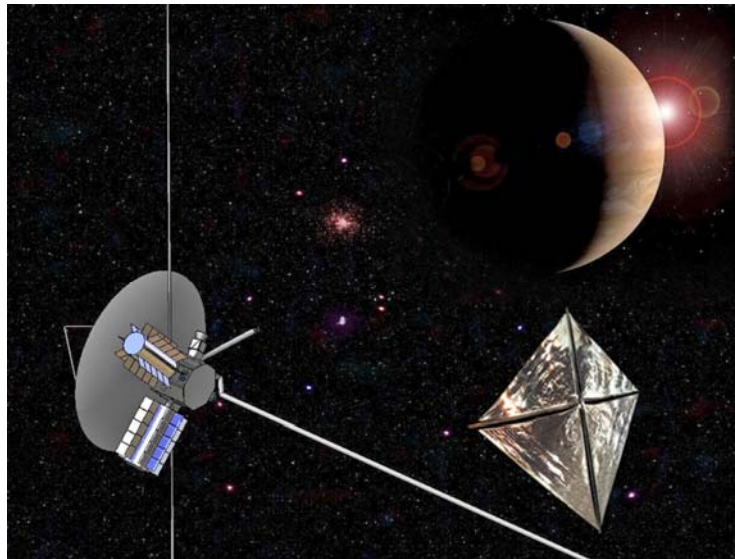


STUDY OVERVIEW OF THE INTERSTELLAR HELIOPAUSE PROBE



AN ESA TECHNOLOGY REFERENCE STUDY

Planetary Exploration Studies Section (SCI-AP)
Science Payload and Advanced Concepts Office (SCI-A)



Sun



Mercury



Venus



Mars



Jupiter



Saturn



Uranus



Neptune



Pluto



Comets



Asteroids

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SCI-A/2006/114/IHP
3
4
17/04/2007
Released
Public report

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Interstellar Heliopause Probe Technology Reference Study Mission Concept Summary		
Key scientific objectives	Investigation of the <ul style="list-style-type: none"> • Outer heliosphere (up to ~120 AU) • The interface between the heliosphere and the local interstellar medium (100 – 200 AU) • Properties of the local interstellar medium (> 150 AU) 	
Strawman reference payload suite assumed for this study	Plasma particle analyzer, plasma wave analyzer, magnetometer, neutral and charged atom analyzer and imager, energetic particle imager, dust analyzer, UV-photometer	
Launch and trajectory	<ul style="list-style-type: none"> • Launch into direct Earth escape by Soyuz-Fregat 2-1B (Kourou) • Solar sail propulsion trajectory with two solar photonic assists towards heliospheric nose (7.5° inclination) • Closest approach to Sun: 0.25 AU • Solar sail jettison at 5 AU (at a spacecraft speed of 10.4 AU/year) • Flight time to 200 AU: 27 years 	
Operational lifetime	<ul style="list-style-type: none"> • Solar sailing phase: 6.7 years • Science acquisition phase: > 20 years 	
Mass budget summary	Mass (kg)	Remarks
P/L mass allocation	21	
S/C platform mass	182	Incl. P/L mass
Sail assembly mass	248 206	At launch After sail deployment (sail container is jettisoned)
Launch mass (without system margin)	431	
Launch mass (with system margin)	517	20% system level margin
Solar sail characteristics	Value	Remarks
Sail size (m ²)	246 × 246	Square sail
Attitude control	2-axis	Sail pitch and yaw by controlled by gimballed boom. Roll axis uncontrolled.
Boom specific mass (g/m)	100	4 booms, each 174 m long
Sail film density (g/m ²)	1.9	
Sail assembly loading (g/m ²)	3.4	Sail mass excluding system level margin
Characteristic acceleration @ 1 AU (mm/s ²)	1.0	Assuming efficiency $\eta = 0.89$, 5% performance margin. S/C and sail mass including system level margin
S/C design	Value	Remarks
Stabilization (science phase)	3 rpm	Spinning S/C for plasma investigations (after sail jettison)
Power @ Beginning of Life (W)	240	Radioisotope Thermoelectric Generators (7 W/kg)
Telemetry band	Ka	
Down-link rate @ 200 AU (bps)	200	Carrier-free low-duty cycle Pulse Position Modulation
Key mission drivers	<ul style="list-style-type: none"> • Travel distance (propulsion, power, communication, thermal) • Close approach to Sun (thermal design) • Operational lifetime 	
Key technological challenges	<ul style="list-style-type: none"> • Solar sail propulsion system (sail material and manufacturing, low-mass booms, sail deployment, attitude and trajectory control, and sail performance) • Radio-isotope power source • Very large distance communication system • Fully miniaturized payload suite • Operational lifetime of all S/C subsystems 	

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1 INTRODUCTION

This document provides an overview of the Interstellar Heliopause Probe system design study. The Interstellar Heliopause Probe (IHP) is one of the Technology Reference Studies (TRS) introduced by 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 that are of likely relevance to potential future science missions. This is accomplished through the study of technologically demanding and scientifically interesting missions, which are currently not part of the ESA science programme (see also <http://sci.esa.int/concepts>). The TRSs subsequently act as a reference for possible future technology development activities.

A mission concept to explore the boundary of the heliosphere has been targeted for a TRS because of the extreme requirements on travel distance. The large distances from the Sun and the Earth drive many aspects of system design: propulsion, communication, power, thermal design and last but not least, subsystem lifetime. Missions to the heliopause and beyond have been studied and proposed before. In the early 1980's the "Thousand Astronomical Units" (TAU) mission was considered based on a 1 MW nuclear powered electrical propulsion system [Etchegaray87]. More recently, McNutt proposed solar thermal propulsion to reach 1000 AU [McNutt04]. Mission concepts for the exploration of the heliopause and the local interstellar medium include the Small Interstellar Probe [Mewaldt95], the Interstellar Probe Mission [Liewer00, Wallace00], the Heliopause Explorer [Leipold06] and the Innovative Interstellar Explorer [McNutt06]. Most of these concepts utilize solar sail propulsion, but also chemical propulsion and radioisotope electric propulsion have been suggested. The concept of a mission to explore the outer heliosphere is thus not new. The primary focus of the IHP TRS is to identify the enabling technologies associated with such a mission concept, and to provide requirements for the development of these technologies in a 10-15 year time frame.

This study overview summarizes the system design study and payload assessment study performed by, respectively, Kayser-Threde [Leipold05], and Cosine Research [Kraft05]. The results of this study have also been presented at a number of conferences and published in several journals (see section 8 for an overview).

2 THE OUTER LIMITS OF THE HELIOSPHERE

The heliosphere is defined as the region in space which is dominated by space plasma originating from the Sun. As the distance from the Sun increases, the Sun's supersonic solar wind becomes weaker and less dense, until it is slowed down and diverted by interaction with the local interstellar plasma, or Local Interstellar Medium (LISM). Little is known about the properties of the LISM. It is e.g. unknown whether it is supersonic or not.

The heliosphere is shaped and structured by the interaction of the solar wind with the Local Interstellar Medium, just like the Earth's magnetosphere is shaped by the solar wind. Because the Sun with its accompanying Heliosphere moves with respect to the surrounding LISM at a relative speed of around 25 km/s, the Heliosphere is tear-drop shaped with the nose at 7.5 degree latitude and 254.5 degree longitude (ecliptic coordinates), as shown in Figure 2-1.

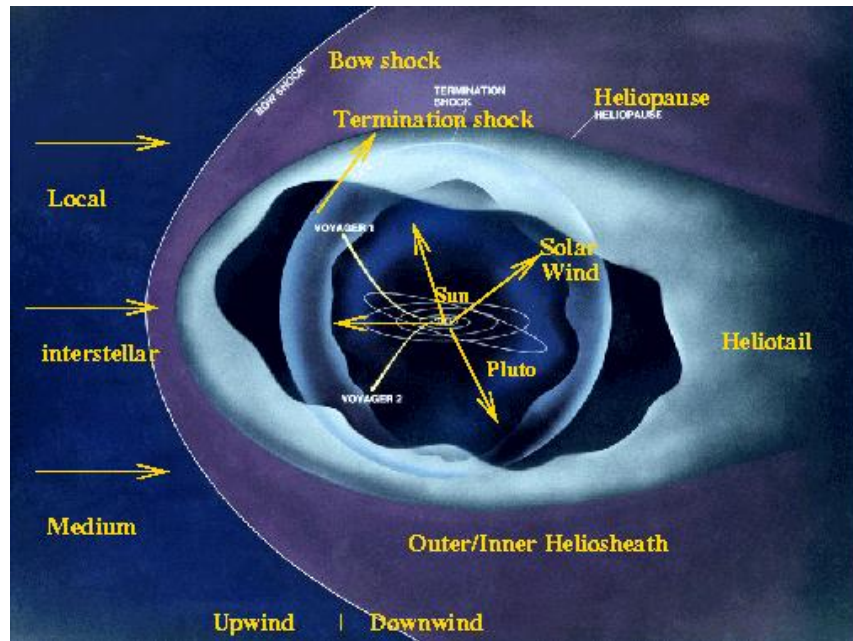


Figure 2-1: The Heliosphere in the Local Interstellar Medium

Because the density of both the solar wind and the LISM plasma particles is so low that direct particle collisions are very rare (collisionless plasma), the solar wind deceleration process is a complex interaction between electromagnetic fields and particles over an extended region, the (inner) heliosheath. The (inner) heliosheath starts at the termination shock (at 80 – 100 AU), where the solar wind decelerates from supersonic to subsonic, and ends at the Heliopause (at 120 – 170 AU, at the nose). The heliopause essentially separates the solar wind plasma from the LISM. If the interstellar plasma is supersonic as well, a bow shock would form on the other side of the heliopause (at 300 – 400 AU). The region between the Heliopause and the (probable) bow shock is known as the outer heliosheath. For a comprehensive review of models describing the physical processes that are likely to occur in the outer heliosphere, the reader is referred to [Zank99].

Though NASA's Voyager 1 has crossed the termination shock at 94 AU in 2004 and is providing valuable measurements of the inner heliosheath, many questions on processes occurring in the outer heliosphere, the heliosheath and the Local Interstellar Medium will remain unsolved, primarily because the 30-year old instruments on Voyager were designed for investigating planetary magnetospheres, which have significantly different plasma characteristics. Additionally, the power source of Voyager 1 will expectedly run out latest in 2020, when the spacecraft has travelled to 148 AU.

The key science objectives of the Interstellar Heliopause Probe are:

- *What are the characteristics and location of the termination shock and the heliopause, and what is the temporal variation?*
- *How are (anomalous) cosmic rays accelerated in the Heliosheath?*

- *What are the processes in the Heliosheath that (possibly) affect the characteristics of the interstellar neutral gas and dust as well as galactic cosmic rays interactions?*
- *What are the characteristics of the local interstellar medium beyond the heliopause?*

To achieve above scientific objectives, in-situ measurements from the outer heliosphere into the Local Interstellar Medium are required.

3 MISSION ANALYSIS

3.1 Propulsion trade-off

The primary driver for the IHP TRS is the requirement to travel to the heliopause within approximately 25 years. To reach the heliopause and the Local Interstellar Medium in the shortest possible time, the spacecraft will have to travel in the direction of the heliosphere nose, which is located at 7.5° latitude and 254.5° longitude in the ecliptic coordinate frame. Even then, a solar system escape velocity of more than 10 AU/year (~47 km/s), equivalent to about three times the current speed of Voyager, is required. Table 3-1 lists the propulsion options that have been considered for the IHP-study. All scenarios assume a launch with a Soyuz Fregat 2-1b from Kourou.

Table 3-1: Propulsion options for the IHP TRS.

	Scenario	Description	Launch mass
A.	High Thrust Chemical Propulsion	Chemical propulsion with $I_{sp} = 320s$ to $370s$; Cryogenic first stage $I_{sp} = 470s$, and Nuclear Thermal Propulsion (NTP)	3020 kg (to GTO)
B.	Mixed High/Low Thrust propulsion	Combination of chemical propulsion with Solar Electric Propulsion (SEP) with $I_{sp} = 3500s$ to $8500s$	3020 kg (to GTO)
C.	Low Thrust Nuclear Electric Propulsion (NEP)	Ion Propulsion with nuclear electric power supply: advanced RTG or fission reactor, $I_{sp} = 5000s$ to $10000s$	3020 kg (to GTO)
D.	Low Thrust Solar Sail Propulsion	Ultra-light weight solar sail for cruise in the inner solar system, with maximum acceleration at 1 AU of $a_c = 0.75 \text{ mm/s}^2$ to 3.0 mm/s^2	2030 kg (to Earth Escape)

For the propulsion system trade-off, the following criteria have been taken into account:

- Launch vehicle (and associated launch costs)
- Time to travel to 200 AU
- Achievable payload mass at 200 AU
- Technology horizon
- Launch window flexibility (e.g. dependency on planetary constellations, i.e. Jupiter)
- Radiation dose received during Jupiter gravity assist (if any) and/or close solar fly-by (< 0.5 AU)
- Safety/Risks associated with radio-isotope power sources, particularly during launch and possible lunar and Earth gravity assists

The mission analysis trade-off showed that scenario A., B. and C. are not attractive:

- High thrust chemical propulsion will only be capable to bring a spacecraft net-mass of 55 to 120 kg to 200 AU in a 30-year transfer trajectory.
- The mixed high-low thrust propulsion will allow a useful spacecraft mass of 200 kg in 25 years at 200 AU, provided that a very close solar approach is made at 0.018 AU. The latter is not considered feasible from a thermal point of view.
- The low-thrust nuclear electric propulsion option would require a 5 - 20 kW nuclear reactor with total mass less than 500 kg (radioisotope power systems are not feasible due to the large amount of plutonium required). Additionally, the electrical propulsion system should be capable of thrust times in excess of 20 years. Because both technology requirements are considered extremely challenging, the nuclear electric propulsion option has been discarded as a realistic solution for the medium term.

