UPDATE ON ESA´S TECHNOLOGY REFERENCE STUDIES

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ABSTRACT

The concept of Technology Reference Studies (TRS), set up by ESA´s Science Payload and Advanced Concepts Office (SCI-A) to focus the development of strategically important technologies of likely relevance to future science missions, has already been introduced in 2004 at the 55th IAC in Vancouver [1]. Significant progress in the definition of the mission concepts and related technology requirements has been achieved since then. At the present time the Planetary Exploration Studies Section of SCI-A has finished the study of the first four TRSs, the Venus Entry Probe (VEP), the Jupiter Minisat Explorer (JME), the Deimos Sample Return (DSR) and the Interstellar Heliopause Probe (IHP). Current study activities are now focusing on the extension of the Jovian Explorer scenario towards magnetospheric and atmospheric investigations by means of additional orbiter(s) and entry probes. New introduced concepts deal with cross-scale constellation (CSM) of up to 12 spacecrafts to further explore the Earth magnetosphere and a Near Earth Asteroid Sample Return (ASR). All TRS mission profiles are based on small spacecraft, with miniaturized highly integrated payload suites (HIPS) and launched on Soyuz Fregat-2B (SF-2B) as baseline. TRSs are set up to provide thematic context for technology development based on feasible mission concepts, which may be also used by the scientific community as embryonic building blocks for future mission proposals. This paper describes the current status of the new concepts under study (CSM, JEP, ASR) and the final results of the first four TRSs (JME, DSR, VEP and IHP) in further detail.

1. INTRODUCTION

Exploration of the solar system under cost pressure demands development of small spacecrafts and many new technologies. Feasibility, cost and programmatic aspects (e.g. development of individual technology items) of new mission concepts and proposals depend on a clear understanding of the required technologies and their development in time. Significant cost drivers during later programme phases are the product of technically failed or delayed technology developments. Technology Reference Studies (TRS) are set up to provide a set of realistic requirements and thematic context for technology developments within ESA's technology programmes in preparation of future scientific missions and in particular for ESA's scientific Cosmic Vision 2015-2025 [2].

2. TECHNOLOGY REFERENCE STUDIES

The TRS are chosen to cover a wide area of scientific topics ranging from astrophysics and fundamental physics to planetary exploration. They should complement ESA's current scientific programme and must be of potential relevance for the future. Four mission concepts have been successfully studied 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). Feasible mission profiles have been defined and individual sets of required technology items determined.

The TRSs are a tool to focus technology development activities and to define the required mission environmental conditions. They are based on feasible low cost mission concepts, which are not part of current ESA's science mission programme, but of potential relevance for the future. The careful selection of the TRSs has been based on analysis of world wide trends and in particular on the input provided by the European Scientific Community in the frame of Cosmic Vision 2015-2025 call for themes [2].

All TRSs must be compatible with a single Soyuz Fregat 2B (SF-2B) low cost launch vehicle (launched from Kourou) and have to stay within the individual targeted cost caps. Envisaged technologies should have a technology readiness level compatible with a launch in the 2015-2025 timeframe. It is important to ensure that only realistic mission scenarios are studied and that the technology requirements can be properly defined and developed in time.

3. VENUS ENTRY PROBE

More than twenty missions have been flown to Venus to date, including fly-bys, orbiters, and in-situ probes, providing a basic description of the planet, its atmosphere, its ionosphere and a complete mapping of the surface by radar. The upcoming ESA Venus Express orbiter (launch October 2005) and Planet-C from JAXA (launch 2007) [3] 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).

3.1 Scientific Objectives

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 to address open scientific questions and topics like:

- Origin and evolution of Venus atmosphere
- Comparative planetology (Venus vs. Earth)
- Composition of lower atmosphere
  - Atmospheric chemistry
  - Runaway greenhouse effect
  - Tracking active volcanism
- Aerosol analysis/exobiology
- Analysis of large (ø ~ 7 µm) cloud particles
- Atmospheric dynamics/thermal balance
  - Super-rotation
  - Hadley cell circulation?
  - Weather patterns in main cloud deck
  - Polar vortices

3.2 Mission Profile

The VEP consists of a pair of small satellites (VEO and VPO), one entry probe with aerobot and fifteen microprobes. VEP is launched on a single SF-2B into a high thrust heliocentric interplanetary transfer to Venus. Launch windows repeat every 1.6 years due to the synodic period of Venus. The VEP composite will enter an elliptical polar orbit (250 x 215.000 km) around Venus after a 120 to 160 day transfer (dependent on launch window).

