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Technology Requirements of Exploration Beyond Neptune by Solar Sail Propulsion

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This paper provides a set of requirements for the technology development of a solar-sail-propelled Interstellar Heliopause Probe mission. The mission is placed in the context of other outer solar system missions, ranging from a Kuiper belt mission to an Oort cloud mission. Mission requirements are defined and a detailed parametric trajectory analysis and launch-date scan are performed. Through analysis of the complete mission trade space, a set of critical technology development requirements are identified that include an advanced lightweight composite high-gain antenna, a high-efficiency Ka-band traveling-wave tube amplifier, and a radioisotope thermoelectric generator with power density of approximately 12 W/kg. It is also shown that the Interstellar Heliopause Probe mission necessitates the use of a spinning sail, limiting the direct application of current hardware development activities. A Kuiper belt mission is then considered as a precursor to the Interstellar Heliopause Probe, and it is also shown through study of an Oort cloud mission that the Interstellar Heliopause Probe mission is the likely end goal of any future solar sail technology development program. As such, the technology requirements identified to enable the Interstellar Heliopause Probe must be enabled through all prior missions, with each mission acting as an enabling facilitator toward the next.

Nomenclature

A	=	solar sail reflective-surface area, m^2
a_s	=	solar sail characteristic acceleration, mm/s^2
C_3	=	launch energy, km^2/s^2
m_p	=	spacecraft mass minus solar sail subsystem, kg
m_{sf}	=	mass fraction of the spacecraft that is not the solar sail subsystem
N	=	number of Oort cloud objects encountered
P	=	magnitude of solar radiation pressure on a perfectly absorbing surface at 1 AU (astronomical unit), 4.56×10^{-6} N/m^2
R	=	radius of sphere within which Oort cloud objects will be encountered, m
r_a	=	radius of aphelion, m
r_p	=	radius of perihelion, m
t	=	time, s
v_∞	=	velocity at infinity, AU/year
β	=	sail lightness number
Δv	=	change in velocity
η	=	solar sail reflective efficiency
ρ	=	density, kg/m^3
σ_s	=	solar sail assembly loading, g/m^2

I. Introduction

AS THE Voyager spacecraft continue toward interstellar space, and with the launch of the New Horizons spacecraft in January 2006, the concept of a dedicated mission for the in situ exploration of the Heliopause boundary and interstellar space, up to 200 AU, is reemerging within the science community. It is important to define and prepare critical technologies far in advance of novel science missions, ensuring the technologies are developed in a timely manner and that associated cost, risk, and feasibility of potential future mission concepts can be properly estimated. Herein, the development of a set of requirements for the technology development of a solar sail Interstellar Heliopause Probe (IHP) mission are considered far before one is proposed by the scientific community. The IHP mission and the technology requirements are thereafter placed in the context of other outer solar systems missions, ranging from a Kuiper belt mission to an Oort cloud mission.

Investigation of the Heliopause boundary would expand knowledge of the nature of the interstellar medium and its interaction with the solar wind. Such a mission would also be a precursor to the migration of deep space exploration beyond the solar system through the Oort cloud and on into true interstellar space. An IHP mission is from a trajectory, propulsion, and spacecraft systems viewpoint, which is a demanding endeavor to achieve within a reasonable timescale. Long cruise times must be endured, with the thermal environment ranging from close solar passes to deep space. Sufficient power generation and telecommunications downlink data rates at extreme distances from the sun are also crucial, as is science-driven autonomy.

A significant quantity of work in the past decade has been performed to assess the problem of trajectory design of an Interstellar Heliopause Probe, with a particular focus of this work looking at the application of solar sail technology and the design of solar sail trajectories [1–14]. Additionally, the development of solar sail technology for outer solar system applications has been considered [15–17]. Trajectory design for alternative propulsion systems such as solar and nuclear electric propulsion (SEP and NEP) has also been considered [18,19]. Latterly, in addition to trajectory and propulsion considerations, effort has begun to include the systems design of interstellar probes, with a significant amount of effort considering SEP and NEP missions [19–24]. Additionally, two independent solar-sail-based holistic mission studies are found within the literature: one directed from NASA Jet Propulsion Laboratory (JPL) [25–27] and one from ESA European Space Research and Technology Centre (ESTEC) [28–30]. It is of interest to note that the NASA-JPL-directed IHP mission attempts to gain a solar distance of 200 AU in 15 years, concluding that a spin-stabilized disc solar sail is required for such a requirement. Meanwhile, the ESA-ESTEC-directed study attempts to gain a solar distance of 200 AU in 25 years,