Given the mission requirements (launch vehicle, time to travel) and the technological constraints, the trade-off led to the selection of the Solar Sail option as the preferred baseline for the system design of the IHP TRS.

3.2 *Principle of solar sailing*

The attractiveness of solar sail propulsion is that it uses no propellant: Solar sails utilize the momentum of photons. The maximum or characteristic acceleration (a_c) of a sailcraft at 1 AU from the Sun is given by:

$$a_c = 2\eta \frac{S_0 \cdot A}{c \cdot m_{sc}}$$

with A the solar sail area, η the sail efficiency, m_{sc} the total sailcraft mass, S_0 the solar flux at 1 AU (1368 W/m^2), and c the speed of light ($3.0 \times 10^8 \text{ m/s}$). The factor 2 appears because for an ideal sail with perfect specular reflectivity the momentum change of the photon is twice the original momentum (incoming photon momentum and reverse momentum of outgoing momentum). Conservation of momentum dictates that this is all transferred to the solar sail. Of course, realistic sails also have a diffuse reflectivity, absorption and transmission. These effects, including possible solar sail degradation are accounted for in the sail efficiency η . To obtain a typical characteristic acceleration of 1 mm/s^2 for a 200 kg spacecraft, a solar sail with an efficiency of 0.85 and a size of $154 \times 154 \text{ m}^2$ would be required. Though this acceleration may seem small, a continuously applied acceleration of 1 mm/s^2 , would result in a velocity change of $\sim 30 \text{ km/s}$ per year.

The actual acceleration and direction of the thrust depends on the distance to the Sun as well as the orientation of the sail with respect to the Sun. Figure 3-1 illustrates two opposite cases. On the left, the sail is oriented such that the orbital velocity increases, thus increasing the semi-major axis. On the right, the sail is oriented to decrease the orbital velocity, which causes the sailcraft to 'fall' towards the Sun. Clearly it is also possible to change the inclination of the orbit by manoeuvring the sail normal out of the ecliptic plane.

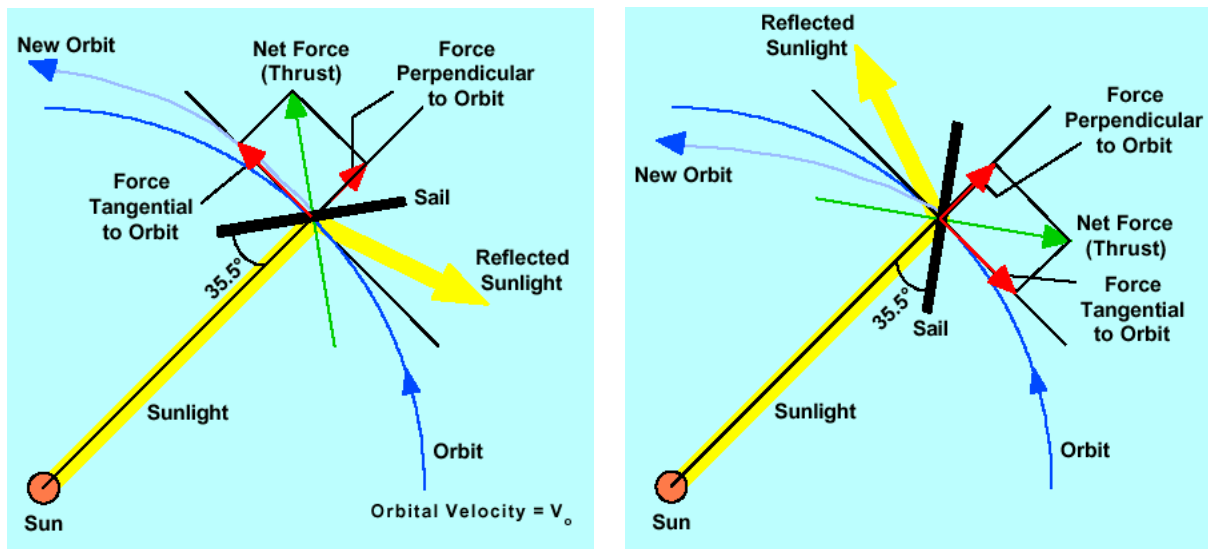


Figure 3-1: Thrust force as a function of solar sail orientation.
 Sailing away from the Sun (left) and sailing towards the Sun (right). After [McInnes04].

The attitude of the sail can be controlled in several ways. This will be discussed in more detail in section 4.3.6. For a comprehensive overview on solar sailing dynamics, the reader is referred to [McInnes04].

3.3 *Solar sail trajectory optimization*

In order to reach the heliopause in the shortest possible time, a trajectory optimization needs to be performed. Continuous outward spiralling will not yield the optimal solution; because the solar radiation pressure decreases with the square of the distance, the final speed of the spacecraft will be limited (at 5 AU, the characteristic acceleration is 4% of that at 1 AU). It is therefore more effective to first spiral inwards towards the Sun and use the increased solar radiation pressure to speed up the spacecraft. Depending on the characteristic acceleration and the minimum distance, one or more of these solar photonic assists (SPA) will yield the optimal result.

The optimization of the IHP solar sail trajectories has been performed with the ODYSSEE program, which uses a sequential trajectory optimization method [Leipold00]. Table 3-2 list the scenarios considered for the IHP TRS. Additional results were obtained in a trajectory optimization study by the University of Glasgow [Macdonald07]. For all scenarios, it has been assumed that the sail will be jettisoned at 5 AU after which the probe travels to the heliopause subject to a small deceleration due to the Sun's gravitational force (this effect increases the travel time with one to two years, depending on the speed at sail jettison).

Clearly, the higher the characteristic acceleration, the less solar photonic assists are required. For thermal reasons, the minimum distance from the Sun has been set to 0.25 AU, with the exception of the scenario with the lowest characteristic acceleration. The last scenario also requires a Jupiter Gravity Assist (JGA) to fulfil the requirement of less than approximately 25 year travelling time to the heliopause.

Table 3-2: Solar sail scenarios analyzed for the IHP mission concept study.

Scenario	D1	D2	D3	D4
Characteristic acceleration, a_c (mm/s ²)	3.00	1.50	1.00	0.85
Hyperbolic excess, C_3 (km ² s ⁻²)	0	0	0	0
Minimum solar distance, r_p (AU)	0.25	0.25	0.25	0.16
Number of Solar Photonic Assists	1	1	2	2 + JGA
Velocity @ 5 AU	15.0	9.6	10.4	10.9 ¹
Travel time to 5 AU	2.7	3.8	6.7	7.1
Travel time to 200 AU	16.0	26.3	27.0	26.2

The differences in total travel time between the different scenarios are notable. As expected, the highest characteristic acceleration will provide the fastest trajectory to 200 AU (scenario D1). Decreasing the characteristic acceleration with a factor of two, increases the travel time with only 65% (scenario D1 vs D2). Further lowering of the characteristic acceleration down to 1.0 mm/s² can be compensated by increasing the number of solar photonic assists. The increase in time spent at distances below 5 AU is for a large part compensated by the increased speed at solar sail jettison. For characteristic accelerations below 1.0 mm/s², it is not possible to travel to the heliopause in less than 27 years without a significantly lower minimum solar distance and the use of a Jupiter Gravity Assist. Both are considerable complications for the mission concept. The former because of a significantly more demanding thermal design (more than factor of two increase in solar flux), the latter because of launch window constraints (Jupiter orbits in 11.8 years around the Sun).

For the system design (discussed in section 3.4), a solar sail characteristic acceleration of 1.0 mm/s² has been assumed. A characteristic acceleration of 3 – 4 mm/s², such as proposed in [Wallace00, Leidpold06], would obviously yield a faster transfer time (down to 15 years), but as shown in the next sections, such characteristic accelerations either require a much larger sail or a longer technology development horizon (reducing spacecraft or sail system mass). Hence a 27 year transfer, based on less demanding but nevertheless challenging technology has been selected as the baseline for the IHP mission concept study.

3.4 *Launch and transfer*

A Soyuz-Fregat 2-1B launch from Kourou has been selected as the baseline for the Interstellar Heliopause Probe TRS because it is a cost-efficient and highly reliable launch vehicle. The Soyuz-Fregat 2-1B has an Earth escape mass capability of around 2,000 kg (for $C_\infty = 0$ km/s, declination between -30° and $+30^\circ$). Since the ‘nose’ of the heliopause is at a fixed location in ecliptic coordinates, the launch window for an optimized trajectory practically repeats each year.

¹ After Jupiter Gravity Assist (JGA)

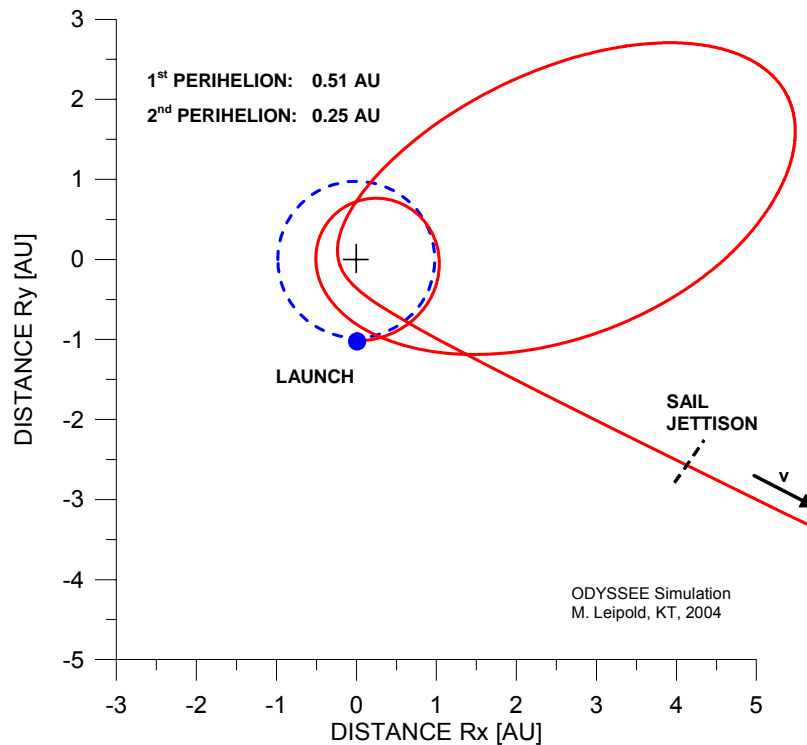


Figure 3-2: Baseline trajectory for the IHP TRS, $a_c = 1.00 \text{ mm/s}^2$ using two solar photonic assists.

The IHP TRS 1.0 mm/s^2 baseline trajectory is depicted in Figure 3-2 and Figure 3-3. The solar sail will be deployed at 50 to 65 hours after the Soyuz-Fregat has brought the spacecraft into an Earth escape trajectory (the *geocentric escape phase*). Subsequently, two *solar photonic assists* are carried out in order to gain ΔV before proceeding to the outer solar system. The characteristics of the first sailing phase are:

- 1st aphelion: 1.05 AU
- 1st perihelion: 0.51 AU
- 2nd aphelion: 5.76 AU
- 2nd perihelion: 0.25 AU

At about 5 AU the sail will be jettisoned. Science operations commence only after the sail has been jettisoned so that the presence of the sail does not have any influence on the science measurements (e.g. due to sail shadowing or charging effects).

