**	Tracking of UV features of cloud layers.	1	1	3
Venus Visible Near IR Mapping Spectrometer **(VN-IMS)**	Tracking of NIR cloud features to study dynamics, esp. super-rotation; monitoring of the O_2 airglow at 1.27 µm; study of the cloud opacity and its variations; spectroscopy of NIR windows, including search for volcanic activity and study of surface temperature.	4	14	10
Venus IR radiometer **(VRad)**	Tracking of cloud IR features (especially at poles); H_2O mixing ratio; heat transfer; measurements of the outgoing thermal spectral fluxes (radiative balance); temperature/pressure sounding	4	3	10
Venus Micro Wave Sounder **(VMS)**	CO and H_2O mixing ratios, temperature/pressure and wind speed profile from Doppler shifts in limb and nadir views.	6	20	10
DPU + CPS	Data processing and power supply	2	4	-
Margin (20%)		4.2	9.2	8.6
Total		**25.2**	**55.2**	**51.6**

The remote sensing payload will provide new studies in the form of microwave and subsurface exploration and improve upon former studies. The orbit of the VPO offers the possibility of complete global coverage of the upper atmosphere over the length of a super-rotation period (4 days) and a temporal resolution of 3 hours, invaluable for study of the polar vortices for example. Most of the instruments can be miniaturised and well integrated into HIPS, with the exception of the radar instrumentation, largely due to the large antenna. For this reason and the requirement of a low altitude perigee, the ground-penetrating radar is accommodated on the VEO.

The remote sensing measurements of VPO are primarily dedicated to support and enrich the Aerobot investigations. The tentative payload that be integrated into the **Aerobot** and its purpose are given in Table 5:

Table 5 Resource allocations and purpose of the Aerobot payload.

Instrument	Purpose	Mass (kg)	Power (W)	Data (kbit/s)
Gas Chromatograph/Mass Spectrometer **(GCMS)**	Abundance and isotopic ratios of noble gases, minor gases (e.g., SO_2, COS, HCl, H_2S and H_2O)	0.8	5	TBD
Aerosol analysis package **(AAP)**	Analysis of particles of Venus' atmosphere	0.3	2	TBD
Solar and IR Flux radiometers **(FR)**	Measure the radiation transport and heat transfer properties of the atmosphere	0.2	1	TBD
Meteorological package **(MP)**	Pressure, temperature, light level, flux, acceleration	0.5	1	0.3
Inertial package **(IP)**	Measure acceleration and changes in attitude	0.05	1.2	
Radar altimeter **(RALT)**	Determine the position of the Aerobot	0.9	10	
DPU	Data processing	0.25	0.25	-
Structures	Optical bench and mounting structures	0.3	-	-
Margin (20%)		0.7	4.09	
Total		4.0	24.95	TBD
Total (incl. duty cycle)		**4.0**	**5.15**	TBD

For reasons such as mass distribution and to be able to keep the option to observe the atmosphere on both sides of the Aerobot, the payload has been split into two HIPS, which are fully integrated into the gondola. Here the resources are extremely low, therefore requiring extremely high miniaturisation and integration of the instruments.