Mass Budget	VPO (kg)	VEO (kg)
Payload	25.2	0
Entry Probe	0	91.1
Communications	20.2	20.3
Structure	82.7	84.1
Propulsion	63.7	42.3
AODCS & Safety	9.5	9.5
OBHD	4.2	4.2
Environment	21.4	16.7
Power	18.6	7.1
20% system margin	49.1	55.0
Spacecraft dry mass	**294.5**	**330.3**
Propellant	594.2	206.2
Spacecraft wet mass	888.7	536.5
Total mass	**1425**	
Total LV capability	**1446**	

Table 1: VEP mass budget

Instrument	Mass (kg)	Power (W)	Data rate (kbps)
UV spectrometer	4.5	4	7
UV-VIS-near IR camera	1	2.5	2
VIS-near IR mapping spectrometer	4	16	20
Imaging Fourier transform spectrometer	4	3	30
Submm wave spectrometer	6	31	5
Central power supply	1	8.5	
CPU	0.5	2	
20% margin	4.2	13	
TOTAL	**25.2**	**80 (60)**	**64**

Table 2: VPO scientific instrumentation

The Venus Polar Orbiter (VPO) will transfer from the initial insertion orbit into a polar orbit (2.000 km x 6.000 km) optimized for remote sensing in support to the in-situ atmospheric measurements of the aerobot and to address the global atmospheric scientific objectives.

Figure 1: The Venus Entry Probe

The Venus Elliptical Orbiter (VEO) releases the entry vehicle for a steep Venus entry at around 10 km/s, leading to high peak heat flux, but reduced soak time. The current baseline for the entry probe is a 45° sphere-cone aeroshell (Figure 1). The entry angle is limited to 40°, constrained by the maximum allowed 200g peak acceleration for the subsystems. The peak heat flux is around 20 MW/m^2, which requires a dedicated heat shield development and qualification effort. A disk-gap-band parachute will be deployed by a pyrotechnic mortar just above 1.5 Mach (Figure 2). The parachute will slow down the probe to a velocity of 13.7 m/s at an altitude of 54.8 km where the hydrogen super-pressure balloon will be deployed just 716 seconds after the entry (Figure 3). The balloon will float in the middle cloud layer of Venus (55 km at 30°C and 0.5bar) to perform the in-situ science measurements within the lifetime of 15 to 22 days (2 to 3 Venus circumnavigations).

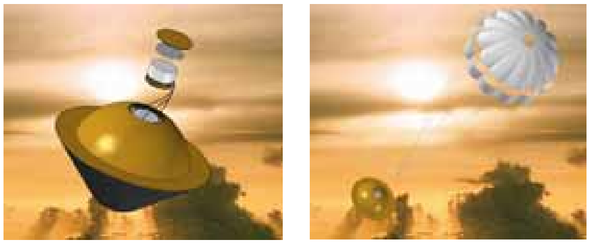

Figure 2: Pilot chute deployment and probe descending on the main parachute [5]

The balloon envelope material needs to have an extremely low leakage rate with possibly welded seams. Remaining gas leakage is compensated by an ammonia gas replenishment system and ballast dropping in the form of fifteen microprobes. The microprobes will perform measurements during their descent in the Venus atmosphere, tracked by the aerobot to define their descent trajectory.

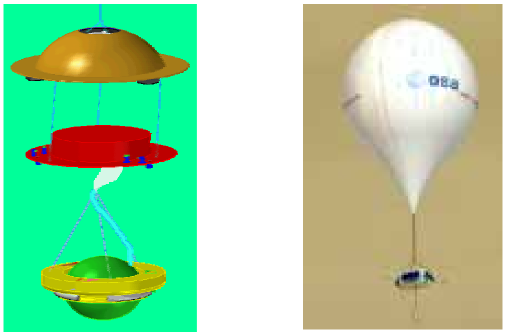

Figure 3: The Venus Aerobot, during deployment and after inflation

The aerobot gondola has a highly miniaturized payload package with an extremely low average power demand. During the day power is provided by amorphous-silicon solar cells, which are mounted on the gondola surfaces. During the night, primary batteries will be used.

3.3 Mission Challenges and Technologies

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.

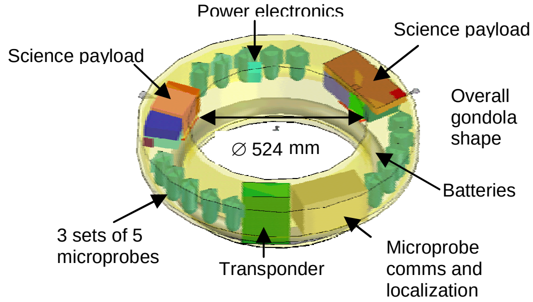

Figure 4: VEP aerobot gondola

The microprobes require substantial development, as they should be limited to around 115 gram mass to meet the stringent constraints 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 performed by QinetiQ [6],[7].