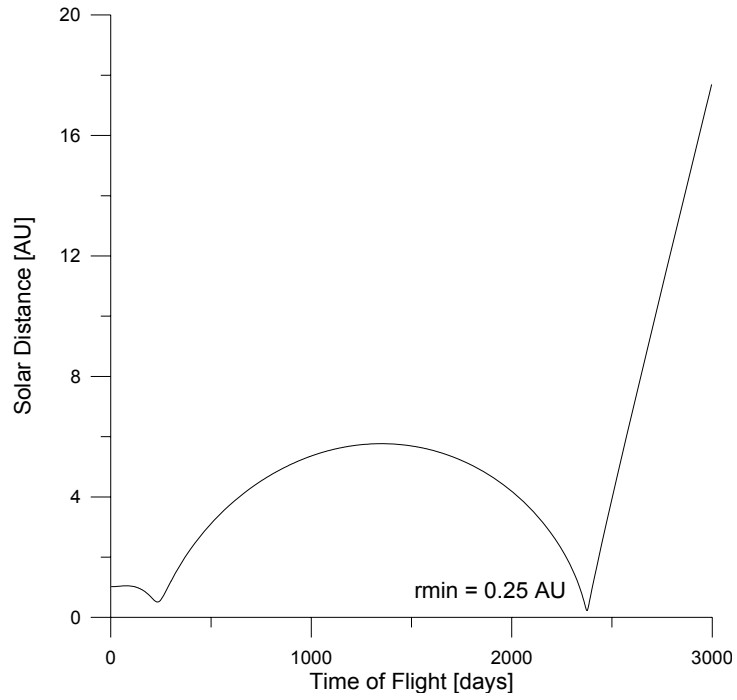


Figure 3-3: Heliocentric distance evolution for the IHP TRS baseline trajectory.

4 MISSION CONCEPT DESIGN

4.1 Margins

During the spacecraft design study, the margins listed in Table 4-1 have been used. For the solar sail propulsion system, a performance margin of 5% has been assumed. This margin is added to account for possible deviations from the ideal trajectory. The nominal mass and power budgets are determined after application of the subsystem margins. The power subsystem is designed and sized to provide the spacecraft required power, including system level power margin. The margins comply with the ESA margin philosophy for assessment studies [Atzei05].

The solar sail propulsion subsystem is sized to accommodate the mass budget after application of the system level margin, i.e. the solar sail is sized to provide 1.05 mm/s^2 characteristic acceleration if the system level margin is completely used (both for the platform and the sail subsystem).

Table 4-1: Margin overview.

Item	Margin
<i>Solar sail propulsion</i>	
Solar sail performance margin	5%
<i>Subsystem mass margin</i>	
Off-the-shelf equipment	5%
Off-the-shelf equipment requiring minor modifications	10%
New designs/major modifications	20%
<i>Power subsystem margin</i>	
Off-the-shelf equipment	5%
Off-the-shelf equipment requiring minor modifications	10%
New designs/major modifications	20%
<i>Data processing</i>	
On-board memory capacity margin	50%
Processing peak capacity margin	50%
<i>Communications</i>	
Communication link	3 dB
Telecommand and telemetry data rates	3 dB
<i>System level</i>	
System level mass margin	at least 20%
System level power margin	at least 20%

4.2 System overview

The IHP spacecraft can roughly be divided into two subsystems based on the two main operational phases: A solar sail propulsion system and a spacecraft platform. During the solar sailing phase (the first 6.7 years), both systems are coupled together. At 5 AU, the solar sail propulsion system is jettisoned and the platform starts the science investigations while travelling to the outer heliosphere and the heliopause (science phase).

Table 4-2 summarizes the overall mass budget. The launch mass is well below the launch vehicle capability. The driving factor, however, for the IHP spacecraft design is the solar sail size. For efficient solar sail propulsion, the in-flight spacecraft mass needs to be as low as possible to keep the sail size within reasonable limits. To obtain a characteristic acceleration of 1.0 mm/s^2 for a 467 kg in-flight spacecraft mass, a solar sail with an area of $\sim 60,000 \text{ m}^2$ is required. Any mass growth, both in the platform and the sail propulsion system, will result in a corresponding increase in sail size and complexity. Of course, also the opposite holds, i.e. any decrease in the sail system (e.g. booms, sail density) or platform mass, results in a reduction of sail size and complexity.

Figure 4-1 shows the spacecraft configuration during the various mission phases. In the following sections the design of the solar sail propulsion system, the platform and the reference payload instrument suite will be discussed in more detail.

Table 4-2: Spacecraft mass budget summary.

Subsystem		Mass (kg)	Remarks
Science payload		21	Highly integrated plasma P/L suite
Communications		20	Ka-band
Attitude control system (platform)		22	Thrusters, gyros, star trackers
On-board data handling		9	Leon 2 processor, 2 Gbyte SSM, SpaceWire interface
Harness		12	
Power		49	240 W RTG (7 W/kg), power conditioning unit
Thermal		13	MLI, radiators and heat pipes
Structure		16	
Nominal platform mass		162	
<i>Solar sail</i>	<i>Mass (kg)</i>		
- Booms	70		4 booms (100 g/m), each 174 m long
- Sail film	124		$246 \times 246 \text{ m}^2$ with a density of 1.9 g/m^2
- ACS	20		20 m gimbal control boom
- Deployment module (in-flight)	13		
<i>Subtotal</i>	<i>227</i>		
Sail and platform system (in-flight)		389	
<i>Sail and platform system including 20% system margin</i>		<i>467</i>	<i>For solar sail sizing (see section 4.1)</i>
Deployment module (jettisoned)		42	Total mass 55 kg. 42 kg jettisoned after deployment.
Launch mass		431	
20% system margin		86	
Launch vehicle adapter		45	
Total launch mass		562	Including system level margin
Launch vehicle capacity		~ 2000	Soyuz-Fregat performance into $C_\infty = 0 \text{ km/s}$, declination between -30° and $+30^\circ$

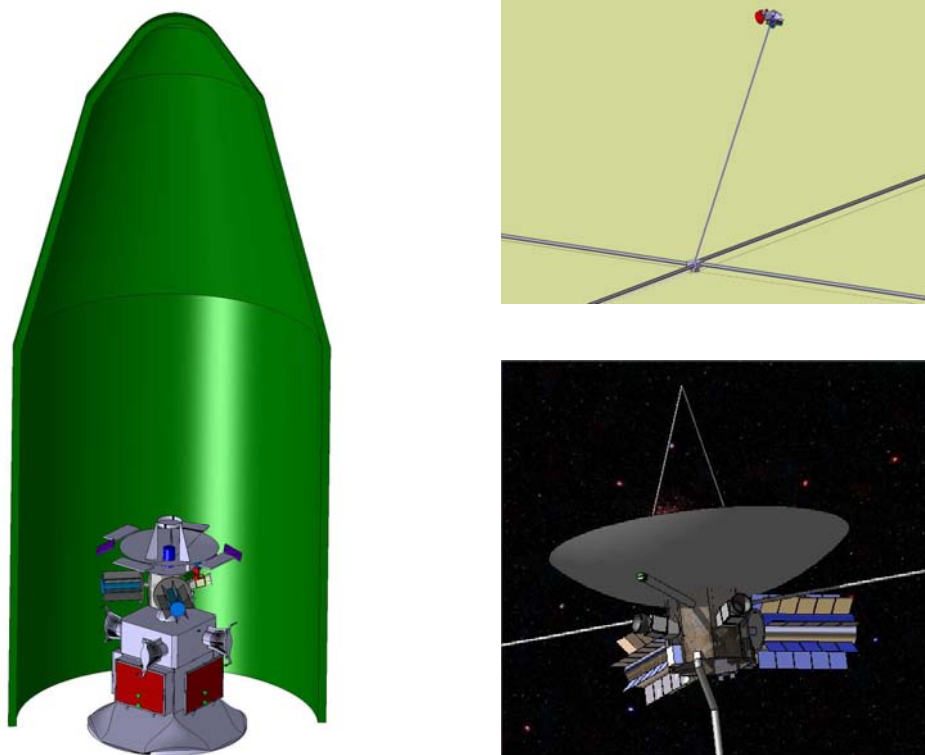


Figure 4-1: IHP spacecraft in Soyuz-Fregat S-T fairing (left), during sailing mode (top right), and during science mode (bottom right). The last picture is an earlier design with a larger antenna.

4.3 *Solar sail propulsion system design*

Many different solar sail design concepts have been studied and proposed (square sail with booms, circular sail with booms, inflatable circular sail, inflatable petal sail, circular sail without booms, and a heliogyro concept). These design concepts can in general be classified in two categories: three-axis stabilized sail concepts and rotating sail designs. Rotating solar sail designs have been discarded for IHP because of the huge moment of inertia of large solar sail systems. This impacts not only the propellant mass budget (for spin-up), but also the attitude control system (due to gyroscopic stiffness). The IHP solar sail design concept uses low-weight stiff booms for deployment and rigidization, primarily because of the availability and relative maturity of this technology.

4.3.1 SAIL FILM MATERIAL

The sail film mass is one of the dominant parameters in sailcraft design, especially when a large solar sail (>100 m) is manufactured. Currently, 7.5 μm Kapton films can be readily manufactured and have been used for the DLR solar sail deployment demonstration [Leipold03]. For the IHP mission concept, a much thinner material is required. The most suitable candidate is the Polyimide

thin film material CP-1 that has been developed by NASA Langley Research Center and is manufactured by SRS Technologies (see also <http://www.stg.srs.com/atd/advpolymers.htm>) [Talley02]. The CP-1 polyimide film has good thermal characteristics ($\alpha/\varepsilon = 0.1$) compared to the more conventional Kapton film ($\alpha/\varepsilon = 0.7$). When aluminized, CP-1 films have a surface reflectivity of up to 90% across the solar spectrum. The material has a bulk density of 1.43 g/cm^3 , and the current minimum thickness is $1.5 \text{ }\mu\text{m}$. For the IHP TRS, it is assumed that in the future, the thickness of this material for use in solar sail applications can be further reduced to $1 \text{ }\mu\text{m}$.

The front side of the film will be coated with 80 nm Al and an additional 30 nm SiO_x to reduce degradation effects, resulting in a calculated reflectivity of 0.85. The thickness is a compromise between optical density and adhesion properties. The backside of the sail is covered with a highly emissive 30 nm Cr coating ($\varepsilon = 0.7$), which is needed to emit the solar radiation during the close solar approaches. The total area density of the film, including coatings, is 1.9 g/m^2 .

To prevent differential charging across the non-conductive polyimide film due to photoelectron emission on the sun-illuminated front side, it is proposed to short-circuit both sides of the film at regular intervals.

4.3.2 SAIL DEGRADATION EFFECTS

4.3.2.1 *Micro-meteoroid impact*

Micrometeoroid impacts on the solar sail could lead to degradation or even total failure of the sail. In the past, tests have been carried out at the Technical University of Munich to assess this potential issue. Glass bullets between 6 and $88 \text{ }\mu\text{m}$ were accelerated in a vacuum chamber for impact on pieces of sail film. These sail samples measured $60 \text{ mm} \times 90 \text{ mm}$ and had thicknesses of $7.5 \text{ }\mu\text{m}$ (Kapton) and $4.0 \text{ }\mu\text{m}$ (Poly Ethylene Naphtalethe). The samples were subjected to an appropriate level of stress in order to simulate the sail tension. The impact velocity had a range between 6.0 km/s and 12.2 km/s . The projectiles created a hole and a melting zone around it with a diameter of typically $98 \text{ }\mu\text{m}$. In addition, ejecta material was detected around the impact crater and primarily in a radial direction. However, in none of the cases, tear was initiated in the film regardless of the film material or membrane stress that was used. This indicates that the melting zone around the hole acts in a 'self-healing' way. However, in these tests, only perpendicular impacts on the samples were simulated. Future testing with oblique impacts should be done before concluding on the long-term survivability of the sail in the space environment.

4.3.2.2 *Degradation due to radiation environment*

Limited tests have been performed on radiation, but reference is made to tests performed in the DLR vacuum room facility (KOBÉ) which allows simultaneous exposure to protons, electrons, and vacuum ultraviolet [Lura01]. It is concluded that absorptivity and emissivity of the sail film increases after irradiation. Also the tear strength of the sail is significantly reduced with typically 30% after an exposure to 170 krad .

4.3.2.3 Sail efficiency

From the previous sections, it is clear that the solar sail properties will be affected by the space environment. It is recommended that any future solar sail film developments are accompanied by extensive testing in representative environments, including folding tests, to ensure that the solar sail performance can be predicted with a high level of confidence. For this system design study, an overall degradation in the sail reflectivity of 7% has been assumed as compared to the calculated properties, resulting in a solar sail efficiency η of 0.89.

4.3.3 BOOM TECHNOLOGY

The DLR Carbon Fibre Reinforced Plastics (CFRP) booms are selected for the IHP mission concept. They combine a high strength and stiffness with a low density and low packaging volume. The booms consist of two laminated sheets which are bonded at the edges to form a tubular shape (see Figure 4-2). During storage, the two sheets are pressed flat around a central hub. When this coil is unwound, the boom regains its original tubular shape with high bending stiffness. The resistance against high thermal loads can be improved by adding a Kapton film on the outer surface.

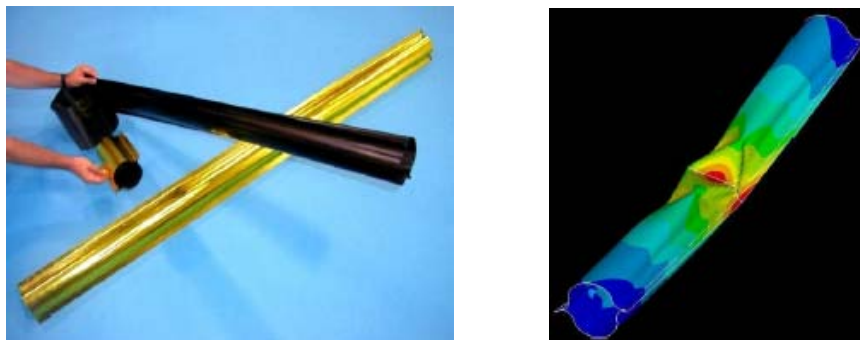


Figure 4-2: Deployable CFRP booms of DLR, with local buckling analysis.