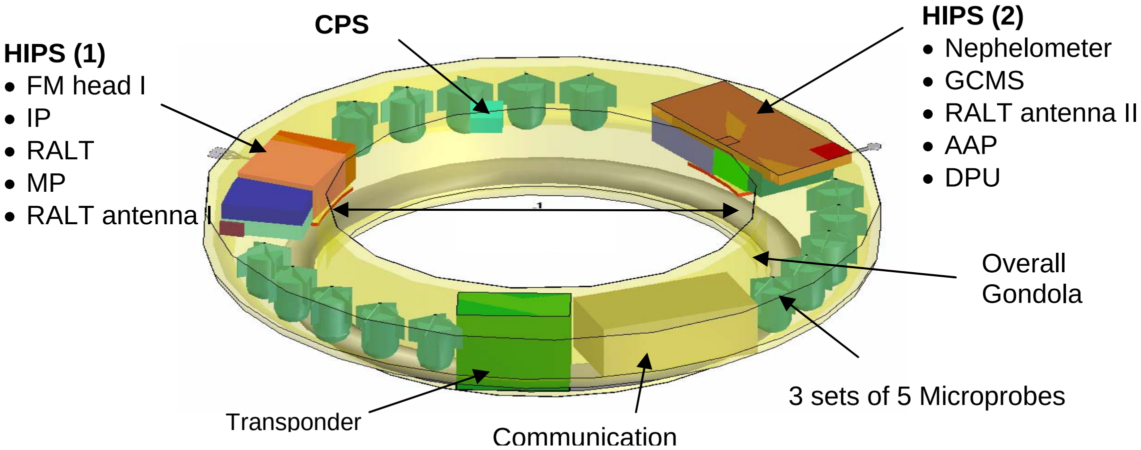
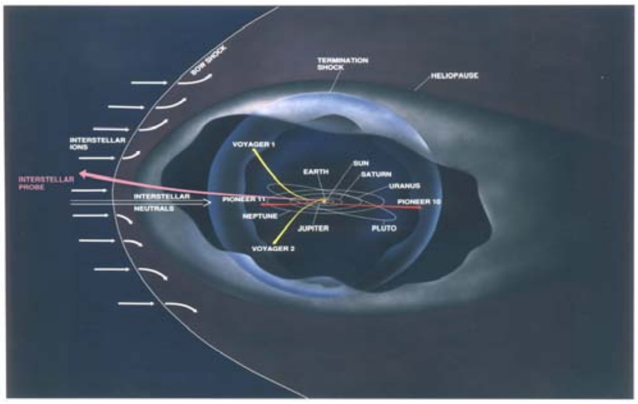


Figure 3 Conceptual design of the payload core of the gondola of the Aerobot with the two envisaged HIPS.

With the exception of the Aerobot, the VEP mission is not particularly constrained by power nor are the mass requirements particularly demanding, although lowering the mass of the VPO payload allows a less eccentric orbit, more suited to the type of global mapping that can unravel the mystery of the Venusian dynamics. Thus in this case, the introduction of the HIPS concept mainly allows an enhancement of the instrument performance and thereby the scientific objectives through resource savings.

2.3 Interstellar Heliopause Probe (IHP)

IHP is to perform chemical and plasma measurements in the heliosphere, the interstellar medium and the interface region between them. The vehicle shall reach a distance of 200 AU from the sun within 25 years. In order to explore the interstellar medium in the shortest time possible the spacecraft shall travel in the direction of the Heliosphere nose, which is located at 7.5° latitude and 254.5° longitude in ecliptic coordinates. In order to minimize the attitude manoeuvring a spinning spacecraft is envisaged. IHP will be the first spacecraft designed to leave the solar system and to enter the interstellar medium. No direct observations of this region exist today. Hence the main objectives of the IHP will be to:



1. explore and investigate the interface between the local interstellar medium (LISM) and the heliosphere,
2. to investigate the influence of the interstellar medium on the solar system,
3. to investigate the influence of the solar system on the interstellar medium, and
4. to explore the nature of the interstellar medium and the outer solar system and the heliosphere.

Additionally a secondary objective might be to observe Trans-Neptunian Objects (TNO) during cruise.

The main purpose of this payload is therefore the study of plasma, energetic particles, magnetic fields, and dust in the outer heliosphere and nearby interstellar medium with a focus to the investigation of the conditions close to the termination shock. The 3-dimensional characteristic of the heliopause requires in principle observations from multiple sides. Since only one S/C is available it is at least tried to have a large coverage of the observations asking for large field of views of the instruments.

Observations aim at the determination of the composition of the plasma and the determination of particle energies and travelling directions of the plasma. The rather broad range of energies from suprathermal to high energetic GeV particles and even neutral atoms requires a whole suite of instruments. The dust grain composition and directional information shall be investigated in-situ. Remote sensing of the dust and the interstellar clouds shall be enabled by UV, VIS-NIR and FIR measurements. The strawman payload is limited in mass and power to 20 kg and 20 W, respectively. This requires a high degree of miniaturisation, integration and demands resource sharing among all instruments. The limited time for communication and lack of interaction requires highly autonomous instruments and a high degree of data compression. The total mass that can be shipped by solar sailing transportation is less than 20 kg.