4. DEIMOS SAMPLE RETURN

Deimos and Phobos have accreted ejecta material from all over Mars's surface during different eras. Modelling suggest that approximately 10% of the upper regolith material on Deimos, likely originated from Mars [8]. This Mars component generally consists of Noachian basin forming (4.6 to 3.8 billion years ago) and late heavy bombardment impact material (4.0 to 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, made it an attractive target for a dedicated TRS.

4.1 Mission Profile

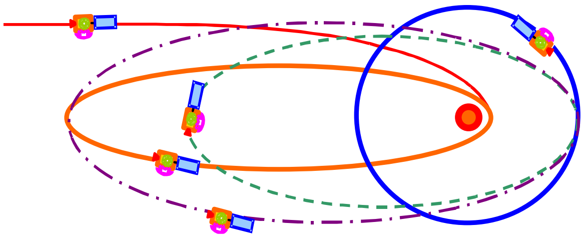

Figure 5: DSR Mars insertion and co-orbit with Deimos

DSR is launched on a SF-2B into a highly elliptical Earth orbit, using its own main engine to escape for transfer to Mars [9]. Insertion at Mars is done into a 500 km x 100.000 km orbit, followed by orbit circularisation to obtain co-orbit with Deimos at approximately 20.069 km. A slightly modified inclination and eccentricity in respect to Deimos' orbit allows for observations of the body before landing and sampling.

The different eccentricity will produce a relative elliptical motion about Deimos with a 30-hour period. Touch-and-go sampling is performed to avoid difficult anchoring procedures on the surface of Deimos. The DSR spacecraft will return after sampling to the same highly elliptical Mars orbit from where it will perform the Mars-Earth transfer followed by a direct entry at Earth return.

Launch	DeltaV (km/s)	Duration (yrs)	Stay Time (days)	S/C Dry Mass (kg)	Launch Mass Margin (%)
2011	4.86	3.27	148	606	32.1%
2013	5.25	2.85	445	621	21.8%
2016	5.76	2.76	518	642	6.4%
2018	5.08	3.04	111	611	27.9%
2020	5.16	3.17	90	614	25.8%

Table 3: DSR launch opportunities, with 2016 being non feasible within the current mission baseline

The current mission profile requires a total delta-V of approximately 4.9 to 5.8 km/s, with total mission duration of 2.7 to 3.3 years, depending on the launch date. Several launch opportunities exist in the 2010 to 2020 timeframe as given in Table 3.

4.2 Spacecraft

The main driver for the design of the spacecraft configuration is its capability to perform the touch-and-go sampling manoeuvre. The baseline consists of two stages, the propulsion stage and Earth Return Vehicle (ERV). The propulsion stage consists of the main engine for the propulsive manoeuvres prior to sampling (Earth-Mars transfer, Mars Orbit insertion, Mars orbit manoeuvres, Deimos Observation Orbit), the GNC instruments, the mechanisms for obtaining and transferring the sample and the landing pads.

The propulsion stage is no longer required after the sampling operations and will be separated and left behind in Martian orbit. For this reason the propulsion stage was placed 'below' the ERV (Figure 6) so that it could act as a buffer in case of any hard impact during the touch-and-go sequence. All delicate components required for the return transfer are placed on the ERV, as far as possible away from where the spacecraft will contact the surface, to protect these components from any ejecta that result from the touch-and-go and sampling sequence.

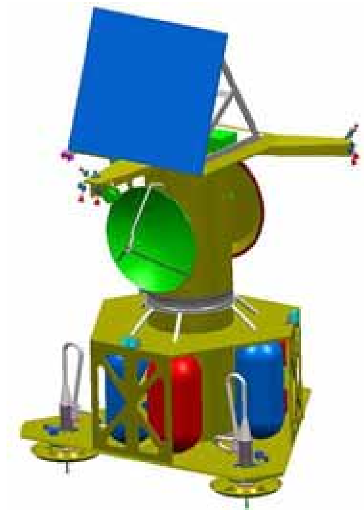

Figure 6: DSR spacecraft

The solar array is fixed in order to withstand the impact and placed on top of the spacecraft to minimize possible damage or degradation from ejecta. The HGA and ERC are located on the sides of the ERV, facing in opposite directions. Two LGA antennas have been placed on the extreme edges of the upper panel. Thrusters for attitude control and for use during the touch-and-go sequence are also located on these edges of the upper panel.

4.3 Mission Challenges and Technologies

The sampling mechanism is of prime importance. 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.

Sampling is done during touch-down. A penetrator injects gas into the soil, which is compressed by spring loaded landing gears (Figure 7). Small soil particle will mainly follow the pressure gradient towards a intermediate storage container.

