Interstellar Heliopause Probe

H.W. Loeb, K.-H. Schartner, B. Dachwald, A. Ohndorf, W. Seboldt

Abstract

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 traveling 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 European Space Agency (ESA). Two propulsion options for the mission have been proposed and discussed so far: the solar sail propulsion and the ballistic/radioisotope-electric propulsion (REP). As a further alternative, we here investigate a combination of solar-electric propulsion (SEP) and REP. The SEP stage consists of six 22-cms 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 at begin of mission (BOM) is provided by a light-weight solar array.

The REP stage consists of four space-proven 10 cm diameter RIT-10 (radio-frequency ion thruster) that will be operating one after the other for 9 yrs in total. Four advanced radioisotope generators provide 648 W at the beginning of mission (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, 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 Xenon propellant. With a hyperbolic excess energy C3 = 45.1 km^2/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 Xenon for the SEP stage, jettisoned at 3 AU. We interpret the solar and SEP/REP as a competing alternative to solar sail and ballistic/REP. Omitting a Jupiter fly-by even allows more launch flexibility, leaving the mission duration in the range of the ESA specification.

Keywords

radio-frequency ion thruster; solar electric propulsion; interstellar heliopause probe; mission strategy; radioisotope-electric propulsion

1. Introduction

Most challenging for the application of solar-electric propulsion (SEP) are very long distance missions with an extremely high velocity increment v D , like the Interstellar Heliopause Probe (IHP) or the Deep Space Gravity Probe (DSGP). Both ESA and NASA take those missions into account for their future programs. However, the challenge to reach a solar distance of 200 AU within 25 yrs (ESA) or within only 15 years (NASA) needs the realization of mean flight velocities between 38 km/s and 53.4 km/s, corresponding to 8 AU/yr and 13.3 AU/yr, respectively. A ballistic system, even in combination with near-Sun or multiple gravity assists, is unable to perform the task, and nuclear power plants are still not available for space applications. In consequence, both agencies favored solar sail propulsion [1, 2]. However, the actual performance data, especially with respect to required size and area-specific mass do not yet meet the propulsive requirements. In addition, flying a solar sail to within 0.25 AU of solar distance is also problematic from a thermal control point. As an alternative, a promising way for this challenge may be offered by SEP, provided that the thrusters are able to generate specific impulses around 8000 s. These specifications may be realized by using rf-ion engines of the RIT-22 type with high-voltage grids (6 kV).

2. Mission Strategy

The mission strategy using solar-electric ion propulsion considers first a trajectory leg into the inner region of the solar system in order to accumulate a large flight speed. On the outbound leg of the trajectory, the SEP-module will be jettisoned after its burnout. The electric power, which is then needed at far distance for the scientific instruments and data transmission, has to be provided by radioisotope generators. Since their electric energy is available all the time, and thus also during the long distance cruise, it is probably reasonable to equip the scientific probe with ion thrusters that start working once the SEP-module was jettisoned. The velocity increment resulting from using a REP-stage was already investigated earlier though in a combination with chemical propulsion (C3>0) and single or multiple gravity assists [3]. Such a ballistic/REP combination needs a very heavy launcher. Here we investigate a SEP/REP combination as an alternative, which has the advantage of state-of-the-art technology for the major components.

3. Spacecraft

The design of our IHP-probe follows more or less other existing concepts, e.g. [2]. Eleven different scientific instruments are foreseen, including a radar system and a telecamera to observe eventual Kuiper Belt objects along the trajectory. Package mass and power consumption are assumed to be 35 kg and 35 W, respectively. To enable all instruments to scan the entire space, the probe will rotate with about 3 RPM, yielding also spin stabilization. The rotation axis points in flight direction towards the nose of the heliosphere. An instrument package is mounted at an axial boom, 1 m to 2 m long, in order to reduce disturbances by the probe body (right part of Figure 1).