Table 6 Resource allocations and purpose of the IHP payload.

Instrument	**Purpose**	**Mass (kg)**	**Power (W)**	**Data (bit/s)**
Interstellar Plasma Analyser **(IPA)**	Determine the elemental and isotopic composition of plasma and the associated energy levels at temporal composition	2	1	10
Interstellar Plasma Wave and Experiment **(IPWE)**	Determine the plasma and radio wave environment in outer space CO	5.5	2.5	23
Interstellar Magnetometer **(IMAG)**	Magnetic field measurements in very low fields	3.2	2.5	8
Interstellar Neutral and Charged Atom Detector and Imager **(INCADI)**	Energy levels, composition, mass, angular and energy distribution of neutral atoms	0.5	1	16
Interstellar Energetic Particle Detector **(IEPD)**	Measure supra-thermal, and energetic ions and electrons energy distributions	1.8	1.2	14
Interstellar Dust analyzer **(IDA)**	Determine the energy levels of cosmic rays	1	0.5	1
Interstellar UV photometer **(IUVP)**	Surface and subsurface topology with high resolution, altimetry	0.3	0.3	10
Interstellar Visible NIR Imager **(IVI)**	Determine the radial distribution of Small Kuiper belt objects and TNO	1	0.5	10
Interstellar FIR Radiometer **(IFIR)**	Measurement of the radial distribution of dust and the cosmic infrared background	0.3	0.2	1
DPU + CPS	Data processing and power supply	2	3.5	-
Structures	Optical bench and mounting structures	2	-	-
Margin (20%)		3.92	2.64	18.6
Total		**23.52**	**15.84**	**111.6**

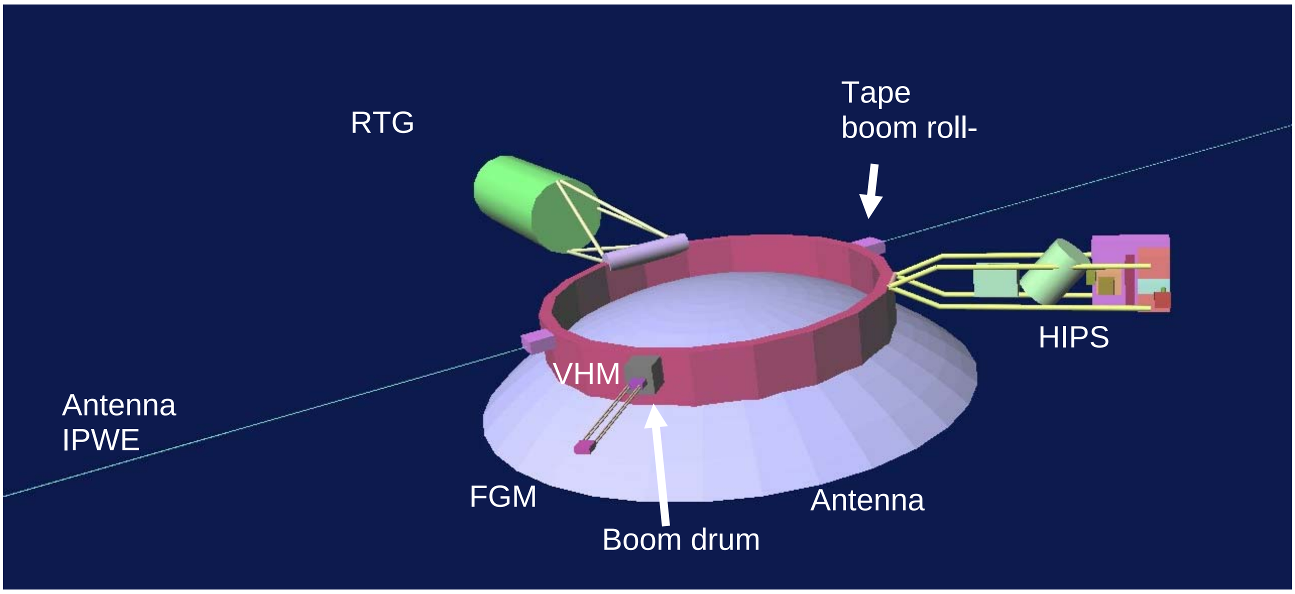


Figure 4 Potential S/C accommodation as far as the payload is concerned. S/C units not included.