4.3.4 SOLAR SAIL SIZING

As shown in Table 4-3, a square sail size of $246 \times 246 \text{ m}^2$ is required to achieve a design characteristic acceleration of 1.05 mm/s^2 for an in-flight spacecraft mass of 467 kg (including 20% system margin). If the system level margin would not be used during the sailcraft design and development, the sail size can be reduced to $224 \times 224 \text{ m}^2$.

Table 4-3: Sail sizing.

Sail sizing parameters	Value	Remarks
Characteristic acceleration (mm/s^2)	1.0	Assumed for trajectory design
Design characteristic acceleration (mm/s^2)	1.05	5% performance margin (see also section 4.1)
Sail efficiency η	0.89	Ref. section 3.2
Sail and platform mass (kg) (including 20% system margin)	467	Ref. Table 4-2
Required sail size (m^2)	246 × 246	Square

4.3.5 STORAGE AND DEPLOYMENT

For compact stowage, the solar sails are folded using *frog-leg folding*, since this method has been successfully applied in the ESA/DLR solar sail breadboard model [Leipold03]. In *frog-leg folding* the film is first folded to form a long strip, and then the strip is accordion folded from both sides towards the centre of the package. The advantage over alternative methods is that the folding lines do not interfere with each other, and it is quite compact when folded. Figure 4-3 shows the sail container design. Also the sail container is based on the previously built ESA/DLR solar sail ground demonstration. The four containers each have a size of $78 \times 57 \times 18 \text{ cm}^3$. The sail containers will be jettisoned once the sails are unfolded.

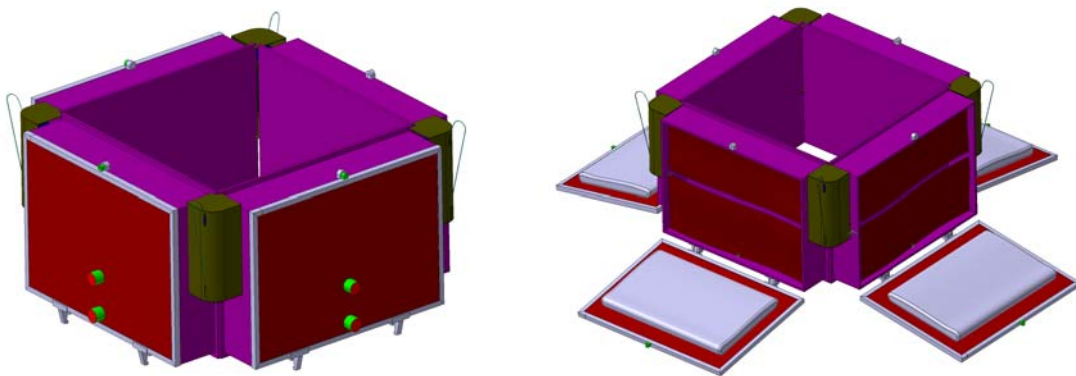


Figure 4-3: Solar sail container for the IHP TRS in closed and open configuration.

4.3.6 ATTITUDE CONTROL

The large inertia and mass of most solar sailing mission concepts, makes the solar sail attitude control systems very complex and challenging [Wie04, Mangus04, Lappas05]. Conventional attitude control systems (ACS) such as thrusters, reaction wheels and gyros are difficult to implement or require significant propellant mass. Several options for the IHP solar sail attitude control system have been considered, including *tip vanes*, *control mast/thrust vector control*, *quadrant tilt translation*, *boom tip thrusters*, *ballast trim* and *quadrant tilt*. The different solar sail attitude control options have been traded, and the key results are shown in Table 4-4.

Tip vanes and quadrant tilt schemes have been discarded because of mechanical complexity. Tip-mounted thrusters are considered too difficult to implement, and the trim control system requires an additional mass ballast. Therefore a Thrust Vector Control (TVC) scheme using a 2D-gimballed boom, depicted in Figure 4-4, has been baselined for IHP sail attitude control system. In a TVC system, the attitude of the solar sail with respect to the Sun is controlled by creating an offset between the centre of mass (determined by the control boom azimuth and elevation) and the solar radiation pressure. The ensuing torque is used to control the pitch and yaw angle.

Table 4-4: Trade-off between the different solar sail attitude control options.

Control Schemes	Control axis	Advantages	Disadvantages
Tip vanes	3-axis	Tested in orbit on GEO satellites	Heavy, complex, complex to stow, power hungry, non-redundant
Quadrant tilt	Roll	Easy to use	Need to combine with other schemes; mechanically complex
Quadrant tilt translation	3-axis	Use torque arms	No heritage, non-redundant, heavy
Sailcraft control mast	2-axis (pitch / yaw)	All actuators are located centrally; no mechanisms at boom tips, no feed lines along boom	No roll control. If necessary, this needs to be provided by quadrant tilt or thrusters. Single point failure possibility
Trim control mass	2-axis (pitch / yaw)	High agility	Complex mechanisms, failures of tethers and tether tensioning, needs to be compatible with boom deployment and boom design, additional mass
Boom Tip Thrusters	3-axis	Use of large torque arms, needs roll control	Wireless RF/power needed or local power supply at tips; propellant storage at boom tips; both result in increase of moments of inertia and of requirements on boom strength and stiffness

The two most important hardware components for TVC are the central sailcraft control mast and a 2DOF gimbal mechanism. The boom, based on a CoilABLE boom from ABLE Engineering (<http://www.aec-able.com/Booms/coilboom.html>), is 20 m long and its weight is 5 kg (the stowage canister weighs another 5 kg). The total mass for the gimbal mechanism is estimated to be 10 kg. The required control angle and maximum angular rate are, respectively $\pm 30^\circ$ and 1.5°/s.

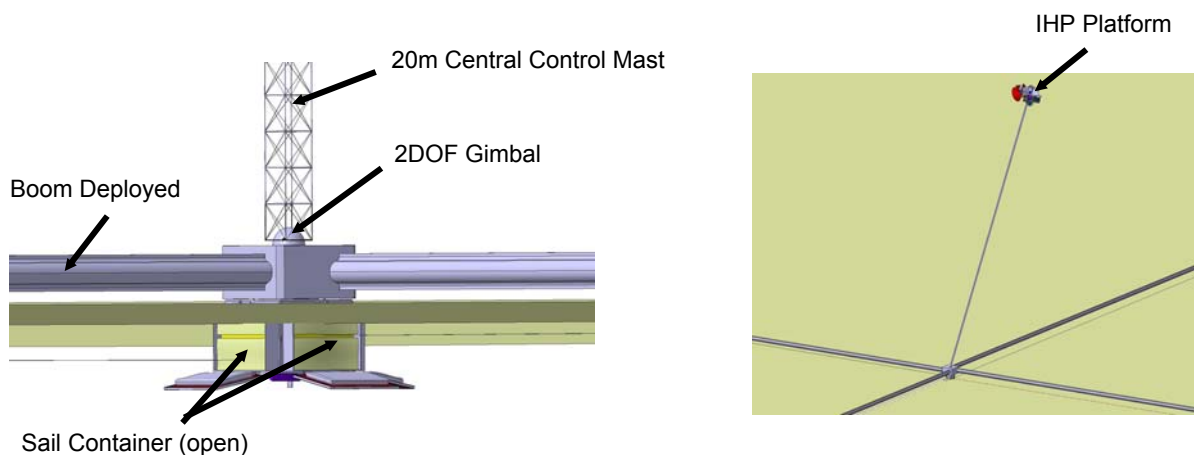


Figure 4-4: IHP thrust vector control system by a gimbaled boom.

An ESA internal assessment study has shown that for IHP, it is best to leave the roll axis uncontrolled: Active control, such as quadrant tilt of thrusters increase the sailcraft mass and complexity, while passive control by spinning would reduce the control authority due to gyroscopic stiffness. If left uncontrolled, the roll-axis of the sail will slowly oscillate with a period of around 20 hrs. This natural roll dynamics is caused by a tangential component of the solar pressure force for a solar sail reflectivity less than 1. Sail degradation (uniform and non-uniform) and billowing will very likely have important consequences for the natural roll dynamics, though it is not expected to impact the control strategy.

4.3.7 SOLAR SAIL PROPULSION SYSTEM – SUMMARY

The solar sail assembly mass budget is detailed in Table 4-5. A structural sizing of the booms has been carried out where the largest force considered is the sail tensioning that keeps the sail membrane stressed. For a sail tensioning stress of 1 psi, the boom radius is 120 mm, yielding a boom specific mass of 100 g/m (for a safety factor of 2). The first natural frequency of the deployed sail assembly has been determined to be 6.5 mHz.

Table 4-5: Sail assembly mass budget.

Item	Mass (kg)	Remarks
Boom	69.6	4 booms with length 174 m, with specific mass 100 g/m
Boom tip covers and reels	0.4	
Boom total mass	70.0	
Sail film	112.5	1.86 g/m ² (1 μm CP1 with 80 nm Al, 30 nm SiO _x , 30 nm Cr)
Bondings	5.6	
Sail edge reinforcements	3.3	
Sail tensioning lines	2.0	
Sail film total mass	123.5	
Boom deployment mechanisms	25.0	
Sail containers	20.0	
Sail deployment mechanisms	10.0	
Deployment module	55.0	
Sail assembly launch mass	248.5	
Release of deployment module	-42.0	After solar sail deployment, part of the deployment module is jettisoned.
Total in-flight mass	206.5	Excluding system level margin

The sail area consists of four separate sails, which are overlapping at the edges to thermally shield the spacecraft, which is of particular relevance during the solar photonic assists. Figure 4-5 shows the inner sail corners, with the overlapping sails, that are slightly offset in axial direction, as well as the accommodation of the low gain antennas on the sail hub. The spacecraft itself is on a long control boom behind the sail (not shown in this picture). Because the primary scientific investigations are commenced after jettison of the sail, the solar sail assembly and the solar sail attitude control mechanisms have no impact on the science payload. To monitor the spacecraft radiation dose, it can be considered to turn on the energetic particle detector during the sailing phase. Because of the high energy of the particles measured, this should be no problem.

The manufacturing, verification and folding of such a large solar sail will be very challenging [Talley02]. It is expected that possible demonstration and pre-cursor missions would allow building up experience in this process.

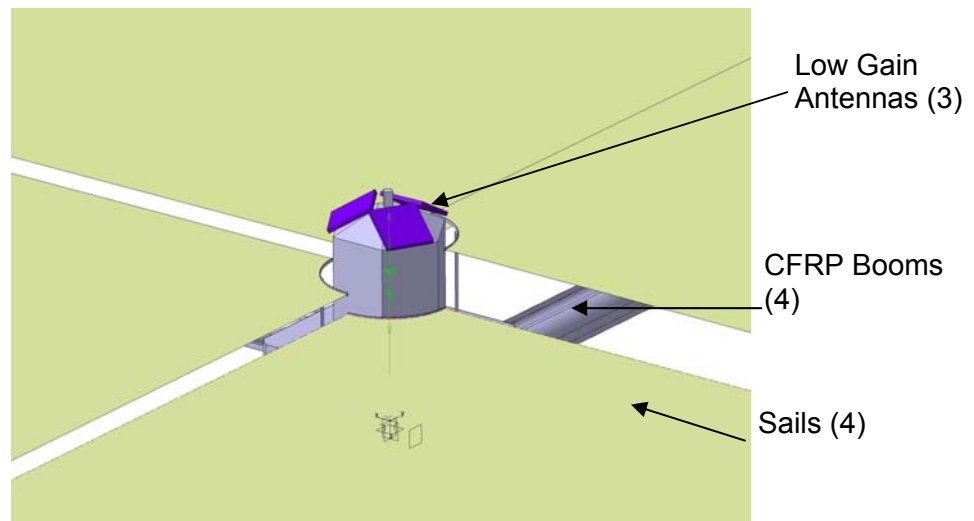


Figure 4-5: Inner sail corner regions with central hub and low gain antennas.

4.4 Platform design

4.4.1 MECHANICAL CONFIGURATION AND STRUCTURE

The mechanical configuration at launch is depicted in Figure 4-6. The spacecraft stack height, consisting of sail containers, boom deployment system, spacecraft platform and high gain antenna is 2.2 m and the maximum diameter approximately 2 meters. The spacecraft platform structure is octagonal shaped and measures only 0.5 m (width) by 0.5 m (height). The platform main structure consists of an aluminium honeycomb structure with skin thickness of 3 mm and overall thickness of 20 mm. The key driver for the structure mass and volume are the two radio-isotope thermoelectric generators (RTGs).

