An Interstellar – Heliopause mission using a combination of solar/radioisotope electric propulsion

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There is common agreement within the scientific community that in order to understand our local galactic environment it will be necessary to send a spacecraft into the region beyond the solar wind termination shock. Considering distances of 200 AU for a new mission, one needs a spacecraft travelling at a speed of close to 10 AU/yr in order to keep the mission duration in the range of less than 25 yrs, a transfer time postulated by ESA.
Two propulsion options for the mission have been proposed and discussed so far: the solar sail propulsion and the ballistic/radioisotope electric propulsion. As a further alternative, we here investigate a combination of solar-electric propulsion and radioisotope-electric propulsion. The solar-electric propulsion stage consists of six 22 cm diameter "RIT-22"ion thrusters working with a high specific impulse of 7377 s corresponding to a positive grid voltage of 5 kV. Solar power of 53 kW BOM is provided by a light-weight solar array. The REP-stage consists of four space-proven 10 cm diameter "RIT-10" ion thrusters that will be operating one after the other for 9 yrs in total. Four advanced radioisotope generators provide 648 W at BOM. The scientific instrument package is oriented at earlier studies. For its mass and electric power requirement 35 kg and 35 W are assessed, respectively.
Optimized trajectory calculations, treated in a separate contribution, are based on our "InTrance" method. The program yields a burn out of the REP stage in a distance of 79.6 AU for a usage of 154 kg of Xe propellant. With a C_3 = 45,1 (km/s)^2 a heliocentric probe velocity of 10 AU/yr is reached at this distance, provided a close Jupiter gravity assist adds a velocity increment of 2.7 AU/yr. A transfer time of 23.8 yrs results for this scenario requiring about 450 kg Xe for the SEP stage, jettisoned at 3 AU. We interpret the SEP/REP propulsion as a competing alternative to solar sail and ballistic/REP propulsion. Omiting a Jupiter fly-by even allows more launch flexibility, leaving the mission duration in the range of the ESA specification.

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Nomenclature

AU	=	astronomical unit
BOM	=	begin of mission
C_3	=	hyperbolic excess energy
DLR	=	German Aerospace Center
EP	=	electric propulsion
GA	=	gravity assist
IHP	=	Interstellar Heliopause Probe
JGA	=	Jupiter gravity assist
PPU	=	power processing unit
REP	=	radioisotope electric propulsion
RTG	=	radioisotope thermal generator
RPM	=	revolutions per minute
SEP	=	solar electric propulsion

I. Introduction

Since more than 30 years, the space community proposes an interstellar heliopause probe to investigate the outer regions of the heliosphere and the very local interstellar medium and their mutual interactions as well[1]. As a recent review of the earlier and the ongoing studies we cite the interstellar white paper worked out by McNutt for the U.S. Heliophysics Decadal Survey[2]. Both NASA and ESA included an IHP-project in their future plans.[3,4]

However, the challenge to reach a solar distance of 200 AU within 25 yrs (ESA) or in only 15 years (NASA) needs the realization of mean heliocentric flight velocities between 38 km/s and 53.4 km /s, corresponding to 8 AU/y and 13.3 AU/y, respectively. This requires doubtlessly a very advanced and sophisticated propulsion system. A ballistic system even in combination with a near-Sun swingby or multiple gravity assists (GA) is unable to perform the task, and nuclear power plants seem to be disfavoured by a specific mass being too large for realizing the mentioned mission durations. In consequence both agencies favoured a solar sail propulsion method[5,6]. However, the performance data, especially with respect to required size and surface area-to-mass ratio are very challenging. In addition, flying a solar sail within a solar distance of 0.25 AU is problematic, too. As an alternative, electric propulsion with radioisotope power sources in combination with a very large C_3 from the launcher and with GAs has been studied[7,8,9]. In two contributions to this conference, we investigate a combination of solar electric propulsion (SEP) with radioisotope electric propulsion (REP), and with a JGA[10,11] with respect to the flight time to reach 200 AU. In the here presented contribution the spacecraft and the propulsion modules are described. The paper is organized as follows: First the scientific task is shortly mentioned. The probe itself is described followed by the layout of the REP and SEP stages. The details of the trajectory calculations and a discussion of the effect of the different steps within the mission scenario on the flight time are presented in our second contribution IEPC-2011-051[12].