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concluding that a three-axis-stabilized square solar sail can satisfy such a requirement [30]. Noting that the IHP mission is often highlighted as an exemplar far-term solar sail mission [31], the disparity in required sail architecture is of critical importance to the development of solar sail technology. It is highly unlikely that even a successful midterm solar sailing mission such as Solar Polar Orbiter (SPO) [32], using a three-axis-stabilized square solar sail architecture, would provide much, if any, confidence to then progress to a spin-stabilized disc solar sail architecture for an IHP mission. As such, for a far-term solar sail mission such as IHP to be enabled, the preceding missions [31–34] must act as enabling facilitators and must therefore develop the sail architecture required for far-term missions such as IHP.

This paper, for the first time, considers the IHP mission in context with other solar sail missions in the outer solar system. The paper begins by considering the IHP mission and attempting to resolve the apparent discrepancy over the required sail architecture while determining the key technology development requirements to enable the mission. Subsequently, this paper considers missions to the two other primary sites of scientific interest beyond the orbit of Neptune, the Kuiper belt and the solar gravitational lensing effect at 550 AU, in order to place the IHP mission into context. Finally, this paper briefly considers a mission to the Oort cloud, which is the final region of scientific interest within our solar system and the definitive demarcation between our solar system and true interstellar space.

II. Science Objectives of the Interstellar Heliopause Probe

The scientific objectives of any IHP mission have been covered in detail through previous studies [25–30]; however, a brief discussion of the Heliopause region is included here for context. The objective of the IHP is to explore the local interstellar medium and the interaction region with the Heliosphere, known as the Heliopause. The Heliopause is the boundary between the outward flowing solar wind (hot ionized gas $\sim 10^5$ K) and the relatively cool interstellar medium of partially ionized interstellar gas [35]. The Heliosphere has a bow shock toward the solar apex, where the Heliosphere moves through the interstellar medium at a velocity on the order of 26 km/s relative to the local interstellar cloud. This bow shock is expected to lie at a distance of between 110 and 200 AU, slightly beyond the Heliopause and before the local interstellar medium, but the actual location is somewhat uncertain. The apex of the bow shock, sometimes defined as the nose of the Heliopause, is located at an ecliptic longitude of 254.5° and latitude of 7.5° [35]. Note that an inner shell is expected to be formed at 80–100 AU by the solar wind terminal shock, which is a predicted standing shock wave formed by the slowing of the supersonic solar wind to a subsonic flow [35,36]. The primary scientific goals of the considered IHP mission are to investigate the influence of the interstellar medium on the solar system (and vice versa), to explore the nature of the interstellar medium and the outer solar system, and to take measurements of the Heliopause region using field and particle instruments.

It is worth noting that a further area of interest is that of organic material abundance in the Heliopause region. Organic material is found in both the solar system and in the interstellar medium, but it is not known if these nonterrestrial organic materials have a similar origin. The IHP could also be used to investigate the phenomenon known as the hydrogen wall [37]: a pileup effect of neutral hydrogen atoms caused by charge exchange collisions, leading to weak coupling between the neutral and ionized hydrogen in the interstellar medium.