2.4 Deimos Sample Return

Two MicroSats launched as a single composite from a Soyuz-Fregat SF-2B shall be inserted into Mars Orbit. One MicroSat acts as a Data Relay Satellite and return vehicle for a Deimos sample and return capsule. The second MicroSat will rendezvous with Deimos to perform a 1 kg sample capture and return to the data relay satellite, which will then leave Mars orbit for a return to Earth, where the capsule will perform a direct re-entry. In the intended single MicroSat scenario, the operations of both satellites are combined aboard one spacecraft. The payload consists as a minimum of a landing system, which allows imaging of Deimos and a distance measurement with the aim to derive landing coordinates and terrain information. Other scientific objectives are the determination of Deimos' size, shape, orbit, gravitational field, rotational properties, surface features and composition. A sufficiently small landing system would allow implementing also some scientific instruments, which could be beside the camera a NIR spectrometer, a UV spectrometer and a scanning system which allows the topographical mapping of the moon. The payload is still under assessment; therefore Table 7 is only indicative.

Table 7 Resource allocations and purpose of the DSR payload.

Instrument	**Purpose**	**Mass (kg)**	**Power (W)**	**Data (kbit/s)**
μ Stereo Imaging Laser Altimeter (**μSILAT**)	Landing coordination, surface topography, shape, size; measure mineralogical composition of the surface (NIR spectroscopy); measure distance during landing and approach	2	3.5	30
Radio Science Experiment **(RSE)**	Measure Doppler shift during approach	2	6	1
Magnetometer **(MAG)**	Search for and map intrinsic magnetic fields	0.5	0.5	1
UV photometer **(UVP)**	Investigate halo and potential exosphere	0.3	0.5	1
DPU + CPS	Data processing and power supply	1	1	-
Structures	Optical bench and mounting structures	1	-	-
Margin (20%)		1.2	2.3	-
Total		**8.2**	**13.8**	**33**

2.5 MiniMarsExpress

The MarsExpress mission is well known and is taken as reference in order to compare the conventional mission with the same mission instrumentation performance implemented in an advanced highly integrated manner. The resources of the instruments of both mission payload concepts are compared in the following table:

Table 8 Resource allocations and purpose of the MEX(*) and MiniMEX(**) payload – still preliminary.

Instrument	**Purpose**	**Mass* (kg)**	**Power* (W)**	**Mass** (kg)**	**Power** (W)**
High Resolution Stereo Camera **(HRSC)**	Stereo mapping of Mars with different colours	21.4	40.4	6	2
NIR spectral imager **(OMEGA)**	Observatoire pour la Mineralogie, l'Eau, Glace, l'Activite	28.8	47.6	5	15
Planetary Fourier Spectrometer **(PFS)**	Investigation of the atmosphere of Mars	31.2	45	5	3
UV/NIR spectrometer **(SPICAM)**	Spectroscopy for the Investigation of Characteristics of the Atmosphere of Mars	4.9	25	1.5	3
Plasma Analyser **(ASPERA 3)**	Analyser of Space Plasmas and EneRgetic Atoms	5.95	6.4	4	4
Subsurface Radar **(MARSIS)**	Radar (Subsurface & Ionospheric Sounding)	15	59	10	15
DPU + CPS	Data processing and power supply	-	-	2	4
Structures	Optical bench and mounting structures	-	-	1	-
Margin (20%)		-	-	6.9	9.2
Total		**107.25**	**223.4**	**39.7**	**55.2**