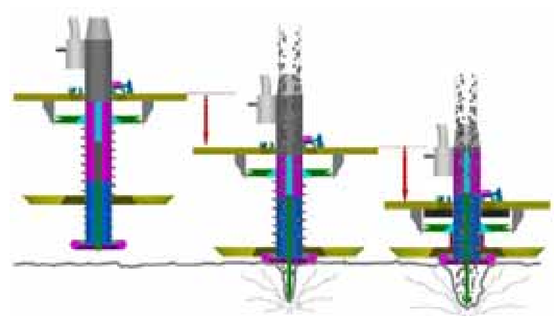

Figure 7: DSR pneumatic sampling mechanism

A highly autonomous guidance, navigation and control system enhanced with narrow angle (NAC) and wide angle (WAC) cameras, triple laser ranger and micro laser ranger, are required to guide the spacecraft during its approach to the surface, sample collection and return to orbit, without interaction from Earth mission control.

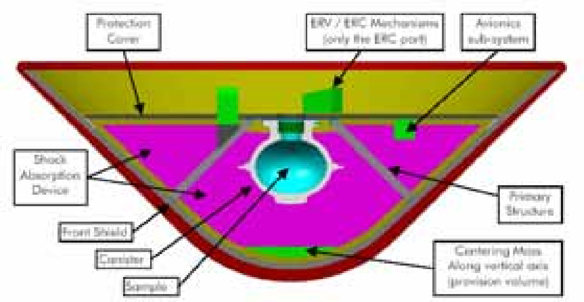

Figure 8: The DSR Earth re-entry capsule

The Earth return vehicle (Figure 8) requires substantial European development, albeit several studies have already been performed on such systems mostly in the frame of Mars sample return mission scenarios.

An additional challenge could be due to planetary protection constraints. 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 all phases of the transfer back to Earth, during re-entry and has to survive any kind of impact scenario to prevent Earth contamination. This would have a severe impact on the mission design.

In summary the main mission challenges are:

- Gas injection penetrating sample mechanism
- Distant and proximity guidance and navigation
- Earth return vehicle
- Thermal protection system for Earth re-entry
- Planetary protection to prevent Earth contamination

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 Voyager 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 (1992) used a Jupiter gravity assist to swing out of the ecliptic plane towards an orbit around the poles of the Sun and at the same time provided valuable information on the Jovian radiation and magnetic environment. Galileo, launched in 1989, provided, in spite of severe data return reduction difficulties, the most extensive study of the Jovian system to date, including in-situ measurements of Jupiter's atmosphere by means of an atmospheric probe. Also Cassini delivered interesting data during its Jupiter fly-by in December 2000 on its way out to Saturn.

5.1 Objectives

In the first phase of the study [10], [11] emphasis was on remote sensing of Europa, one of the few places in the solar system where liquid water may be found, and hence one of the prime candidates for exobiological investigations. A detailed exploration of the surface and subsurface of Europa, performed on a 200 km circular orbit with 'classical' remote sensing instrumentation onboard of the Jupiter Europa Orbiter (JEO) and possibly a deployment of a microprobe for surface in-situ analysis was envisaged.

The following top-level objectives have been assumed for the study:

- Determine the presence or absence of a subsurface ocean
- Characterize 3D-distribution of any subsurface liquid water and its overlaying layer and other compositions
- Understand formation of surface features
  - Topography
  - Crater densities and composition
  - Presence of cryo-volcanism
  - Ridge systems
- Tidal processes and libration
- Composition and presence of atmosphere or exosphere
- Magnetic field and its origin
- Comparative study of the Galilean moons (fly-by opportunities during the Jovian 1.5 year tour)
- Identify candidate landing sites for future landing missions
- Radiation and Plasma Environment

5.2 Mission Profile

The current scenario [10], [11] foresees two small spacecraft, the Jovian Relay Spacecraft (JRS) and the Jovian Europa Orbiter (JEO) with 580 kg and 373 kg dry mass respectively, launched on a single SF-2B using a classical Venus-Earth-Earth gravity assist (V EEGA) profile for the transfer to Jupiter (Table 4).

Launch Date	Earth escape	DSM	JOI	Total	Mass after JOI	Arrival Date	Duration
	ΔV (m/s)				(kg)		(years)
Jan-17	1321	113	1438	2872	1201	Dec-22	5.9
Jun-18	1306	370	1239	2915	1185	Jul-27	9.1
Feb-20	1906	86	1273	3265	1060	Apr-26	6.2
May-23	1180	288	1245	2712	1264	Aug-29	6.2
Oct-24	1836	87	1321	3244	1067	Jan-32	7.2
Aug-26	1520	4	1364	2889	1195	Sep-32	6.1
Nov-29	1314	555	1353	3222	1075	Apr-37	7.4

Table 4: JME transfer characteristics

The JRS acts as a data relay (X and Ka-band) for JEO, placed in an elliptical orbit (12.7 R_J x 26.3 R_J) outside the harsh radiation zones around Jupiter (Figure 9). JRS will carry all subsystems that are not directly required for the Europa exploration: the communication system for the data and command link between Earth and the JEO, data storage and processing units (Leon based) and a small scientific HIPS, dedicated to explore the Jovian system.