Figure 4-7 provides a graphical overview of the platform architecture. For the accommodation of the subsystems particular consideration has been given to balancing the moments of inertia and thermal design aspects. Apart from the magnetometer and the wire boom antenna, all instruments are accommodated on a 1 m instrument boom, to ensure a good field-of-view as well as to maximize the distance from the RTGs. Another driver is the FOV for the star trackers.

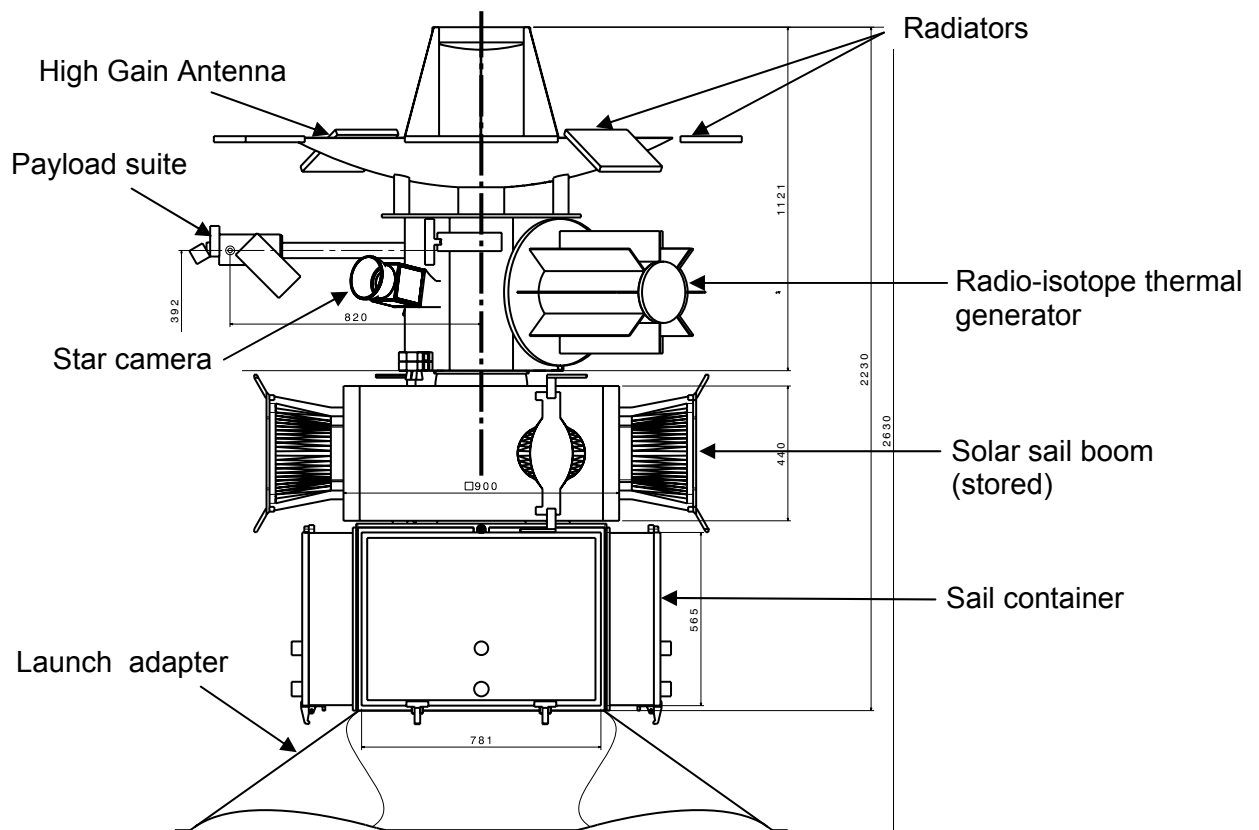


Figure 4-6: Mechanical configuration and dimensions of the IHP spacecraft.

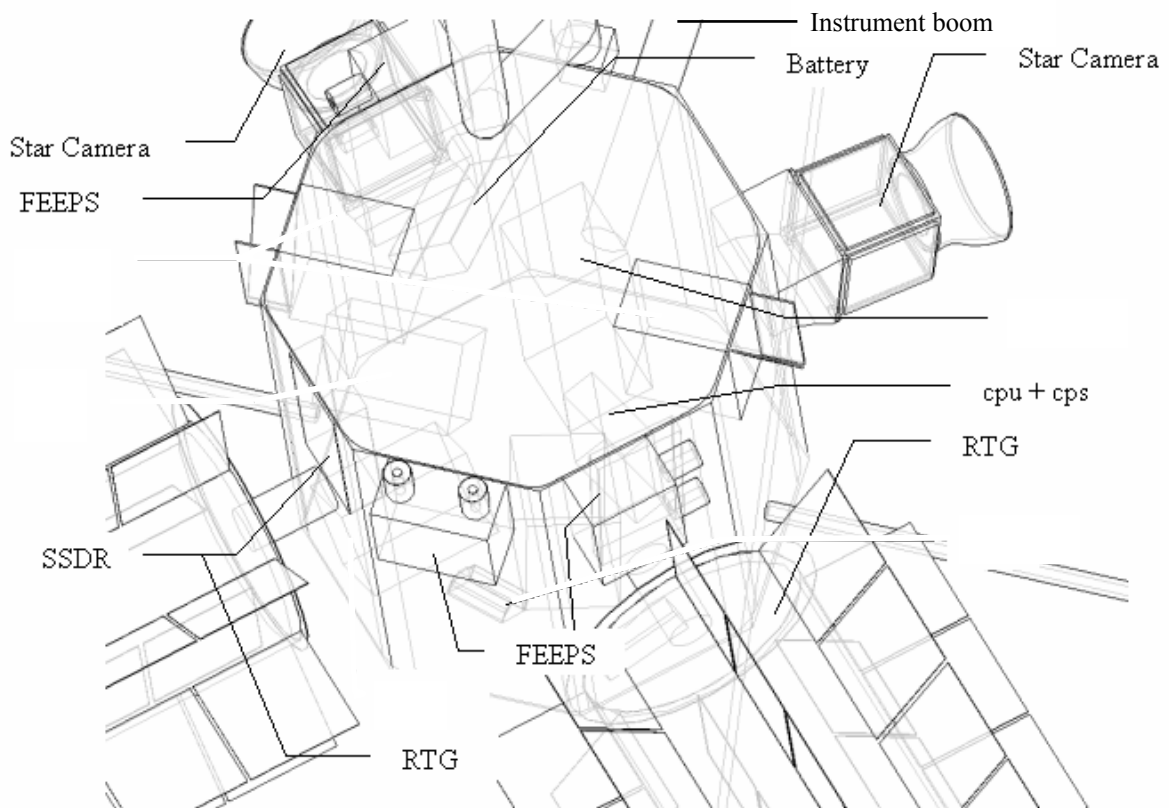


Figure 4-7: IHP platform architecture (without High Gain Antenna).
 (FEEPS = micro-thruster, SSDR = Solid state data recoder)

4.4.2 ATTITUDE DETERMINATION AND CONTROL

Attitude determination during both the sailing phase and science phase is achieved by two star trackers in combination with MEMS gyros.

The attitude control during the sailing phase has already been discussed in section 4.3.6. During the science operational phase, after jettison of the sail, the platform baseline configuration is a spinning mode with a spin rate of 1 to 3 rpm. This provides a robust and stable attitude, while allowing a 4π field of view for the payload (per spin). The spin axis is oriented towards the Sun, so that the high gain antenna, along the spin axis, is always directed towards Earth. The spin axis pointing accuracy (0.5°) and stability (10^{-3} °/s) are dictated by the communication link and science payload.

For attitude and spin maintenance (during the science operations phase), six Field Emission Electric Propulsion (FEEP) thrusters are baselined [Marcuccio97,Tajmar04], mainly because of their high specific impulse ($I_{sp} \sim 5,000$ s) thus requiring less propellant mass than for e.g. hydrazine

thrusters. Indium FEEP thrusters produce thrust by exhausting a beam of field evaporated singly ionized Indium atoms. FEEPs have a high electrical efficiency ($> 95\%$), but they have a small thrust ($\sim 20 \mu\text{N}$). Considering that the attitude disturbances are very small due to the large distance from the Sun and other planetary bodies, the small thrust is not considered a problem. The main disadvantage of the FEEP thrusters is the limited flight heritage and the limited lifetime tests, but both are expected to improve in the near future. Additionally, possible contamination of the payload by the thruster plume will need to be assessed.

4.4.3 ON-BOARD DATA HANDLING

The On-Board Data Handling subsystem (OBDH) is based on a SpaceWire architecture. It consists of 2 LEON core processors and 2 solid state memory boards each with a capacity of 2 Gbyte. The OBDH system provides S/C health monitoring, data storage, communication and telecommand support and controls the ADCS system during the sailing and the science mode. The mission lifetime (25 years) requires the system to have a large reliability, i.e. components that are fault-tolerant towards single event upsets and a dual redundant system.

The downlink data rate is 1 kbps during the sailing mode (< 7 AU from the Earth), and 200 bps during science mode (up to 200 AU from the Earth). The typical storage requirement is 2 Mbyte for 24 hours, yielding a down-link data storage requirement of less than 16 Mbytes for weekly communication during the science mode.

4.4.4 COMMUNICATION

Because the IHP mission concept needs communication up to a distance of 200 AU from the Earth, the TM/TC subsystem is considered one of the main challenges of the mission concept design. At this distance, the system should allow a downlink data rate of 200 bps, and an uplink data rate of 5 bps. In sailing mode (distances below 5 AU), the downlink requirement is ~ 1 kbps. Furthermore, the resource allocations (mass, power and volume) for the communication subsystem shall be minimized, in particular the antenna diameter shall be compatible with the payload FOV requirements. Additionally, the communication subsystem should not require any active S/C pointing.

Figure 4-8 shows a functional block diagram of the communication subsystem for the IHP TRS that fulfils above requirements. The block diagram shows only one of five low gain antennas distributed over the spacecraft bus and solar sail box to achieve omni-directional coverage. The High Gain Antenna is a light-weight and highly stable fiberglass Cassegrain 1.25 m diameter antenna dish, with a steerable radiation beam ($\pm 2.5^\circ$) by means of a piezo-controlled deformable secondary reflector.

At distances below 5 AU, during the solar sailing phase, 5 Low Gain Antennas (LGAs) are used to achieve omni-directional coverage. For medium distance communication (5-50 AU), a piezo-controlled defocusing mechanism is used to widen the antenna beam of the High Gain Antenna (HGA) to about 15° . At longer distances the HGA is always pointed towards the Sun and thus also directed towards the Earth. An antenna steering mechanism ensures that the Earth orbit is within the antenna beam width of 1.6° .

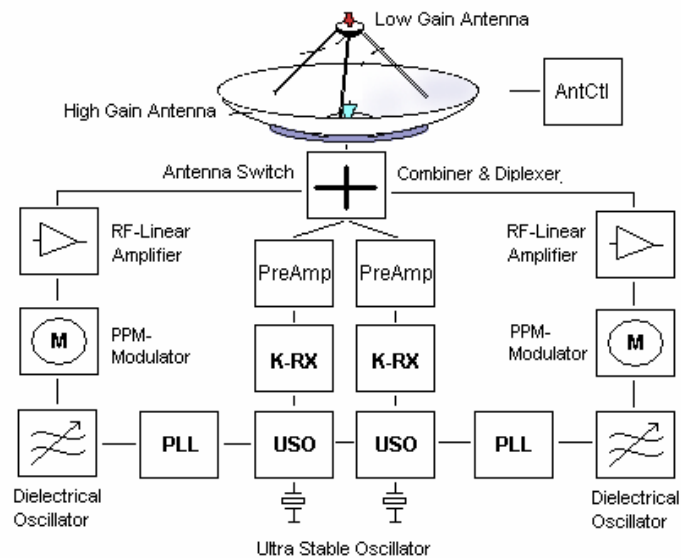


Figure 4-8: IHP TM/TC subsystem block diagram

The TM/TC system uses the Ka-band (29 GHz) as the baseline frequency. In order to limit the RF power requirements, the communication link is based on Impulse Radio: The signal is transmitted as carrier-free short duration pulses ($< \mu\text{s}$) that are time-synchronized by an on-board ultra-stable oscillator. The very low duty cycle (on the order 1/100, 1/1000 or less) reduces the input power and concentrates all the RF power on a single impulse. Information is transmitted by On/Off Keying (Pulse-Position Modulation). The pulse position modulation technology is often used in low-power industrial RF applications that require operation with single battery for up to several years (e.g. remote keys and chip-cards). For high power deep space communications, this modulation has not been used, but the technology for producing short high power pulses is applied in radar systems.

A summary of the link budget at 200 AU distance is provided in Table 4-6. By using a larger ground station, the down-link rate could be further increased. The transmit power of 1000 W is only used during the very short pulse lengths. The average RF power is approximately 10 W (losses included). At shorter distances the achievable data rate will be higher by reducing the peak power, while increasing the duty cycle (e.g. in steps of factor 2). The total mass for a dual-redundant communication subsystem is 20 kg (including subsystem margin). The nominal power when transmitting is 34 W.