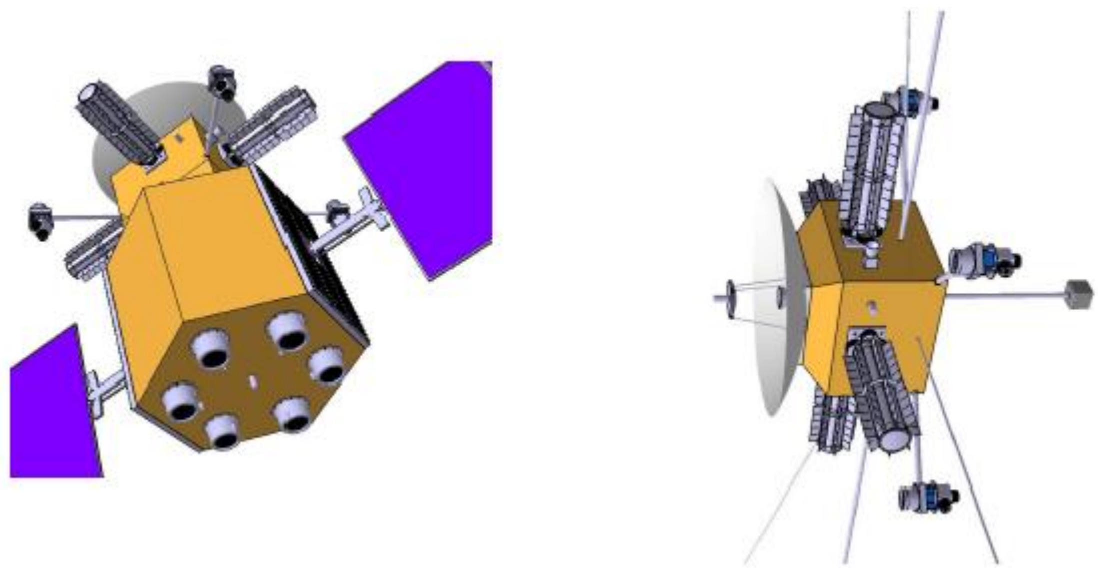

Figure 1 – Spacecraft at Begin of Mission, Consisting of SEP-Stage, and Combined REP-Stage and IHP Probe (Left), IHP-Probe with REP-Stage (Right)

Like in earlier interplanetary mission studies by the authors [4], the 22-cm diameter Xenon-ion thruster RIT-22, having been qualified by EADS Astrium Space Transportation, is selected for the SEP-stage. 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 at full power operation are guaranteed. As a further advantage, an increase of the high grid voltages, i.e. of the specific impulse, does not change the ionizer (e.g. a biasing of discharge electrodes), provided the grid dimensions are suitably scaled [5]. The results of the trajectory calculations (see below) 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. For the selected throttling at 1AU (begin of mission (BOM)) of 65 %, the total thrust amounts to 1.05 N. 53 kW of solar power has 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 (UltraFlex system or the SquareRigger system).

Figure 1 displays the complete spacecraft at the BOM, consisting of the IHP probe and the two propulsion stages. The REP stage with the four RIT-10 engines, mounted at booms, is located between SEP-stage and scientific probe. Note that once the SEP-stage has been jettisoned, the integrated IHP-REP module must be turned by 180 degrees and spinned up. In this orientation the antenna will point towards the Earth, and thrusting will occur in flight direction. The extended boom at the top of the spacecraft (Figure 1, right part) will face the boundary of the hemisphere. Figure 2 displays the spacecraft amidst the deployed UlraFlex solar panels. For the same reasons as for the SEP-stage, thrusters of the RIT-type have been selected for the REP-module. However, because of the limited power of the radioisotope generators, the smaller 10-cm diameter RIT-10 thruster has been chosen. This thruster has flight heritage.

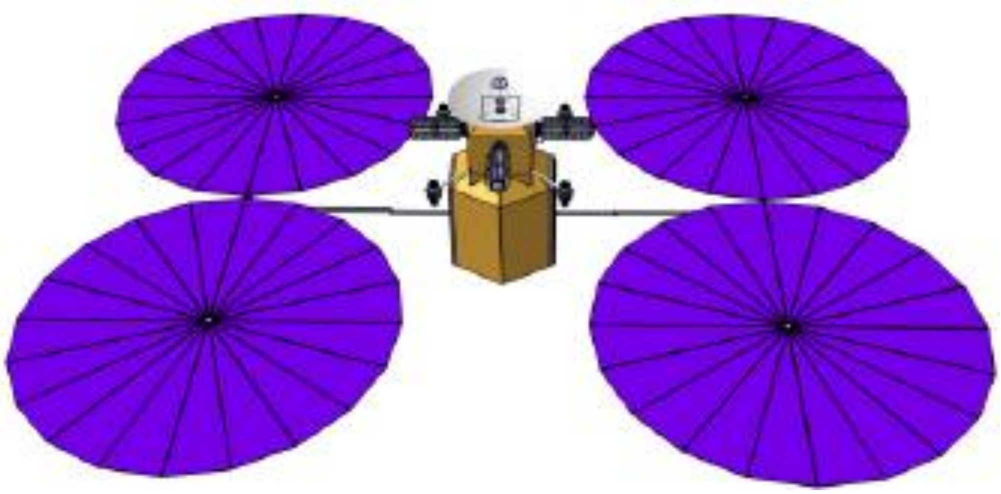

Figure 2 – Spacecraft amidst Deployed Solar Panels

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 at the chosen specific impulse of 3810 s. Four RIT-10 engines were envisaged, running one after the other and thus, accelerating the IHP probe continuously except during phases of data transmission, when the working thruster has to be switched off. Within their lifetime, the four thrusters consume 154 kg of Xenon which adds to the mass of the dry REP-stage. Caused by the slightly larger power requirement of the RIT-10 thruster, four radioisotope thermoelectric generator (RTG) units (isotope Pu238) have to be mounted on the spacecraft. Their specific mass is specified to be 8.5 W/kg resulting in a mass of 76 kg for the four RTGs. 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 is nearly 500 kg results as payload for the SEP-stage.