JEO is on a 200 km circular polar orbit around Europa, equipped with a highly integrated remote sensing payload suite including a subsurface penetrating radar, the communication system for communications with JRS and also for direct communication (back-up) and tracking with Earth. JEO science data is 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. The in orbit lifetime of JEO is limited to 60 days because of perturbations by Jupiter’s immense gravity and the harsh radiation environment.

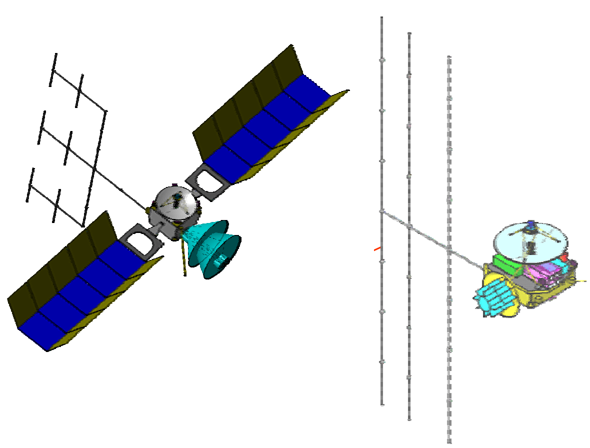

Figure 9: The Jupiter Europa Orbiter (JEO) based on solar power (left) or alternative RPS power (right) with a further developed antenna design for the ground penetrating radar

Europa Orbiter (JEO)	Relay S/C (JRS)
Ice penetrating radar	Magnetometer
Mini stereo camera	Radiation monitor
VIS/NIR mapping spectrometer	VIS-NIR Camera
Laser altimeter	Plasma Wave Analyzer
Magnetometer	Dust detector
Radiometer	
Radiation monitor	
γ- ray spectrometer	
UV spectrometer	
Total: 34 kg, 33 W	**Total: 16 kg, 10 W**

Table 5: summarizing the strawman P/L [14]

The power subsystem is based on 14.3m^2 solar generators (identical for both spacecraft) with LILT-technology (Rosetta heritage), radiation hardened and employing solar concentrators (reflectors) [10],[11],[12]. An alternative concept based on RPS has been studied for comparison [13]. Assuming a power density of the RPS system between 4.3 W_e/kg to 8 W_e/kg, and the use of RPS excess heat for thermal control of the spacecrafts (JEO and JRS) a mass gain between 40 to 200 kg and increased operational flexibility could be achieved. Critical Earth gravity assist manoeuvres, political constraints and significant increase in system cost are major drawbacks of the RPS alternative.

Item	Mass incl. Margin (kg)	
	solar	RPS
JEO platform mass	373	301
JEO science instruments capacity	30	35
JEO dry mass	**403**	**336**
JEO propellant mass	253	214
JEO wet mass	**656**	**549**
JRS platform mass	580	528
JRS science instruments capacity	14	18
JRS dry mass	**594**	**546**
JRS propellant mass	1680	1661
JRS wet mass	**2274**	**2207**
Total JME mass	**2930**	**2756**
Adapter mass	70	70
Total launch mass	**3000**	**2826**
launcher capacity	**3000**	

Table 6: JME mass budget comparison for all-chemical mission profile with solar power or alternative RPS

A high-velocity (500 m/s) hard penetrating micro-probe for in-situ investigation on the surface of Europa has been studied as well [15],[16].

5.4 Mission Challenges and Technologies

The extreme radiation environment at Jupiter requires all spacecraft electronics to be protected against radiation levels in excess of 5 Mrad (after 4 mm Al shielding) and also careful selection of all materials. A combination of radiation hardened electronics (exceeding 1 Mrad), special adapted spacecraft subsystems and additional extensive shielding is required (29 kg are reserved on JEO for shielding). Specific care needs to be taken also for electrostatic discharges (ESD) in insulating all materials [17].

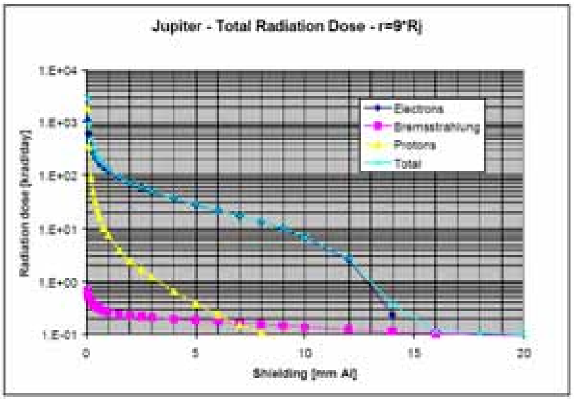

Figure 10: Ionizing dose as a function of shielding thickness for $9*R_J$ equatorial distance from Jupiter

The solar power generators have to be designed for $1/25^{th}$ of the solar flux at Earth using specific adapted GaAs-triple-junction Low Intensity Low Temperature (LILT) cells and additional solar concentrators.