The ground station visibility is determined by the Earth's rotation around the Sun and the Earth's rotation around its axis. Apart from a few days communication outage during solar conjunction, a single ground station should be able to access the IHP spacecraft for approximately 10 hrs per day. However, due to the very long operational lifetime, this would become a significant cost driver. By reducing the science measurement duty-cycle with e.g. a factor 100, the ground station usage can nominally be reduced to approximately 4 hours per week. When crossing interesting regions and boundaries, such as e.g. the termination shock, ground station usage can obviously be increased.

Table 4-6: Link budget summary for downlink at 200 AU.

Operating frequency	
Frequency	29 GHz
Wavelength	0.01 m
Transmitter	
Antenna diameter	1.25 m
Internal losses	-2 dB
Transmit power	1000 W
Transmit power	60 dBm
Antenna gain	48.6 dB
Receiver	
Antenna diameter	17.5 m
Atmospheric losses	-3 dB
Internal losses	-3 dB
Antenna gain	71.5 dB
Free space loss	
Range	200 AU
Free space loss	-331.2 dB
Result	-159.1 dBm

In order to monitor the S/C health status at more regular intervals (in between data-downlink windows), a beacon mode operation can be implemented. The spacecraft then sends out a radio-tone to indicate how urgently a ground station contact is required (e.g. four different frequencies). These tones are easily and quickly detected with low cost receivers and smaller antennas.

The RF communication design requires development of new component technologies, especially the high power transmitter RF pulse technology. Many of the RF communication components are not yet available as space qualified items. Additionally, deep space ground stations will need to be adapted to receive and demodulate the pulse-position-modulation signal.

4.4.5 POWER

As detailed in Table 4-7, the spacecraft requires 159.3 W average design power and 184.4 W peak power. Due to the large distances from the Sun (up to 200 AU), solar generated power is not an option for IHP. The only option with flight heritage is a radio-isotope power source. Radio-isotope power sources produce electrical power by converting heat or energetic particles of radioactive material into electrical power. The best-known radio-isotope power systems are radio-isotope thermoelectric generators (RTGs), which use the thermoelectric effect to convert heat into electricity.

Though there is no European flight heritage, NASA has flown more than twelve space exploration missions using one or more RTGs. They are well suited for use in space exploration, as they are rugged, radiation tolerant, and can be designed for a long lifetime (up to 100 years). Additionally, RTGs can be used as a heating unit at the same time. However, RTGs have drawbacks such as the

availability and cost of the radioisotope material, significant safety precautions during manufacturing, assembly and launch, as well as possible radiation interference with payload instruments. For an extensive overview on NASA's past and future space radio-isotope power systems, the reader is referred to [Schmidt05, Stofan06].

Due to the limited availability of the radioisotope material, NASA is currently focusing the technology development activities on reduction of the use of radio-isotopic material by developing more sophisticated energy converters, at the expense of system mass increase (from 5.5 W_e/kg on Cassini and New Horizons, towards 3 – 4 W_e/kg [Abelson05]). It is expected that in the longer term (after 2015), the specific power will increase again (8 – 10 W_e/kg) [Wong06].

For the IHP TRS, 2 RTG units, each with a specific power of 7 W_e/kg have been assumed. The RTG units provide 240 W beginning of life, and assuming a conservative degradation of 1.7% per year [Abelson05], this yields a nominal mission lifetime of 30 years. The RTGs will need to be qualified for this long lifetime. Including additional items such as structural support, a power conditioning unit as well as 20% subsystem maturity margin, the total mass budget for the power subsystem becomes 49 kg (see also Table 4-2).

Table 4-7: Summary of the power budget as function of spacecraft modes.

Subsystem		Spacecraft recovery	Sail deployment	Solar sailing	Science mode	Telecomm. mode
Payload	Average	16.0	0.0	3.0	16.0	0.0
	Peak	16.0	0.0	3.0	16.0	0.0
ADACS	Average	17.5	17.5	20.0	17.5	17.5
	Peak	25.0	25.0	25.0	25.0	25.0
TM/TC	Average	25.2	25.2	53.4	25.2	25.2
	Peak	27.7	7.9	57.0	27.7	27.7
OBDH	Average	15.0	15.0	15.0	15.0	15.0
	Peak	15.0	15.0	15.0	15.0	15.0
Power supply	Average	1.0	1.0	1.0	1.0	1.0
	Peak	1.0	1.0	1.0	1.0	1.0
Thermal control	Average	7.5	7.5	0.0	7.5	7.5
	Peak	9.0	9.0	0.0	9.0	9.0
Deployment actuators	Average	0.0	40.0	0.0	0.0	0.0
	Peak	0.0	65.0	0.0	0.0	0.0
Subtotal	Average	82.2	106.2	92.4	82.2	66.2
	Peak	93.6	122.9	101.0	93.6	77.7
Total (incl. margin)	Average	123.2	159.3	138.6	123.2	99.3
	Peak	140.4	184.4	151.5	140.4	116.6

4.4.6 THERMAL

The thermal design of the IHP spacecraft is driven by two extreme load cases. The first is during the solar photonic assists, when the spacecraft is in close proximity to the Sun (0.25 AU). The second is when the platform is far away from the Sun, at 200 AU.

4.4.6.1 Platform thermal design

The thermal design of the spacecraft (platform and sail) has to guarantee acceptable temperatures at all times. A purely passive mass-optimized thermal design is targeted, because mass and power are key system drivers (ref. section 4.2). The key features of the platform thermal design are summarized in Figure 4-9. The platform is covered with $\sim 5 \text{ m}^2$ high temperature Multi-layer-insulation (MLI), and has four radiators viewing cold space during the sailing phase. Six heat pipes ensure a good thermal conductivity from the platform to three of the radiators. The fourth radiator is directly coupled to the payload instrument boom. The RTG units, which each produce $\sim 2000 \text{ W}$ thermal power, are loosely coupled to the spacecraft bus. The heat flux input of the RTGs assist in keeping the platform warm at far distances (“cold phase”), while limiting the thermal load during the solar photonic assists.

Due to the internal dissipation of the platform in all operational modes, no active heating is required, with the exception of the payload instruments that require only moderate heating ($\sim 3 \text{ W}$) when at far distances from the sun.

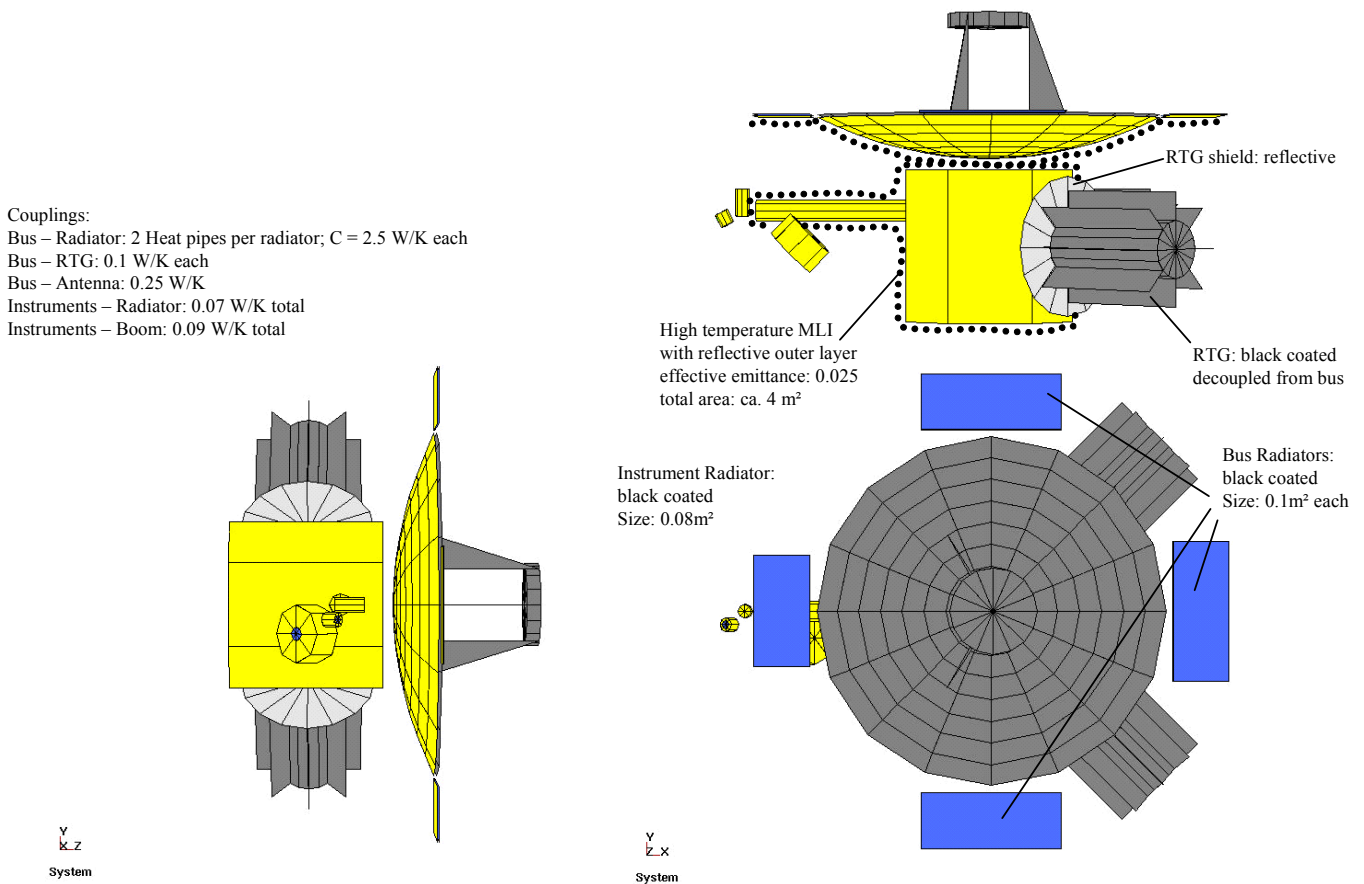


Figure 4-9: Key features of IHP thermal design.

4.4.6.2 Hot case thermal analysis

When the probe approaches the Sun at a distance of 0.25 AU, i.e. 54 solar radii, the solar flux equals 22.000 W/m^2 . During those solar photonic assists, the solar sail acts as a thermal shield for the platform. The highly reflective solar sail front side will reflect a significant fraction of the incoming solar flux. Figure 4-10 shows the thermal balance during the closest approach for a solar sail orientation of 35° , which is the optimal solar sail orientation for the solar photonic assist.

In this simple model, the solar sail temperature reaches a maximum of 240° C , which is only 20° C below the glass transition temperature for CP-1 (www.stg.srs.com/atd/advpolymers.htm). Whether this is sufficient margin for a non-ideal realistic sail (after possible degradation due to folding, deployment and exposure to the space environment) will need to be assessed in future dedicated studies and tests. The solar sail coating and film bonding material will need to be qualified to withstand the extreme temperatures as well.

Hot Case – Sailing Mode @ 0.25 AU

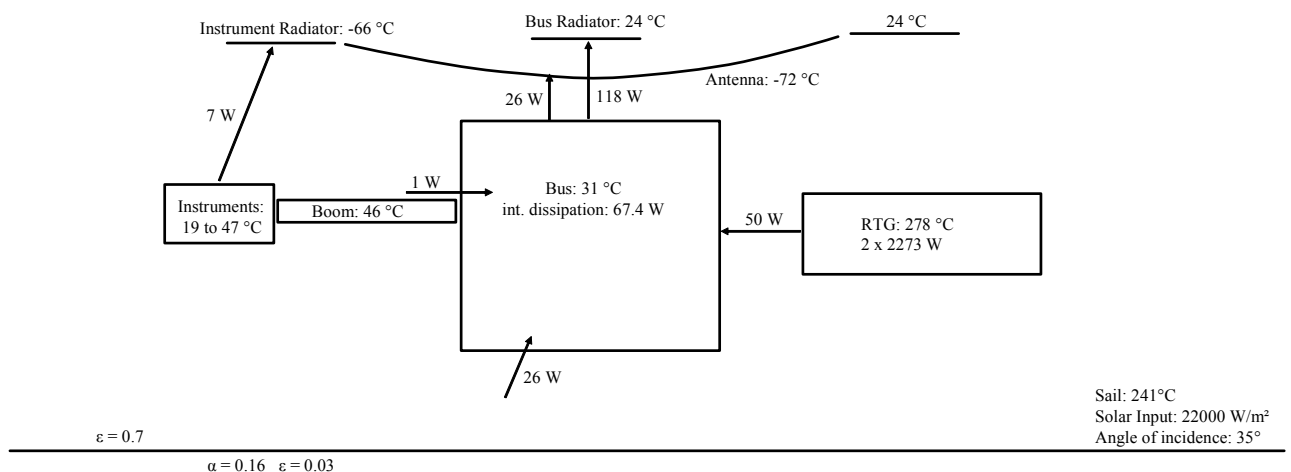


Figure 4-10: IHP hot case during sailing mode (at 0.25 AU perihelion).