III. Mission Requirements and Architecture of the Interstellar Heliopause Probe

The IHP mission requirements and goals are detailed in Table 1 and are largely derived from [28–30]. It is shown that the mission requirements are tightly constrained, with mission duration, spacecraft mass, and launch vehicle predetermined. It is of note that the upper limit on mission duration is 10 years longer than that considered by the NASA-JPL studies [25–27]. The upper feasible limit on mission duration is difficult to quantify; for example, the Voyager spacecraft remain operational over three decades since launch, yet the primary missions of these spacecraft were approximately 3 and 12 years for Voyager 1 and Voyager 2, respectively. However, both spacecraft have continued to provide scientifically interesting data and, as such, operations have continued. As per requirement 5 in Table 1, the IHP mission will provide continuous science data from 5 AU onward; thus, it is anticipated that the spacecraft will provide scientifically interesting data from an early stage. However, the primary goal of the mission is measurement of the interstellar medium, which therefore necessitates a funding commitment over a much longer period than originally envisaged for the Voyager spacecraft.

Within the tight mission requirements detailed in Table 1, several potential solar sail mission architectures can be envisaged. Requirement 5 in Table 1 is derived from the desire that science data collection should cover a solar radius that is as large as possible; therefore, the sail will be jettisoned at 5 AU. It should be noted, however, that by maintaining control of the sail after jettison, perhaps through a secondary sail-specific onboard computer, it may be possible to infer the density of the local medium from observations of the sail deceleration.

Four top-level solar sail mission architectures are detailed in Table 2; it is noted that a close solar approach of less than 0.25 AU is generally considered to be very challenging [1–17]. Conventional polyimide sail substrates (thermal limit $\sim$520 K [1]) with conventional aluminum and chromium coatings will not be able to survive the local environment at less than 0.25 AU for any significant duration. Hence, a solar approach of less than 0.25 AU is highly unadvisable due to the large thermal loads placed on both the sail and the spacecraft system.

The proposed mission architectures are broadly similar; however, the variations between each are significant, and each mission merits closer examination before definition of the selected solar sail solution.

With IHP_1 multiple photonic assists, the architecture is envisaged to be suitable for a three-axis-stabilized square solar sail, with multiple photonic assists used to attain the requirements and goals defined in Table 2. The use of a square sail is important to consider, as this sail architecture has formed the focus of much of the recent solar sail hardware development by both ESA [38–40] and NASA [41–47]. It is noted that the hybrid solar sail SEP IKAROS project is based on a spinning square sail architecture [48] and could be a critical stepping stone toward the development of future pure solar sail technology. An advantage of a square solar sail is the likely availability of high turn rates during the close solar pass (also termed

Table 1 IHP mission requirements and goals

Identifier	Objective	Requirement/goal
1	IHP shall traverse 200 AU within 25 years of launch.	Requirement
2	IHP shall travel through the nose of the Heliopause (7.5° lat 254.5° long.).	Requirement
3	IHP shall be launched on a vehicle of similar capabilities to the Soyuz-Fregat.	Requirement
4	IHP shall provide a telemetry rate of 200 bps from 200 AU.	Requirement
5	IHP shall continuously collect science data from 5 AU outward.	Requirement
6	IHP will have a dry mass of 200 kg at 200 AU.	Goal
7	IHP instrument payload suite will be 15–20 kg.	Goal

Table 2 Solar sail IHP mission architectures

Mission	Name	Architecture
IHP_1	Multiphotonic assist IHP	Use a low-acceleration square sail and multiple photonic assists and/or a Jupiter gravity assist.
IHP_2	Single photonic assist IHP	Use a higher-acceleration sail to allow only a single photonic assist and a close solar approach thermally limited to 0.25 AU.
IHP_3	Fast IHP	Use a very-high-acceleration sail, a single photonic assist, and a close solar approach thermally limited to 0.25 AU to reach 200 AU in $\ll$ 25 years.
IHP_4	Sail and chemical IHP	Use positive launch C_3 and gravity assists, if required, to reach 0.25 AU, before sail deployment at the minimum solar radius.

a solar photonic assist). A final consideration is the effect of multiple photonic assists, which may lead to increased optical surface degradation, although the impact of this is unclear [49]. The three-axis-stabilized square solar sail is consider to be a lower-performance sail, due to the large structural mass required to support the sail reflective surface, but which is negated through spin-induced forces on a spinning sail.

With IHP_2 single photonic assist, the architecture is envisaged to be suitable for a modest spinning-disc solar sail, using a single photonic assist to reduce optical surface degradation over IHP_1. The likely drawbacks of this mission architecture are the relatively immature nature of spinning-disc solar sail hardware and the reduced maneuverability of such a sail.