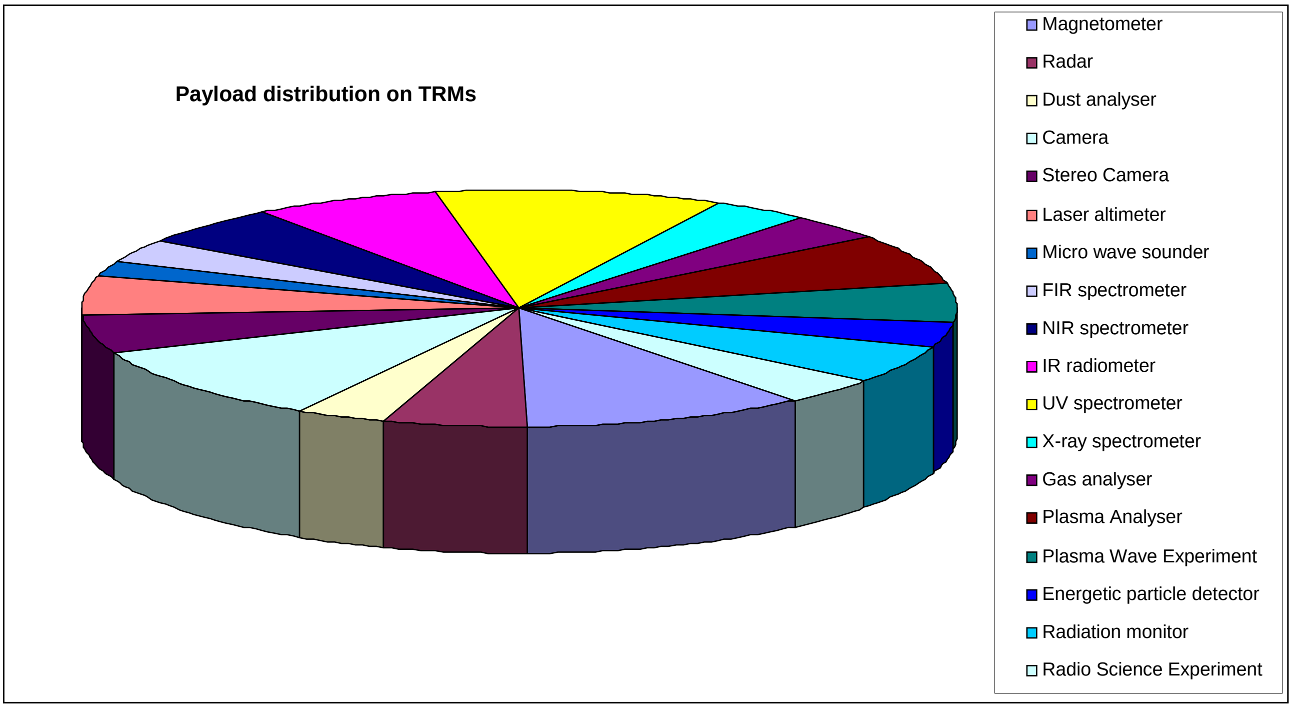
The given resources are preliminary and are still under assessment. The main gain of resources results from the provision of a high performance and centralised DPU, which serves the instruments HRSC, OMEGA, PFS and MARSIS, and from the use of common resources. Instrument concepts and detector technologies are mature for most of the instruments, but must be revisited in the frame of recent developments. It can however already be seen that a saving of about 50% is expected in mass and even 70% in power. Including the snowball effect (multiplication factor of satellite weight for a given increase of payload mass) which is usually ~3, this means that a modern MiniMEX mission would give room for a second S/C being launched with the same rocket and at the same time it would even endorse more or better performing science payload. A new mission to Mars would most likely shift the scope of the scientific instruments, but would still be within the here given resource envelope.

3. GENERIC PAYLOAD

3.1 Instruments

An overview of the payload for all mentioned missions including the payload for the BepiColombo mission shows clearly the need of future technology developments. 18 types of instruments are required in total to cover the scientific demands of the presented 8 orbiters having a total of 52 instruments as strawman payload.

Figure 5 Statistical visualization of the amount of payload types assessed in the technology reference studies.