The communication system requires technical development to perform deep space inter-satellite links between JRS and JEO in both X- and Ka-band at high data rates (0.9 Mbps) and for communication with Earth (20 kbps) under the harsh radiation environment.

The long mission duration (including a 1.5 year Jovian tour), the hostile environment and far distance from Earth ask for a highly autonomous mission including the commissioning and operational phase of the instruments.

The high-speed hard penetrating microprobe for Europa requires very challenging technology development. The high velocity impact (500 m/s) needs materials and subsystems capable of withstanding very high impact shocks and g-loads.

JEO will impact Europa after the science phase, imposing strict COSPAR planetary protection requirements to the spacecraft. Constraints on material selection, increased complexity and cost are the consequence. In-flight decontamination by the severe radiation in the Jovian system must be exploited as much as possible.

In summary the top challenges are:

- Radiation hardened components ($\geq$ 1 Mrad) with effective radiation shielding
- Deep space as well as Jupiter's extreme radiation environment
- Radiation optimised solar cells, LILT GaAs development
- Solar Concentrator development to maximise solar power at ~5 AU from Sun
- RPS systems, should solar cells be unfeasible (and for other planets)
- Thermal variations (Venus fly-by hot case, Jupiter cold case)
- Development of highly integrated systems (incl. low resource P/L and avionics)
- Low power deep space communication
- Communication and tracking of entry probe (s) and hard impact micro-probes
- Highly autonomous mission capability
- Planetary protection compatible systems
- Penetrator technologies (500 m/s)
- Radar antenna deployment mechanism

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 (TS) marking the boundary between the interstellar medium and the heliosphere is believed to be at a distance of 80-100 AU from the Sun [19]. This interface region is of particular interest and is the primary target for the IHP [20].

6.1. Objectives

Key scientific questions are [21]:

- What are the characteristics of the solar wind termination shock and heliopause (HP), where are these structures located and how do their characteristics and locations change with time?
- How does the solar wind TS act as an accelerator of (anomalous) cosmic rays?
- How does the heliosphere shield the Solar System from galactic cosmic rays and the interstellar neutral gas?
- What is the state and composition of the local interstellar medium beyond the HP?

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

6.2. Mission Profile

The IHP requires an extreme high Δv to reach the necessary solar system escape velocity of approximately 10 AU/year. Various transfer possibilities have been studied including solar or nuclear electric propulsion and chemical transfers, but all would require larger launchers, than baselined for the TRSs and hence would exceed the low cost approach. Solar sailing has proven to be the only feasible solution for IHP under the given TRS constraints [20].

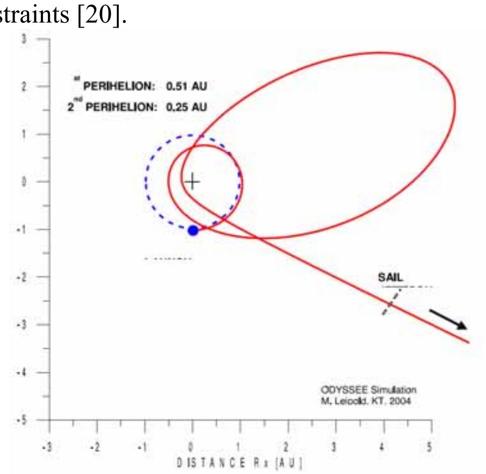

Figure 11: Double photonic assist for the solar sail based IHP transfer [21]

Solar sails utilize the photons emitted by the Sun to accelerate the spacecraft. The achieved acceleration is very small (in the order of a few mm/s^2) and strongly dependent on the distance from the Sun. A 245m x 245m (60.000m^2) deployable solar sail with a characteristic acceleration of 1.1 mm/s2 is needed to satisfy the extreme Δv requirements of IHP. A double photonic assist (Figure 11) close to the Sun at 0.25 AU is selected [21]. Thermal constraints on the booms, spacecraft bus and sail limit the closest possible distance to around 0.25 AU, leading already to a 525K sail temperature. The solar sail is jettisoned at 5 AU after an acceleration phase of around 5 years.

The large extension of the sail structure poses great challenges on storage and deployment of the ultra-thin sail, its supporting structures and on the Attitude Determination and Control System (ADCS) during and after deployment [22]. Four CFRP booms are unrolled from the central deployment module and the sail film segments are released from the sail containers. The deployed booms are limited to less than 100 g/m specific mass and must be able to withstand very high thermal fluxes due to the close approach to the sun. A part of the boom deployment mechanism will be jettisoned after deployment to further reduce the system mass. The baseline for the sail material is a 1.5 µm Polyimide film coated with Aluminium on the front side and Cr on the backside.