With IHP_3 fast, the architecture is very similar to the one considered by NASA-JPL [25–27]. It will require a larger sail than IHP_2 but will reduce the mission duration and hence operations cost. A notable drawback of this mission architecture is the increased Heliopause boundary transit velocity, which will negatively affect science data collection.

The novel IHP_4 sail and chemical architecture is envisaged to consider the use of excess launch C_3 and unpowered gravity assists to reach 0.25 AU before sail deployment, hence maximizing the impact of solar sail propulsion while minimizing any impact of severe thermal aging on the sail performance. This architecture is analogous to an all-chemical mission, with the final large chemical kick stage replaced with a solar sail and with the required sail performance investigated to see if any advantage can be achieved over other mission architectures.

IV. Trajectory Design for the Interstellar Heliopause Probe

Recall from requirements 1 and 2 that the IHP is required to reach a solar radius of 200 AU in 25 years and pass through the nose of the Heliopause at a longitude/azimuth of 254.5° and a latitude/elevation of 7.5°. As previously stated, a vast quantity of work has been performed considering the design of solar sail trajectories for the IHP mission [1–14]. However, it is difficult to gain clear conclusions from this previous work, due to variations in trajectory design (i.e., minimum allowable solar approach) and used solar sail propulsion model [1,49]. As such, a detailed parametric analysis is required to allow consideration of each of the mission architectures identified previously.

The NASA-JPL solar sail interstellar mission studies have focused largely on the use of high-performance spin-stabilized disc solar sails, exhibiting a characteristic acceleration of approximately 3 mm/s^2 [5,25–27]. A number of novel fast heliopause trajectories have also been presented, using a strategy of orbital angular momentum reversal; however, this is not considered to be necessary for the IHP mission, due to the level of sail performance required to enable such a trajectory [4]. Note that parametric studies of multiple photonic assists have also been conducted by NASA-JPL for missions to Pluto and the Kuiper belt, which could be extended to approximate the Heliopause scenario [7].

Recent trajectories considered by Leipold et al. [15] and ESA-ESTEC have considered lower-performance sails to achieve a slower transit to 200 AU with a conventional three-axis-stabilized square sail, with composite booms derived from DLR, German Aerospace Center activities [28–30]. In particular, a 21.2-year, 0.75 mm/s^2 ideal sail trajectory has been presented that spirals down to 0.37 AU and then executes a dual photonic assist; however, the closest solar approach is just 0.1 AU, placing severe thermal loads on the spacecraft and sail assembly [15]. The mission scenario considered herein shall use the sail from Earth departure until it is jettisoned at 5 AU, as per requirement 5, and shall apply a strict solar approach limit of 0.25 AU due to the thermal effects on the sail, as discussed previously. Furthermore, due to thermal cycling issues, when possible, the trajectory design shall strive toward a single photonic assist to minimize optical surface degradation. Trajectories presented within this paper were developed using a sequential quadratic programming algorithm, NPOPT 6.2 [50], along with the accessibility and deficit, or A^nD, blending method of locally optimal control laws [14,51].

A. Parametric Trajectory Analysis

For any level of sail performance, the solar-photonic-assist trajectory is optimized by minimizing the solar approach and maximizing the corresponding instantaneous radius of aphelion [14]. As such, any portion of the trajectory before the final close solar approach is simply seeking to maximize the instantaneous radius of aphelion while delivering the sail to the final close solar approach in as short of a time as possible.

A parametric analysis was conducted through an approach of optimizing half-arcs from the final perihelion to 5 AU, as per requirement 5. In this way, the effect of final close approach with low characteristic accelerations on trip time can be rapidly ascertained. Fixing the mission duration allows the available time to reach the final perihelion to be determined and the mission feasibility to be assessed. Figure 1 shows the time to 200 AU from a range of final close-approach radii at corresponding instantaneous aphelion values of 2 and 5 AU. Figure 1 was determined using a locally optimal energy gain control law that has been shown to give results very close to global optimal for such a scenario [14,51]. However, it should be noted that for this preliminary analysis, the trajectory is simulated in two dimensions with an open final escape asymptote; such a simplification can result in a significant deviation from the time-optimal, fully constrained, three-dimensional trajectory and, as such, is only valid as an initial analysis [14].