II. Scientific Task

The boundary between the heliopsphere, i.e. the region filled by the solar plasma, and the undisturbed interstellar space is expected to exist near or beyond about 200 AU (in the direction of the Sun motion). Models describe the structure of the heliosphere as follows[13]:

The originally supersonic solar wind, flowing outwards, becomes subsonic at about 90 to 100 AU and generates there a spherical shock wave, the so-called termination shock. The inner heliosheath follows. From there the anomalous cosmic radiation is assumed to originate. The heliopause is the area between the heliosphere and the interstellar space, ranging from about 120 to 150 AU. In the heliopause, solar and interstellar plasmas and fields are colliding. Analogously to the Earth's magnetosphere, the heliopause has a drop-like shape with its nose in the direction of the motion of the solar system and with a tail at the opposite end. Beyond the heliopause an outer heliosheath and a so called hydrogen wall are modelled. Furthermore, it is speculated that a second shock wave front called bow shock appears in about 270 AU where the supersonic interstellar plasma collides with the heliopause. Measured data are so far supplied by Voyager 1 and 2; their power from radioisotope generators will allow them to work until they reach about 130-140 AU. Voyager 1 passed the termination shock in 2004, Voyager 2 in 2007. The ongoing in situ data submitted by these two ageing probes for the plasma in the range of the inner heliosheath are unexpected with respect to the model predictions and ask for a reconsidering of the modelling. With respect to a more sensitive and comprehensive instrumentation, an IHP – mission with far reaching measurements is certainly needed.

III. Probe Concept

The design of our IHP-probe follows more or less other concepts [e.g.[6]]. 11 different scientific instruments are foreseen including a radar system and a telecamera to observe eventually Kuiper belt objects along the trajectory. Package mass and power consumption are assumed to amount to 35 kg and 35 W, respectively.

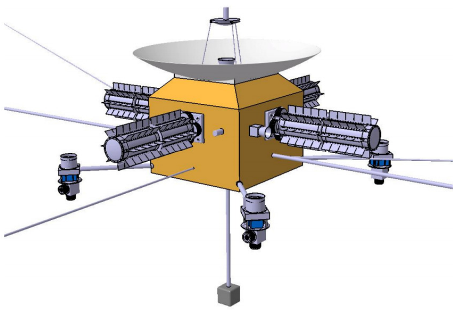

Figure 1 Drawing of the proposed IHP probe showing structure, radioisotope generators, EP-propulsion and control engines, high gain antenna, low gain antenna, instrument block, magnetometer booms, plasma wave antennas, swivel camera.

To enable all instruments to scan the entire celestial space, the probe will rotate with about 3 RPM taking care also of spin stabilization. The rotation axis points in flight direction towards the nose of the heliosphere. The instrument package is mounted at an axial telescopic boom, 1m to 2m long, in order to reduce disturbances by the probe body. At the rear side of the probe, facing the Earth, a 2 m diameter high gain antenna is mounted; using its movable secondary reflector, the transmission beam direction can be controlled. The recorded data are transmitted to Earth block wise once per week for some hours. The main body of the probe is made of Al-honeycomb; its dimensions amount to 1.2 m in width and height.

For the power supply of the scientific instruments, the telemetry, and housekeeping,at least three advanced radioisotope batteries (8.5 W/kg) would be required delivering at BOM 486 W of electric power. This power is not used for most of the interplanetary flight, and is thus available for an electric ion engine working with relatively small power but for a long duration. For the here selected REP-propulsion stage, described in the following section, in total four RTG are considered delivering 648 W at BOM.