4.4.6.3 Cold case thermal analysis

Figure 4-11 shows an overview of the thermal balance at 200 AU, when the IHP platform is exposed to cold space. The internal power dissipation of $\sim 90 \text{ W}$, in combination with the heat flux input from the two RTGs keeps the platform at $\sim 14^\circ \text{ C}$. To sustain an instrument temperature in excess of -10° C , a heating power of $\sim 3 \text{ W}$ is required.

Cold Case – Science Mode @ 200 AU

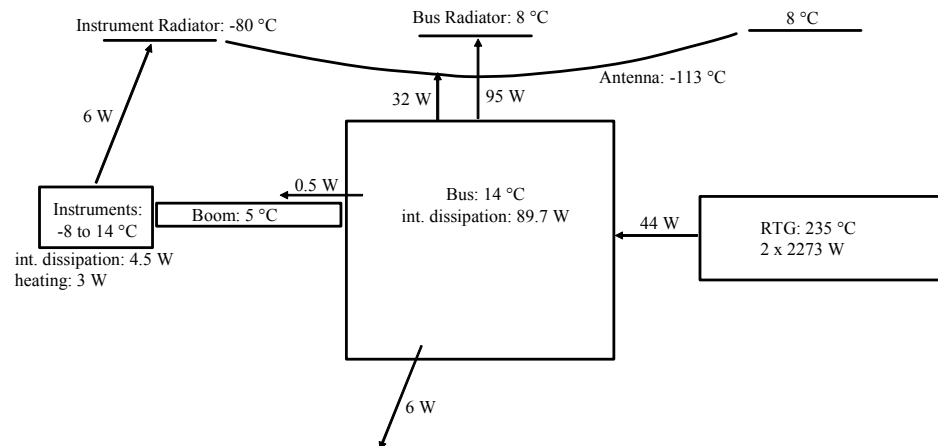


Figure 4-11: IHP cold case during science mode (at 200 AU).

5 REFERENCE PAYLOAD SUITE

The reference payload suite assumed for this study consists of a comprehensive set of typical instruments for the investigation of the outer Heliosphere, the Heliopause and the Local Interstellar Medium. As mass and power are key drivers for the IHP TRS, an assessment study has been carried out to minimize the payload resources [Kraft05]. The instruments, the key measurements and their allocated resource budgets are listed in Table 5-1. The total mass of the reference payload suite package is 21.4 kg including 20% maturity margin, with corresponding power requirement of 20.4 W. The reference payload suite consists of two electro-magnetic field analyzers, three particle analyzers as well as a dust analyzer and a UV-photometer. The instrument accommodation and field-of-views (FOV) are shown in Figure 5-1. The majority of the instruments are accommodated on a 1 metre boom to prevent shadowing effects of the high gain antenna.

The plasma and radio wave instrument consists of a single pair of ~35 m wire booms, which are deployed in the spin-plane. The highly sensitive magnetometer (the magnetic field is expected to go down to ~0.03 nT at the termination shock), consists of a pair of Flux-Gate Magnetometers on a 5 – 7 meter long boom.

The plasma particle analyzer is a conventional charged particle analyzer with integrated electron/ion optics. The ion analyzer part includes a time-of-flight analyzer to determine the isotopic abundances. The field of view is approximately $5^\circ \times 180^\circ$, so that a complete 3-D analysis can be performed per spin. The neutral and charged atom analyzer and imager is pointed in the direction of the heliospheric nose with a $15^\circ \times 15^\circ$ FOV. The analyzer does not uniquely identify isotopic abundance, charge and energy, but rather provides statistical distributions. Neutral and charged atoms are discriminated by applying a voltage across a collimating optics, the velocity

distribution and direction is determined by means of an ultrasonic chopper and a micro-channel plate (across half of the focal plane), while the energy distribution is determined by a solid state detector (across the other half of the focal plane). The energetic particle imager consists of 2 dual double-ended solid state particle telescopes with discrimination between electrons and protons. They each have two 60° FOV viewing cones.

The dust analyzer is an impact ionization detector combined with a Time-of-Flight mass spectrometer, which is able to measure the dust density and dust composition. It has a large FOV (~100°) directed towards the heliospheric nose. The UV-photometer measures the 121.6 nm Lyman- α line emitted by interstellar atomic hydrogen illuminated by the Sun. It has a very small FOV (~1°) perpendicular to the spin axis. During the operational lifetime, a complete 3-D intensity profile is obtained.

The science data rates are calculated assuming a very low duty cycle (< 0.1%) for all instruments, i.e. one measurement cycle per hour. This is not considered an issue for the science return, since the plasma scale lengths are quite large in comparison with the distance (~1 × 10⁻³ AU) travelled by the spacecraft in one hour.

Table 5-1: Baseline instruments for IHP

Instrument	Key measurements	Mass (kg)	Power (W)	Science data rate (bps)
Plasma particle analyzer (integrated electron/ion optics)	<ul style="list-style-type: none"> Elemental and isotopic ion abundance Plasma density and energy distribution High temporal distribution 	2	< 1.3	10
Plasma and radio wave instrument	<ul style="list-style-type: none"> Plasma wave dynamics (field and density fluctuations) Radio waves (at boundaries) 	4.5	4	10-200
Magnetometer	<ul style="list-style-type: none"> 3-D magnetic field Two magnetometers 	3.7	3.4	1-10
Neutral and charged atom analyzer and imager	<ul style="list-style-type: none"> Elemental/isotopic abundance Energy distribution Ionization of charged atoms 	0.5	1.8	16
Energetic particle imager: <ul style="list-style-type: none"> Suprathermal protons Anomalous and galactic cosmic rays 	<ul style="list-style-type: none"> Energy distribution: <ul style="list-style-type: none"> Electrons (20 keV to 400 MeV) Protons (20 keV to 300 MeV) Gamma-rays (up to 2 GeV) 	1.8	1.2	24
Dust analyzer	<ul style="list-style-type: none"> 10⁻²⁰ – 10⁻¹¹ kg Size, charge, velocity distribution 	1	1	8
UV-photometer	<ul style="list-style-type: none"> Ly-α hydrogen line 	0.3	0.3	1
DPU and Centralized Power Supply		2	4	
Structure		2		
Subtotal		17.8	17.0	
Margin (20%)		3.6	3.4	
Total		21.4	20.4	70 - 270

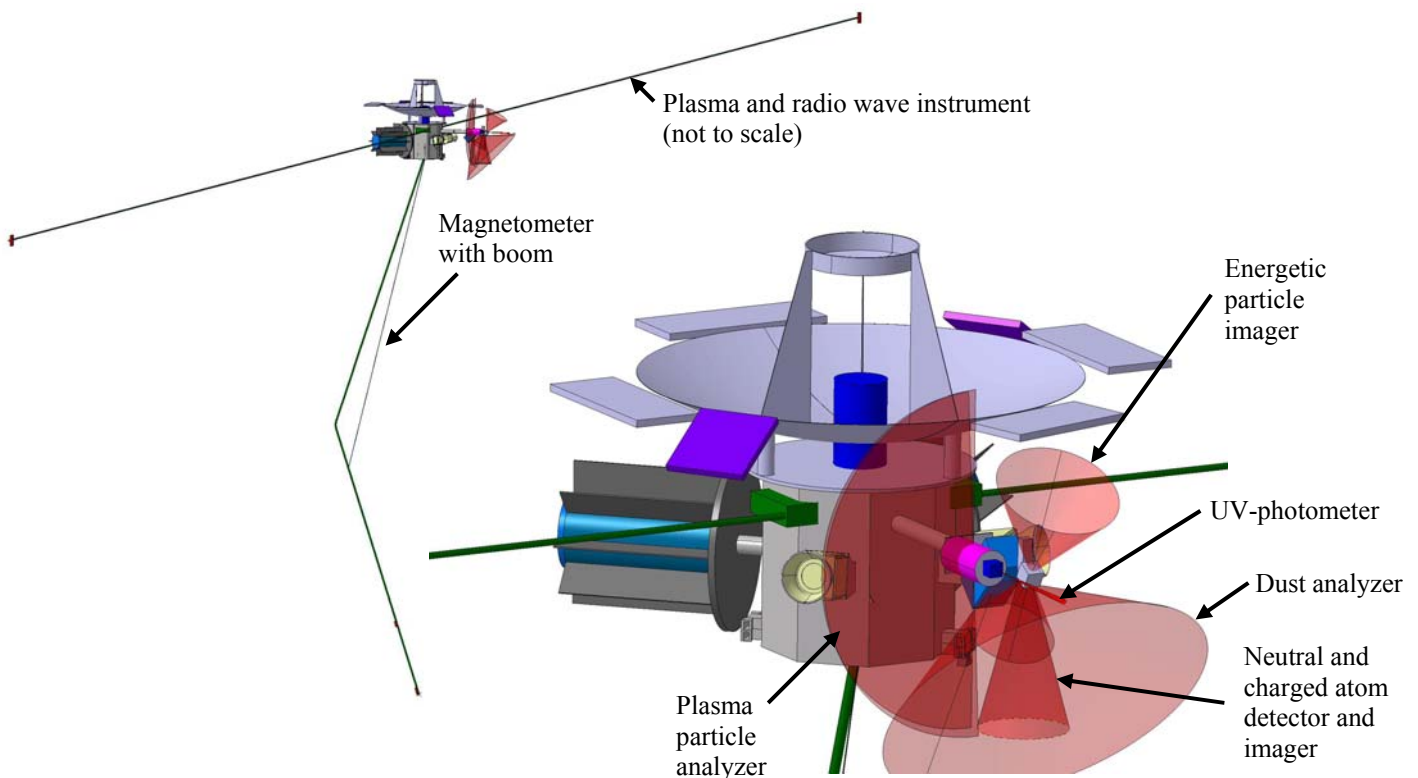


Figure 5-1: Science payload instruments and accommodation on the IHP platform.

6 CONCLUSION

Technology Reference Studies are detailed mission concept studies with the aim to identify enabling and enhancing technologies that are relevant for potential future science missions. The Interstellar Heliopause Probe TRS concentrates on exploration of the outer reaches of the heliosphere, thus requiring unconventional propulsion, communication and power subsystems. Additionally, the long lifetime requirement of approximately 30 years will drive all subsystem components.

A summary of the key technological challenges is provided in Table 6-1, and the definition of the Technology Readiness Levels (TRL) in Table 6-2. Though the IHP mission concept clearly requires many new and improved technologies, only those technologies that are considered technically realistic in the medium term have been baselined (assuming sufficient funding for technology development activities is allocated). It should be noted that many of the listed technologies not only require development, but also demonstration.

As a concluding remark, it is emphasized that due to the solar sail propulsion system, the overall mission concept is highly mass critical; a 1 kg increase in the platform or sail system requires an additional 130 m² sail area.

Table 6-1: Key enabling and enhancing technologies for the IHP mission concept.

Subsystem	Technology /Demonstration	Current TRL	Notes
Solar sail propulsion	Ultra-thin solar sail film material with reflective coating(s)	2	Currently, the best film has a 1.5 μm thickness. A 1.0 μm sail film with excellent mechanical and optical properties will need to be developed. The coating and film will have to withstand close packaging and extreme temperatures without significant degradation. Additionally, the sail film will need to be compatible with the launch (no/limited outgassing) and space environment (particle and UV radiation, micro-meteorites, charging effects). The manufacturing process, including coating(s), will need to be compatible with the production of large areas with a high quality.
	Solar sail assembly and verification	2	The assembly of a large solar sail area is non-trivial, particularly edge reinforcement, folding process and verification of required characteristics are considered challenging.
	Boom technology	3	For the deployment of the solar sail, long light-weight booms are required. Key issues are the required length and the extreme temperatures during the mission lifetime.
	Boom deployment mechanism	3	A European boom deployment mechanism already exists, but it will need to be modified to allow jettison of a significant mass fraction after deployment.
	Solar sail monitoring	3	Light weight sensors will need to be developed for monitoring the dynamic behaviour of the sail assembly in-flight. Additionally, for in-orbit solar sail demonstration, sail performance sensors are required.
	Attitude Control System (software)	3	The proposed attitude control system for the IHP mission concept will need to be further developed, tested and validated.
	Attitude Control System (hardware)	3	A light-weight, highly reliable 2 DOF gimbal mechanism is required.
	Solar sail jettison	2	After the solar sailing phase, the sail assembly needs to be jettisoned in a controlled and safe manner. Development, qualification and demonstration of this mechanism will be required.
	On-ground verification of deployment	2	The testing of the deployment system for a light-weight large area sail is difficult due to 1-g environment and the sail size.
	In-orbit demonstration	3	Deployment and sailing demonstration (sail propulsion performance, AOCS), possibly as part of a science mission, such as e.g. Geosail TRS concept. Also sail jettison will need to be demonstrated.