It is noted that the perihelion velocity is relatively insensitive to the corresponding instantaneous radius of aphelion when perihelion lies in the range of 0.05 to 0.4 AU and aphelion is from 1 to 5 AU, with perihelion velocity varying by only $\pm 6\%$. However, as shown in Fig. 1, significant variations can occur once the trajectory is propagated to 200 AU, particularly at lower levels of sail performance and increased values of final close-approach radius. For example, an ideal sail providing a characteristic acceleration of 0.5 mm/s^2, with final solar close approach limited to 0.25 AU will take approximately 175 years to reach 200 AU if the instantaneous radius of aphelion was 2 AU at the final solar close approach. However, if the instantaneous radius of aphelion is increased to 5 AU, the time to 200 AU is reduced to approximately 43 years: a 75% reduction. Meanwhile, the variation for an ideal sail, providing a characteristic acceleration of 1.5 mm/s^2, drops by only 13%. It is noted that using the velocity at 5 AU to approximate the trip time to 200 AU, assuming a constant velocity, is similarly flawed, with an error in the two previous scenarios of 260 and 10%. Figure 1 allows

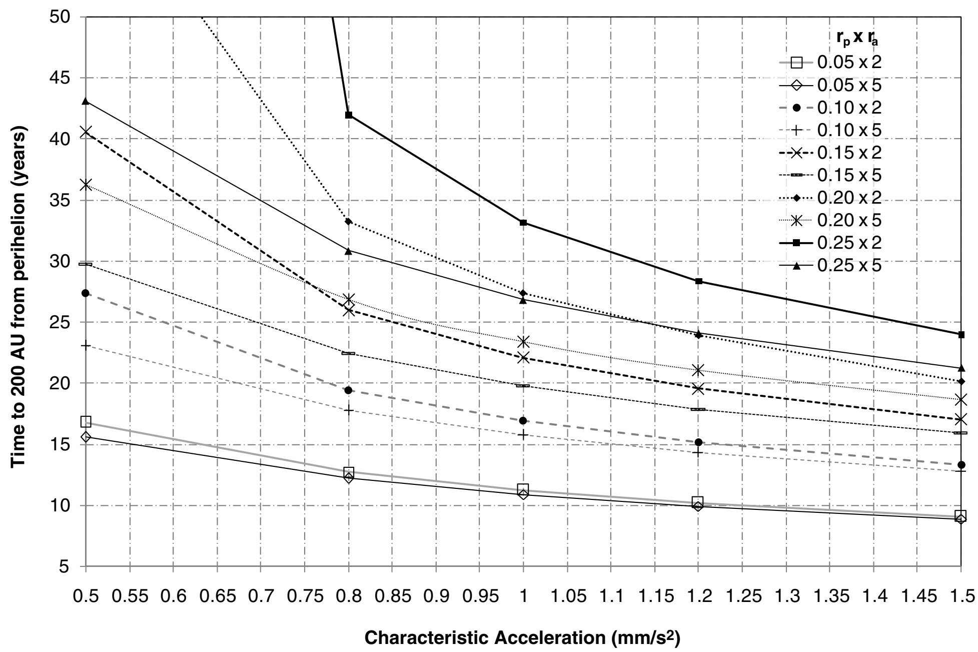

Fig. 1 Closest approach to 200 AU half-arc trajectory analyses.

determination of the available time to attain the final close solar approach for any given level of sail performance between 0.5 and 1.5 mm/s^2 for missions to 200 AU. Note that Fig. 1 assumes an ideal sail as defined in [1]; a nonideal sail can add between 5 and 10% to the trip time [10]. From Fig. 1, a 1.0 mm/s^2 sail with a minimum close solar approach of 0.15 AU is required to reach close solar approach with a corresponding instantaneous radius of aphelion of 2 or 5 AU in approximately 1 or 5 years, respectively.