Fig.1 displays the probe concept with magnetometer booms and plasma wave antennas deployed. The four radioisotope generators are mounted at the four sides of the bus of the probe. Four RIT-10 ion engines (see next section) are fixed at the bottom of the probe. Their mounting structures also hold four attitude control units each consisting of three orthogonally oriented µRIT engines. The probe is fixed on top of the SEP stage described below.

IV. REP Module

Thrusters of the RIT type have been selected for the REP module. Because of the limited power of the radioisotope generators, the small 10 cm diameter RIT-10 thruster has been chosen. This thruster has flight heritage since it was used on ESA´s satellite EURECA and on ARTEMIS[14]. It was tested successfully for its lifetime at ESTEC for 23000 hrs. Performance variations of the REP/SEP combination resulted in a preferable beam voltage of 1.5 kV or specific impulse of 3810 s. With a power consumption of 592 W and a Xenon propellant flow of 0.558 mg/s, the RIT-10 thruster delivers a thrust of 21 mN. Four RIT-10 engines were envisaged, running one after the other and thus, accelerating the IHP probe continuously nearly 10 yrs except during phases of data transmittance when the working thruster has to be switched off. Within their lifetime, the four thrusters consume 158 kg of Xe which adds to the mass of the dry REP stage. The mass of the dry REP-stage results from the four thrusters, two power supplies (one of them are considered for redundancy), telemetric and data handling units, thermal and attitude control elements, structure and harness. Including the scientific payload of 35 kg, a dry mass of 344 kg results (Table 1).

Table 1. Mass balance of the REP-module including the probe and its scientific payload.

Scientific payload (11 instruments)	35 kg
On-board-data handling & control	12 kg
Harness, cabling	13 kg
Power production (4 RTGs)	76 kg
Telemetry, tracking control	40 kg
Thermal control of probe	27 kg
Attitude control system (including propellant)	40 kg
Structure	72 kg
REP-propulsion module (4 RIT-10 with tank)	29 kg
Dry mass of probe and REP-stage	344 kg
Propellant for REP-propulsion	154 kg
Payload of the SEP-stage (without adapter)	498 kg

As already mentioned above, because of the slightly larger power requirement of the RIT-10 thruster with respect to the power needed by the IHP-probe, four advanced RTG units with specific energy 8.5W/kg[15] have to be mounted on the spacecraft. The generators use the isotope Pu238 having a halftime of 86.4 yrs. Assuming that each RTG produces 162 W BOM a mass of the four RTGs of 76 kg results. The Multimission Radioisotope Thermoelectric Generator (MMRTG) looking similar to the RTGs in Fig.1 and 3 does not fulfil these specifications. It has a length of 65cm and delivers $115W_e$ at a specific energy of 2.6 W/kg[16]. An Advanced Stirling Radioisotope Generator (ASRG) with a length of 62 cm has specifications close to the requirement[17], but has the disadvantage of moving parts. On the other hand it has the highest Technology Readiness Level with respect to the alternative Alkali Thermal Electric Converter (AMTEC)[18] or Radioisotope Thermophotovoltaic Generator (RTVP)[19], both being in the status of technological development.

Since the RTGs supply both the probe and the REP stage, they may be regarded as part of the probe spacecraft. Including the propellant mass of 154 kg, a total mass of nearly 500 kg finally results which serves as payload for the SEP stage (Table 1).

V. SEP Stage

Like in earlier interplanetary mission studies by the authors[10,] the 22 cm diameter Xe-ion thrusters RIT-22, having been qualified by EADS Astrium Space Transportation, is selected for the SEP stage (Fig.2). It is equipped with a multi-hole two-grid electrostatic acceleration part and a radiofrequency propellant gas ionizer consisting of an Alumina discharge vessel with the induction coil around it. Due to the electrodeless discharge, a high reliability and a long lifetime (23,000 hrs) at full power operation are guaranteed. As a further advantage holds that an increase of the high grid voltages, i.e. of the specific impulse, does not require a change of the ionizer (e.g. a biasing of discharge electrodes), provided the grid dimensions are suitably scaled[20].