A gimballed boom between sail structure and spacecraft bus has been found as the only viable solution for the attitude control of the sail supporting a maximum slew rate of 29° per day (during photonic assist manoeuvres). Tip vanes or micro thrusters on sail structures have proven to be not feasible. A slowly spinning sail concept has been selected to compensate for solar radiation pressure perturbations [22]. The overall sail system mass is 249 kg including all margins.

Item	Mass [kg]
Platform mass	134.4
Scientific payload	20.9
Platform mass margin	26.4
Platform mass with margin	**181.7**
Jettisoned deployment mechanism	43
Solar Sail Assembly	187
Solar Sail Assembly margin	19
Solar Sail Assembly with margin	**249**
System Level margin (20%)	86
Flight System Total Mass	**517**

Table 7: IHP mass budget

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. A high efficiency RTG power subsystem with a specific power of 8 W/kg is required to enable the described IHP concept based on 140 W peak IHP power requirements.

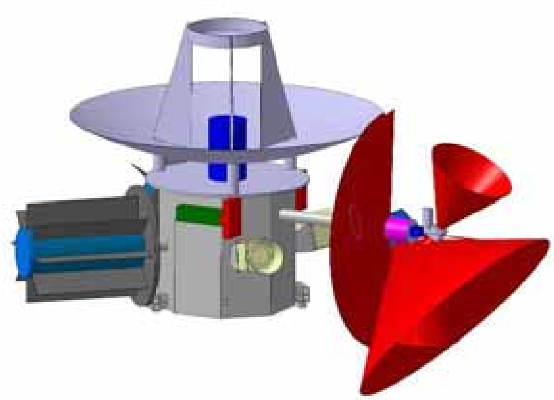

Figure 12: IHP Science configuration with payload field of views

The IHP communication system will be limited to an average downlink data rate of around 200 bps and an uplink rate of 5 bps at 200 AU, based on a pulse modulated Ka-band communication system. This system has been traded against other RF and optical communication solutions.

Instrument	Mass [kg]	Power [W]
Interstellar Plasma Analyzer	2.0	1.3
Interstellar Plasma Wave Experiment	4.5	4.0
Interstellar Magnetometer	3.7	3.4
Interstellar Neutral and Charged Atom Detector and Imager	0.5	1.8
Interstellar Energetic Particle Detector	1.8	1.2
Interstellar Dust Analyzer	1.0	1.0
Interstellar UV- Photometer	0.3	0.3
Solar Activity Monitoring	0.6	3.0
Structures for HIPs accommodation	2.0	-
Central Payload Power Supply	1.0	-
Margin 20%	3.5	3.2
Total	**20.9**	**19.2**

Table 8: IHP strawman payload

The available instrument mass on IHP is strongly constraint. Table 8 summarizes the HIPS, tailored to satisfy the TRS objectives [26].

6.3 Mission Challenges and Technologies

The concept of IHP is very challenging and requires a series of technology developments and space demonstration of solar sailing, similar to that discussed in the roadmap in [23]. The IHP key challenges and requirements are:

- Thin high efficient solar sail
- Sail folding and unfolding technology
- Overall ambiguous system concept
- Highly autonomous with self-maintenance capabilities
- AOCS for the sail during deployment and sailing
- RTG power with 8W/kg output
- Lifetime of the IHP more than 25 years

7. CROSS-SCALE MISSION

ESA's CLUSTER-II mission (launched 2000), a constellation of four spacecraft flying in formation around Earth, is providing in three dimensions very detailed information about how the solar wind affects our planet. Future missions must include close formations of spacecraft with high-temporal resolution, so that spatial variations can be differentiated from temporal evolutions. The logical next scientific step is the multiscale analysis leading to a revolution in our understanding of plasma processes. Multiscale analysis will permit the understanding of how local plasma characteristics are modified by distant perturbations and, in reverse, how local modifications can affect the global scale stability. Important examples of multiscale phenomena are magnetic reconfigurations and substorms, enough motivation to start a new TRS, employing a constellation of up 12 spacecraft (optimum) in three nested tetrahedrons with close (2 to 100 km), medium (50 to 1000 km) and maximum tetrahedron (500 km to 1 R_E) configurations, aiming at high temporal and spatial resolution. A new recently started TRS is focusing on these aspects.

8. NEAR EARTH ASTEROID SAMPLE RETURN

Asteroids, believed to be like fossils, which retain some records of planet-forming, are a unique source of information about the early solar system and the formation of the planets [25]. NEAR-shoemaker (launched 1996) performed a successful visit to an S-class asteroid. Remote sensing and in-situ measurements of asteroids are very useful for characterizing, but have limitations which can only be overcome by returning sample(s) to Earth for use in dedicated high precision analytical laboratory facilities. Hayabusa (MUSES-C, launched 2003) - designed to investigate the asteroid 1989ML (Itokawa) and return a surface sample - just arrived at the target on September 12 and took first close-up images. However it is only expected to return a few grams of material.