The highest demand is obviously on magnetometers, cameras and UV spectrometers, and it seems to be feasible that all these instruments can be built from generic components. The ranking of instrument developments according to that chart is the following:

1. Generic fluxgate magnetometers with optional vector Helium magnetometer and miniaturised electronics
2. Cameras being flexible to be changed in aperture size, sensor adaptation and filtering concept with an option of integrating a stereo channel and a laser altimeter
3. UV spectrometers with scalable aperture (photometry is an additional demand)
4. IR radiometer with optional spectrometric capability and broad band spectral range
5. Plasma analyser with possible accommodation of field-of-view

A limited amount of instrument concepts and technologies is needed to realise the observed instrument requirements. The conducted study gives a great insight into the feasibility of building generic instruments or components for scientific space instrumentation, and it allows proposing a roadmap into the future.

3.2 Components

Within the scope of this paper, the particular needs towards generic instrumentation cannot be addressed sufficiently. However, a short list of some of the identified key technologies is given here:

1. Deployable large antennae (subsurface radar)
2. Deployable booms with flexible length for spinning and non-spinning S/Cs (magnetometers)
3. Advanced instrument structures and materials (plastics and lightweight alloys with similar stiffness and thermal conductivity as Aluminium) and their qualification
4. Smart baffles (reflecting thermal heat)
5. Filter technologies (interference filters); perhaps even integrated onto the sensors
6. Optical fibres, and micro-collimators
7. Linear variable and patched filters
8. Sensors being coupled to a passive cooler (radiator)
9. Sensors with low power consumption (CMOS technology)
10. Room temperature bolometers
11. Field Programmable Gate Arrays (FPGAs) and Application Specific Integrated circuits (ASICs)
12. Highly miniaturised Data Processing Unit (DPU) and bus system

3.3 Electronics

The DPU performance handling different requirements for different missions must be very flexible or scalable. One way to achieve this is to use a scalable processor paradigm such as SPARC (Scalable Processor ARChitecture). This type of system is designed for use in a multiprocessor system and supports the concept well. With the latest advancements in the LEON core design, this is particularly well suited to a space qualified multiprocessor system approach. There are many approaches to multiprocessor systems, although since recommendations have already been made towards the use of the LEON SPARC-V8 architecture (see Figure 6), which is now followed as baseline. The SPARC concept directly supports the SMP (Symmetrical MultiProcessor) idea which itself has a number of approaches. Two of these approaches include the shared memory multiprocessor, and the distributed memory model. The LEON architecture supports the shared memory model and the SPARC standard supports this directly in its memory model. Specific instructions for multiprocessing are also supported within the SPARC concept, which include atomic load-store operations. IP cores for the implementation of the different functions are made available mostly and considered as generic components.

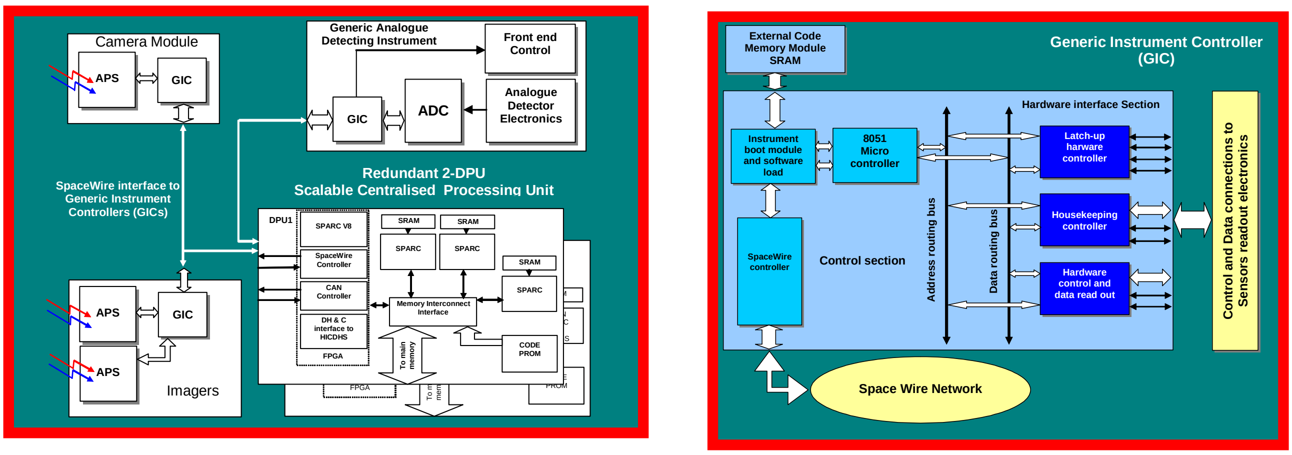


Figure 6 Single Chip Multiprocessor system using LEON SPARC architecture (left) and conceptual layout of the generic instrument controller (right).