In the trajectory optimization program, which is described in our second contribution IEPC-2011-051[12], the number of thrusters and their specific impulse have been varied. As the trajectory in the beginning of the mission is allowed to dive into the inner region of the solar system, the solar power plant has been designed in such a way that it delivers at BOM in 1 AU (only) 65, 75, or 85 % of the total power requirement of the thrusters. This proves beneficial because it reduces mass and dimensions of the solar array without increasing the flight time.

The results of the trajectory calculations made us to select a cluster of six RIT-22 thrusters running with 5 kV of positive high voltage corresponding to a specific impulse of 7377 s. In Fig. 3 the SEP stage with the mentioned 6 RIT-22 engines is displayed. The propellant storage and feed system, electronic components and thermal control parts are mounted within the bus structure. The IHP probe with the REP-stage is mounted on top of the hexagonal SEP stage. They separate at a solar distance of about 3AU. Once the SEP stage has been jettisoned, the integrated IHP-REP module must be turned by 180 deg and spinned up. In this orientation the high gain antenna will point

Figure 2. Photo of the ion thrusters RIT-22 manufactured by EADS Astrium ST. The standard engine´s nominal operation performance date are: 175 mN thrust, 4760 s specific impulse, 5.83 kW power consumption, 3.72 mg/s propellant consumption.

towards Earth, and thrusting will occur in flight direction. The extended boom carrying the instrument package at the top of the spacecraft will face the boundary of the heliosphere.

For the selected throttling at 1AU (BOM) of 65 %, the total thrust amounts to 1.05 N. 53 kW of solar power have to be provided by the solar array. For the latter, a solar array with a specific mass of 5 kg/kW or 200 W/kg has been adopted. It is oriented at the present developments of powerful and light-weight solar arrays for deep space missions especially in the US. For the IHP spacecraft we propose to use either the UltraFlex system or the SquareRigger system. For the UltraFlex system[20], four advanced wings would be sufficient as displayed in Fig.4. In both cases flexible blankets with thin triple-junction GaAs solar cells are the baseline. For the complete thrust unit, a mass of 224 kg results from the addition of thruster mass, PPU , gas feed system and electronics while the mass of the solar array amounts to 265kg. Table 2 displays performance data of the SEP stage, while table 3 contains a mass breakdown of the launcher payload (see section VI).

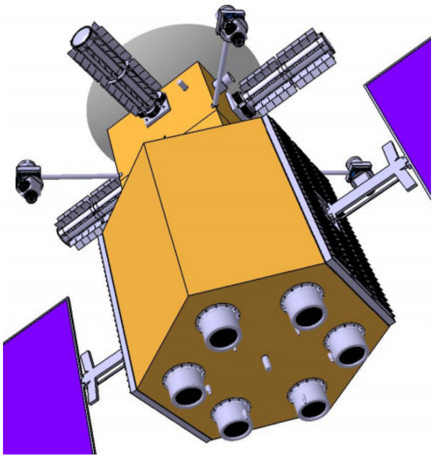

Figure.3. Drawing of the SEP-propulsion stage with 6 RIT-22 ion thrusters and the IHP-probe with REP-stage mounted on top.