4. Trajectory Calculations and Results

The global low-thrust trajectory optimization code «InTrance» [6, 7] was used for trajectory analysis and optimization. The optimization objectives for this multi-objective problem are flight time and propellant mass. Although both objectives were considered in our optimization, the main focus was on flight time. The first baseline design was a pure SEP spacecraft. A variation of the number of RIT-22 thrust units between 4 and 6 showed that more thrusters reduce the flight time but also increase the required propellant mass. This way, only flight times of slightly less than 36 years could be achieved as displayed in the upper part of Figure 3.

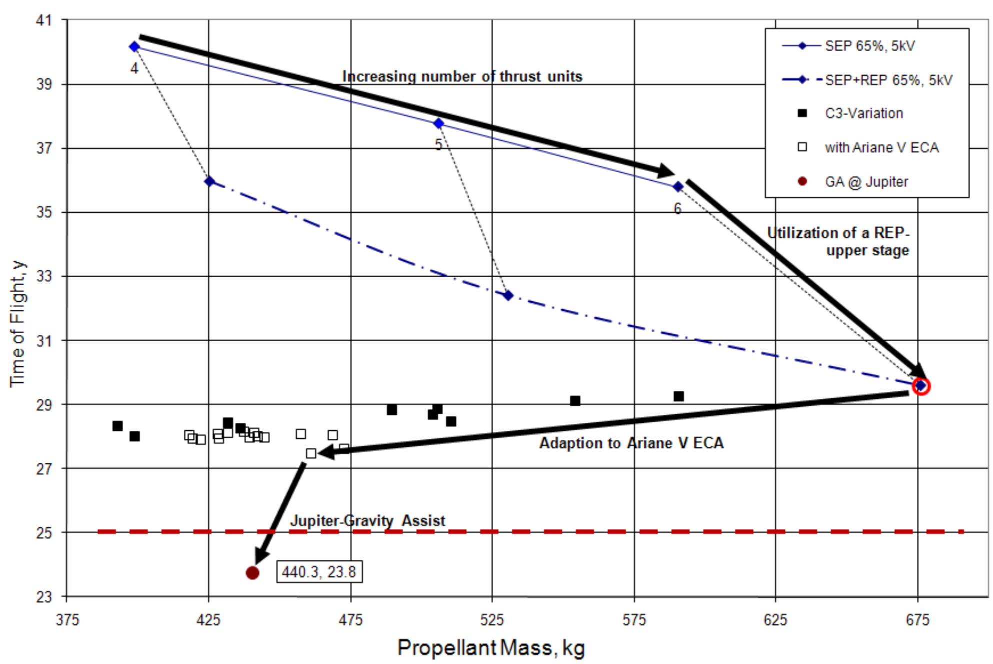

Figure 3 – Results of the trajectory calculations using 4.5 or 6 thrusters without and with REP stage. Reduction of the flight time due to an increasing C3 value and due to a Jupiter gravity assist

The strategy found by the optimizer was to go into the inner region of the solar system to gain orbital energy. This way, it is possible to throttle the engines to 65 % at 1 AU and thus to reduce the required solar array area and mass.

For the remaining calculations adding the REP-stage and varying C3 as well as including a Jupiter gravity assist (JGA), we have chosen to use six RIT-22 thrusters. For the selected throttling of 65 % at 1 AU, the required total power is 53 kW and the total thrust is 1.05 N. A further flight time reduction of about six yrs to slightly less than 30 years could be achieved by using the REP upper stage. For this stage, the four RIT-10 engines with a maximum thrust of 21 mN each and a specific impulse of 3810 s are operated consecutively. This reduction, however, is still based on an interplanetary injection with zero hyperbolic excess energy, C3. A variation of C3 showed that the flight times could only be reduced slightly for the optimal strategy, but the C3 reduced the propellant mass considerably. An adaptation of the spacecraft to the C3 that an Ariane 5 ESA could provide yielded a propellant mass reduction of about 200 kg. The flight time was also reduced by about 2 years, which is still above the 25 years required by ESA.

However, these results show that even without a gravity assist, flight times of less than 28 years can be achieved. This option might strongly be considered for an IHP mission because it is very flexible with respect to launch windows. If the flight time should be lowered below 25 years, this is only possible with a JGA. In this case, the resulting flight time is 23.75 years for the optimal constellation between Earth and Jupiter. C3 = 45.1 km^2/s^2 and the described SEP-stage give a heliocentric velocity of 6.44 AU/yr at 3.05 AU, where the SEP-stage is jettisoned. This velocity drops to 5.65 AU/yr until the JGA, which increases the velocity by 2.64 AU/yr. After the 9-year REP-phase, at 80 AU, the spacecraft has a velocity of 10 AU/yr. For this design, the total launch mass of the spacecraft is 1692 kg, the dry IHP-REP combination contributing 344 kg.

Acknowledgment

This research was supported by DLR under contract no. 50 RS 901.

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