The Concept of the Generic Instrument Controller (GIC), allows the central DPU to be able to communicate with all surrounding instruments in the same way. With only minor modifications to the sensor interface, a standard set of functions in the GIC will enable the DPU to "talk" to many differing types of instrument. This will reduce development efforts, not only at the instrument end, but also in the centralised data processing unit. With only one type of interface for communication, the DPU can be highly standardised, and scalable. Many of these system modules can be realised using FPGA technology. Some generic ASICs shall be developed. This also has advantages in mass, size and power consumption. In some cases, whole circuit boards can be replaced by a single programmable component with inter-module connections being simply handled within the device. The processing performance can be adapted from some up to several hundred MIPS while consuming only a few hundred mW.

4. CONCLUSION

Technology Reference Studies are a tool to identify enabling technologies and to provide a reference for mid-term technology developments that are of relevance for potential future scientific missions. Early development of strategic technologies will reduce mission costs and shorten the mission implementation time. As the enabling technologies mature and mission costs reduce, the scientific community will benefit by an increased capability to perform major science missions possible at an increased frequency. The presented technology reference studies have been taken as a showcase for the investigation of the needs on advanced instrumentation for future highly miniaturised and integrated payloads. The term highly integrated is used here not in a literal sense, and is meant more in the sense to provide the basis for a symbiosis being able to benefit from the synergy effects. Instruments can still be distributed and are only combined in case this is subject to a clear advantage. The autonomy of each instrument can still be high, although the 'central brain' may observe and command executive payloads. For the presented approach the total payload mass of a satellite is typically around 30kg and weighs therefore as much as single instruments aboard former conventional missions. This might open a new road towards many science driven missions and a future approach for the exploration of the solar system and beyond.

ACKNOWLEDGEMENT

We thank the involved colleagues from ESTEC, EADS Astrium, Surrey Satellite Technology Ltd, Alcatel and Kayser-Threde for the provision of the mission study parameters as far as S/C and orbiter implementation aspects were concerned.

REFERENCES

1. Wertz, James, *Radical Cost Reduction Methods*, in Wertz and Larson (1996), pp. 34–53
2. Wertz, James, and Simon Dawson, *What's the Price of Low Cost?* paper presented at the 10th Annual AIAA/USU Conference on Small Satellites, Torrance, California: Microcosm, Inc., 1996.
3. Sarsfield L., *The Cosmos in a String*, National Book Network, 2000
4. Bille M., *Microsatellites and improved acquisition of space systems*, Journal of Reducing Space Mission Cost 1: 243–261, 1998 (2001)
5. R. Carli, *System Challenges in the Development of Low-cost planetary Missions*, ESA SP-542 (2003) p. 143
6. M. Collon, *Design and performance of the payload instrumentation of the BepiColombo Mercury Planetary Orbiter*, ESA SP-542 (2003), p. 501
7. S. Kraft, *On the Concepts of a highly integrated payload suite for use in future planetary missions: the example of the BepiColombo Mercury Planetary Orbiter*, ESA SP-542 (2003) 219
8. D. Renton, P. Falkner and A. Peacock, ESA SP–543 (2004), pp. 3-10 (Deimos Sample Return Technology Reference Mission)
9. A. Lyngvi, P. Falkner and A. Peacock, ESA SP–543 (2004), pp. 11-16 (The Interstellar Heliopause Probe)
10. A.C. Atzei, P. Falkner, M.L. van den Berg, A. Peacock, ESA SP-543 (2004), pp. 17-22 (THE JUPITER MINISAT EXPLORER)
11. M.L. van den Berg, P. Falkner, A.C. Atzei, A. Peacock, ESA SP–543 (2004), pp. 23-27 (VENUS ENTRY PROBE)