Subsystem	Technology /Demonstration	Current TRL	Notes
Power	Radio-isotope power source	2	Currently, there exist no European RTGs, though such power systems are essentially required for any scientific mission beyond Jupiter. The development of a European radio-isotopic power source with a specific power of 7 W/kg will be a major activity. Additionally, the use of an RTG in European facilities and launchers will require a significant amount of programmatic effort (health & safety requirements).
Attitude control	High I_{sp} micro-thrusters	6	FEEP thrusters are available, but additional lifetime tests as well as contamination tests and studies are required.
Communication	Power supply	3	Power supply technologies that provide high power output for short pulses (e.g. power capacitors).
	Power amplifier	2	RF semi-conductor technology for ~2 kW RF power at 29 GHz.
	Ultra-stable oscillator	4	Based on sapphire technology.
	Transponder	2	A Ka band transponder compatible with Impulse Radio scheme.
	Steerable high gain antenna	3	Ka band high gain antenna with piezo-controlled deformable secondary reflector for beam steering and defocusing
	Ground station	2	Upgrade for compatibility with Impulse Radio modulation at Ka-band.
Payload	Highly Integrated Payload Suite	2	A comprehensive highly integrated low resource instrument package with a long lifetime.

Table 6-2: Definition of Technology Readiness Levels.

TRL number	Definition
1	Basic principles observed and reported
2	Technology concept and/or application formulated
3	Analytical and experimental critical function and/or characteristic proof-of-concept
4	Component and/or breadboard validation in laboratory environment
5	Component and/or breadboard validation in relevant environment
6	(Sub)system model or prototype demonstration in a relevant environment (ground or space)
7	System prototype demonstration in a space environment
8	Actual system completed and "Flight qualified" through test and demonstration (ground or space)
9	Actual system "Flight proven" through successful mission operations

7 REFERENCES

- [Abelson05] R.D. Abelson, T.S. Balint, K. Coste et al., “Expanding Frontiers with Standard Radio-isotope Power Systems”, JPL report D-28902, 2005. Available from http://solarsystem.nasa.gov/multimedia/downloads/Standard_RPS_Report_Final_011205.pdf
- [Atzei05] A. Atzei, “Margin philosophy for assessment studies,” SCI-A/2003/302/AA, issue 2.0, 7 April 2005.
- [Etchegaray87] M.I. Etchegaray, “Preliminary Scientific Rationale for a Voyage to a Thousand Astronomical Units”, JPL Publication 87-17, 1987.
- [Kraft05] S. Kraft and J. Harris, “Interstellar Heliopause Payload Definition Document,” CR-PTRM-IHP-PDD, issue 3, Cosine Research BV, 11/11/2005.
- [Lappas05] V. Lappas, S. Wokes, M. Leipold, A. Lyngvi, P. Falkner, “Guidance and Control for an Interstellar Heliopause Probe (IHP) solar sail mission to 200 AU,” AIAA Guidance, Navigation, and Control Conference and Exhibit, San Francisco, August 2005 (AIAA-2005-6085).
- [Leipold00] M. Leipold, “Solar sail mission design,” Ph.D. Thesis, Technical University of Munich, 2000 (DLR Forschungsbericht 2000-22).
- [Leipold03] M. Leipold, M. Eiden, C. E. Garner et al., “Solar sail technology development and demonstration,” *Acta Astronautica*, vol. 51, pp. 317 – 326, 2003.
- [Leipold05] M. Leipold et al., “Interstellar Heliopause Probe summary report,” IHP-TN-KTH-001, issue 2, Kayser-Threde GmbH, 3/5/2005.
- [Leipold06] M. Leipold, H. Fichtner, B. Heber et al., “Heliopause Explorer—a sailcraft mission to the outer boundaries of the solar system,” *Acta Astronautica*, vol. 59, pp. 785 – 796, 2006.
- [Liewer00] P.C. Liewer, R.A. Mewaldt, J.A. Ayon and R.A. Wallace, “NASA’s Interstellar Probe Mission,” *AIP Conference Proceedings*, vol. 504, pp. 911 – 916, 2000.
- [Lura01] F. Lura, D. Hagelschuer, and A.I. Glotov, “Investigation of degradation effects for thin foil surfaces for solar sail missions caused by the complex simulation of space environment effects,” Proc. 4th intern. Symposium on environmental testing for space programmes, *ESA SP-467*, p. 115 – 119, 2001.
- [Macdonald07] M. Macdonald, C. McInnes, and B. Dachwald, “Heliocentric Solar Sail Orbit Transfers with Locally Optimal Control Laws,” *Journal of Spacecraft and Rockets*, vol. 44, pp. 273 – 276, 2007 (in press).
- [Mangus04] D. Mangus and A. Heaton, “Solar Sail Control Actuators Concepts,” Solar Sail Technology and Applications Conference, Greenbelt, MD, 28-29 Sep. 2004.

- [Marcuccio97] S. Marcuccio, S. Gianelli, M. Andrenucci, "Attitude and orbit control of small satellites and constellations with FEEP thrusters," *Proc. 25th Electric Propulsion Conference*, IEPC-97-188, pp. 1152 – 1159, 1997.
- [McInnes04] C.R. McInnes, "Solar sailing: Technology, dynamics and mission applications," Springer-Verlag, 2004.
- [McNutt04] R.L. McNutt Jr., G.B. Andrews, R.E. Gold et al., "A realistic interstellar explorer," *Advances in Space Research*, vol. 34, pp. 192 – 197, 2004.
- [McNutt06] R.L. McNutt Jr., R.E. Gold, T. Krimigis et al., "Innovative interstellar explorer," *AIP Conference Proceedings*, vol. 858, pp. 341 – 347, 2006.
- [Mewaldt95] R. A. Mewaldt, J. Kangas, S. J. Kerridge and M. Neugebauer, "A small interstellar probe to the heliospheric boundary and interstellar space," *Acta Astronautica*, vol. 35, pp. 267 – 276, 1995.
- [Schmidt05] G.R. Schmidt, R.L. Wiley, R.L. Richardson, and R.R. Furlong, "NASA's Program for Radioisotope Power System Research and Development," *AIP Conference Proceedings*, vol. 746, pp. 429 – 436, 2005.
- [Stofan06] E.R. Stofan, W. Anders, R.F. Beebe et al., "Priorities in Space Science Enabled by Nuclear Power and Propulsion," The National Academies Press, Washington DC, 2006. (available from http://www.nap.edu/catalog.php?record_id=11432)
- [Tajmar04] M. Tajmar, A. Genovese, and W. Steiger, "Indium Field Emission Electric Propulsion Microthruster experimental characterization," *Journal of propulsion and power*, vol. 20, pp. 211 – 218, 2004.
- [Talley02] C. Talley, W. Clayton, P. Gierow et al., "Advanced Membrane Materials for Improved Solar Sail Capabilities," 43rd AIAA/ASME/ASCE/AHS/ASC Structures, Structural Dynamics, and Materials Conference, Denver, Colorado, 2002 (AIAA-2002-1561).
- [Wallace00] R.A. Wallace, J.A. Ayon, and G.A. Sprague, "Interstellar Probe Mission/System Concept," *Proc. IEEE Aerospace Conference*, vol. 7, pp. 385 – 396, 2000.
- [Wie04] B. Wie, "Solar sail attitude control and dynamics, parts 1 and 2," *Journal of Guidance, Control and Dynamics*, vol. 27, pp. 526 – 535 (part 1), pp. 536 – 544 (part 2), 2004.
- [Wong06] W.A. Wong, D.J. Anderson, K.L. Tuttle, R.C. Tew, "Status of NASA's Advanced Radioisotope Power Conversion Technology Research and Development," *Proc. Space Technology and Applications International Forum (STAIF-2006)*, Albuquerque, NM, February 12-16, 2006.
- [Zank99] G.P. Zank, "Interaction of the solar wind with the local interstellar medium: A theoretical perspective," *Space Science Reviews*, vol. 89, pp. 413 – 688, 1999.

8 LIST OF PUBLICATIONS RELATED TO THE IHP TRS

- A. Lyngvi, P. Falkner, A. Atzei, D. Renton, M.L. van den Berg, and A. Peacock, "ESA's Technology Reference Studies," *Proc. of 55th International Astronautical Congress*, Vancouver, IAC-04-U.1.06, 2004.
- A. Lyngvi, P. Falkner, S. Kemble, M. Leidold, and A. Peacock, "The Interstellar heliopause probe," *Proc. 54th International Astronautical Congress*, Vancouver, IAC-04-Q.2.A.06, 2004.
- A. Lyngvi, P. Falkner, and A. Peacock, "The Interstellar Heliopause Probe," Proc. 37th ESLAB Symposium, *ESA SP-543*, pp. 11 – 17, 2004.
- P. Falkner, A. Lyngvi, M.L. van den Berg, D. Renton, A. Atzei, "ESA's Technology Reference Studies," Proc. of 8th ESA Workshop on Advanced Space Technologies for Robotics and Automation, 'ASTRA 2004,' *ESA-WPP 236*, A-02, 2004.
<http://robotics.estec.esa.int/AUTOLINKS/ASTRA/index.html>)
- 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, "Study of highly integrated payload architectures for future planetary missions," *Proc. SPIE*, vol. 5570, pp. 133 – 144, 2004.
- P. Falkner, M.L. van den Berg, D. Renton, A. Atzei, A. Lyngvi, A. Peacock, "ESA's Technology Reference Studies," *Geophysical Research Abstracts*, vol. 7, p. 5115, 2005.
- S. Kraft, J. Moorhouse, M. Collon, M. Beijersbergen, J. Harris, M.L. van den Berg, A. Atzei, A. Lyngvi, D. Renton, C. Erd, and P. Falkner, "On the study of highly integrated payload architectures and instrumentation for future planetary missions" *Geophysical Research Abstracts*, vol. 7, p. 8291, 2005.
- M.L. van den Berg, A. Lyngvi, P. Falkner, S. Kemble, M. Leipold, S. Kraft, A. Peacock, "Interstellar heliopause Probe, an ESA Technology Reference Study," *Geophysical Research Abstracts*, vol. 7, p. 4804, 2005.
- M. Leipold, A. Lyngvi, P. Falkner, V. Lappas, H. Fichtner, S. Kraft, and B. Heber, "Interstellar Heliopause Probe. Design of a Challenging Mission to 200 AU," Proc. 39th ESLAB symposium, *ESA SP-588*, pp. 241 – 248, 2005.
- V. Lappas, S. Wokes, M. Leipold, A. Lyngvi, P. Falkner, "Guidance and Control for an Interstellar Heliopause Probe (IHP) solar sail mission to 200 AU," AIAA Guidance, Navigation, and Control Conference and Exhibit, San Francisco, August 2005 (AIAA-2005-6085).
- P. Falkner, M.L. van den Berg, D. Renton, A. Atzei, A. Lyngvi, A. Peacock, "Update on ESA's Technology Reference Studies," *Proc. of 5th International Astronautical Congress*, Fukuoka, IAC-05-A3.2.A.07, 2005.

A. Lyngvi, P. Falkner, S. Kemble, M. Leipold, A. Peacock, “The interstellar Heliopause probe,” *Acta Astronautica*, vol. 57, 104 – 111, 2005.

A. Lyngvi, P. Falkner, A Peacock, “The interstellar heliopause probe technology reference study,” *Advances in Space Research*, vol. 35, 2073 – 2077, 2005.

9 ACRONYMS

ADCS	Attitude Determination and Control System
AI&T	Assembly, Integration and Testing
AU	Astronomical Unit (1.496×10^8 km)
BOL	Beginning of Life
CFRP	Carbon Fibre Reinforced Plastics
CoM	Center of Mass
CoP	Center of Pressure
DLR	Deutsche Zentrum für Luft- und Raumfahrt
DOF	Degrees Of Freedom
EGA	Earth Gravity Assist
EMC	Electro Magnetic Compatibility
ESA	European Space Agency
FEEP	Field Emission Electric Propulsion
FOS	Fibre Optical Sensor
GA	Gravity Assist
GEO	Geostationary Earth Orbit
GTO	Geostationary Transfer Orbit
GSE	Ground Support Equipment
HGA	High Gain Antenna
HIPS	Highly Integrated Payload Suite
IHP	Interstellar Heliopause Probe
ISM	Interstellar Medium
JGA	Jupiter Gravity Assist
LGA	Lunar Gravity Assist
LGA	Low Gain Antenna
LISM	Local Interstellar Medium
MLI	Multi Layer Insulation
NEP	Nuclear Electric Propulsion
NTP	Nuclear Thermal Propulsion
OBDH	On-Board Data Handling
P/L	Payload
RF	Radio Frequency
RTG	Radioisotope Thermoelectric generator
S/C	Spacecraft

SCI-A	Science Payload & Advanced Concepts Office
SEP	Solar Electric Propulsion
SPA	Solar Photonic Assist
SSC	Surrey Space Centre
SSM	Solid State Memory
TM/TC	Telemetry/Telecommand
TRS	Technology Reference Study
TT&C	Telemetry, Tracking and Command
TVC	Thrust Vector Control