Building on lessons learned from the DSR and taking into account the successes of current mission a new TRS is under preparation to address as many as possible of the key questions and investigations [25] on one or possible two asteroids:

- What kind of raw materials formed planets?
- What were the temperature and pressure in the early solar nebula?
- How did asteroids classes form and have their present properties?
- What is the internal nature of asteroids?
- Elemental, Mineralogical and isotopic properties and the geological context.
- Dating of samples

9. JOVIAN MINISAT EXPLORE ENHANCED MISSION SCENARIO

In phase 2 of the Jupiter-TRS additional scientific objectives have been introduced [2], in particular a detailed analysis of the Jovian magnetosphere, aurora, atmosphere and comparative studies of the Galilean moons during fly-bys.

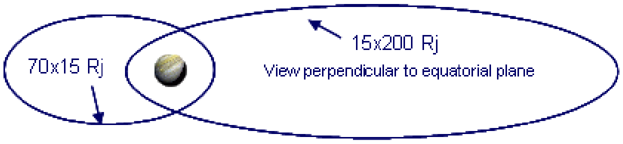

Figure 13: A potential orbit configuration for Jovian Magnetospheric orbiter

Plasma acceleration, magnetic reconnection, structure and dynamics of the magnetic field and its interaction with the Galilean moons are only a few indications of the scientific interests to be investigated with the magnetospheric orbiter(s). One additional SF-2B launch could transfer one or two Magnetospheric Orbiter(s) to Jupiter, to be placed for example at $15R_J$ x $150R_J$ and $15R_J$ x $70R_J$ (Figure 13). Different configurations, inclination and relative orientations are under study. These are strongly influenced by the accommodation of a demanding Jupiter entry probe either with either hyperbolic or orbit deployment.

The Jovian entry, in particular with the ambitious goal to survive even down to 100 bar (Galileo survived down to 30 bar, [18]) atmospheric pressure, is regarded as the most challenging entry in the solar system. The design of the thermal protection system (TPS) suffering heat loads in the order of 300 to 450 MW/m^2 (Galileo Probe 300 MW/m^2) is challenging. TPS mass fractions exceeding 50 % make a probe design difficult. Deployment of (multi-) probes in higher latitudes is scientifically desired, but technically even more challenging because of increased entry velocities and increased system mass. A preliminary mission analysis is done already, the system design studies (magnetospheric orbiter(s), entry probe and the deployment) are currently under preparation.

10. HIGHLY INTEGRATED PAYLOAD SUITES

The concept of TRSs is based on small spacecraft and low cost approaches. The Highly Integrated Payload Suite (HIPS) approach [14], [26]] has been introduced to strongly reduce the payload resource 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 and certainly within the physical limits - is envisaged. The high integration of the instruments also allows for significant reduction of the harness. Payload suites for all TRSs have been studied extensively to provide detailed requirements for the accommodation on the spacecraft and to identify technology development needs in the area of scientific payload. A development of a demonstration breadboard of a selected (reduced) payload suit under ESA contract is envisaged [26]. A set of technology developments for new sensors and payload related items for various instruments aiming further instrument miniaturisation are under way within ESA's technology programmes.

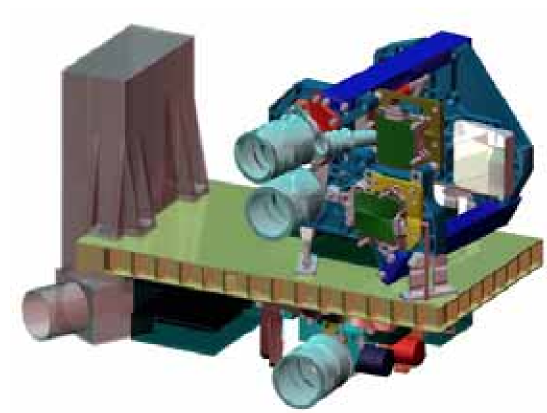

Figure 14: Potential design of a bread board for a reduced HIPS with laser altimeter, cameras, UV and IR mapping spectrometer [26]

11. CONCLUSION

TRS are introduced to identify and develop enabling technologies for potential future science missions and to provide solid references and thematic context during the technology development based on feasible, low cost mission concepts. Four feasible mission concepts have been developed and documents and are serving already as references. Some of the identified technologies are under development while several others are proposed for future development within ESA technology programmes. The TRS concept has proven to be helpful to concretize, select and prioritize technologies for technology roadmaps and plans. A set of new TRSs [24] are under study now, building on lessons learned from the previous activities. More details on the individual TRSs are provided in the references.

12. ACKNOWLEDGEMENTS

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