Table 2. Performance data of the SEP stage

Mass of solar array	265 kg
Thrust units (6 RIT-22 + power units)	224kg
Xe-tank	26 kg
Structure	239 kg
Solar array power at BOM	53 kW
Beam voltage of RIT-22 thruster	5 kV
Throttling rate at BOM	65 %
Thrust (BOM)	1.05 N
Acceleration (BOM)	0.62 mm/s^2
Total propellant load within lifetime	42 %

Table 3. Mass balance of the IHP probe with SEP/REP stages

IHP-REP combination (dry)	344 kg
REP- propellant	154 kg
IHP-REP combination (fueled)	498 kg
SEP-Stage (dry)	754 kg
SEP-propellant (Xe)	440 kg
SEP-stage (fueled)	1194 kg
Total launch mass	1692 kg

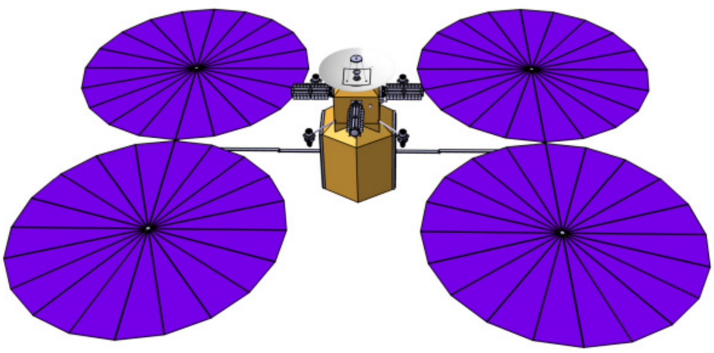

Figure 4 Solar power array consisting of four next generations Ultraflex panels

VI. Trajectory calculations

A detailed description of the applied trajectory analyis and a discussion of its results for flight time, propellant mass and launch mass are the content of IEPC2011-2011-051[12]. For completeness we just mention that the trajectory analysis has been carried out using a low-thrust program "InTrance"[22]. It applies an "evolutionary neurocontrol (ENC)"which is based on a combination of artificial neural networks with evolutionary algorithms. The program attacks low-thrust optimization problems from the perspective of artificial intelligence and machine learning.The main parameter for the optimization procedure is the flight time. A value which is much smaller than the 25 yrs, expected by NASA, is doubtlessly desirable but is certainly also ambitious. Of influence for shortening the flight time is also the used launcher. A hyperbolic excess energy, C3 >0 will allow a reduction of the propellant mass on board of the spacecraft and a corresponding increase of the acceleration by the propulsion system. In tables 2 and 3 the mass balance for the SEP stage and the total launch mass for an Ariane ECA launcher[23] are given, allowing a hyperbolic escape with C_3 = 45.1 $(km/s)^2$. For C_3 = 0 the launch mass will increase by about 300 kg due to an increase of the propellant. A smaller launcher will be possible for this choice of C3, on the cost of an increase of the flight time by about two years. A further important possibility of a flight time reduction is offered by a gravity assist e.g. at Jupiter.

VII. Summary and Conclusions

A strategy of a robotic mission to the boundary between the heliosphere and the interstellar space in about 200 AU was investigated. It aims, in the first place, to investigate the applicability of SEP as an alternative to the so far discussed ballistic/REP approach or the solar sail propulsion treated in the NASA and ESA studies. Instrumentation and probe were oriented at the earlier proposals. Using SEP alone, flight times in the range of 35 to 41 years result. Adding a REP stage which makes use of the RTGs of the probe, the flight time is reduced by about six years. Both values are comparable with the result of the ballistic studies and their REP additions provided they include a projected heavy launcher. In contrast, using SEP requires a smaller launcher and when compared to solar sail propulsion a much less approach to the Sun. Finally, making use of a JGA between the SEP phase and the REP phase, in a flight time of 23.75 yrs results.

This value has so far only been yielded for the ballistic/JGA/REP mission strategy. But is not as short as the 15 yrs envisaged by the NASA, a value of the solar sail propulsion studies. Those are based on critical specifications of areal densities and the total sail area. In conclusion, the presented study documents that SEP combined with REP enables to send a spacecraft within less then 25 yrs to a solar distance of 200 AU using state-of-the art technology.

Acknowledgement

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