B. Architecture Tradeoff

Recall that within this mission scenario a strict thermal limit of 0.25 AU is applied. Thus, from Fig. 1 a 25-year half-arc trip time to 200 AU with an instantaneous radius of aphelion of 2 or 5 AU requires a sail characteristic acceleration of 1.44 or 1.14 mm/s^2, respectively. Hence, sail characteristic accelerations above these values will be required to attain 200 AU in less than 25 years from these orbits. For example, a sail characteristic acceleration of 1.3 mm/s^2 provides approximately -2 or $+2.5$ years to reach a close solar approach with a corresponding instantaneous radius of aphelion of 2 or 5 AU, respectively. Increasing sail performance to 1.5 mm/s^2 provides approximately 1 to 4 years to reach a close solar approach, depending on the corresponding instantaneous radius of aphelion. Therefore, with a perihelion limit of 0.25 AU, a characteristic acceleration on the order of 1.5 mm/s^2 appears necessary to meet requirement 1. This conclusion appears to contradict previous findings [30], which state that a solar sail with characteristic acceleration of 1.1 mm/s^2 can reach 200 AU in 25 years, with a minimum solar approach of 0.25 AU. From Fig. 1 it is noted, however, that such a solar sail would be required to reach the final close solar approach in between -1 and -6 years to enable a 200 AU trip in 25 years. However, if an instantaneous radius of aphelion of 5 AU at close solar approach is assumed, along with constant velocity from 5 to 200 AU, then a period of 2.6 years is available to reach the final close solar approach. Similarly, increasing the instantaneous radius of aphelion to 10 AU and making the same assumption provides a period of 3.6 years. As such, it appears possible that this previous work is based on an invalid assumption within the trajectory design.

The IHP_3 architecture will clearly require a higher-performance sail than IHP_2 to enable a reduced trip time to 200 AU. As found in previous studies, it was found that a sail characteristic acceleration of 3 mm/s^2 is required for a 15-year trip to 200 AU.

The mission architecture considered in IHP_4 is highly novel and has not been previously considered within the literature. Within this architecture, the use of gravity assists and chemical propulsion was considered to reach the close solar approach before sail deployment, hence minimizing sail performance degradation before the vital solar photonic assist. Restricting the selection of the launch vehicle to a Soyuz-Fregat, as per requirement 3, it was found that using only unpowered Jupiter gravity assists would deliver the probe to 0.25 AU in approximately 9 years. Hence, as shown in Fig. 1, a very-high-performance sail would be required to then reach 200 AU within the 25-year target. It was found that a powered gravity assist at Venus ($\Delta v = 4.24$ km/s) and a second at Jupiter ($\Delta v = 0.91$ km/s) can enable an IHP mission with a sail characteristic acceleration of 2 mm/s. However, to gain a sufficient launch mass (assumed at 300 kg) with the Soyuz-Fregat, it was found that the kick stage must have a specific impulse on the order of 500 s. It is thus concluded that mission architecture IHP_4 can only enhance the IHP mission if requirement 3 is relaxed and a larger launch vehicle used.

C. Reference Trajectory

The primary technology driver of the IHP mission concept is clearly the solar sail. Therefore, mission architecture IHP_2 was selected to minimize the technology requirements of the solar sail.

Using the A^nD blending method of locally optimal control laws [14,51], a launch-date scan was performed for an ideal sail with characteristic acceleration 1.5 mm/s^2. It should be noted that the A^nD method produces results that are typically within 2–3% of the global optimal [14], which will offset much of the anticipated error due to the assumption of an ideal sail with no optical surface degradation [10]. The launch-date scan is presented in Fig. 2 for the year 2030; however, it should be noted that the optimal launch will reoccur annually, and hence this launch date is selected simply to facilitate trajectory analysis. The structure of the trajectory is a simple single solar photonic assist, as shown in Fig. 3, in which each of the trajectories within Fig. 2 is shown. It is noted from Figs. 2 and 3 that as the launch date moves through the period considered, the required radius of aphelion passage decreases. It is anticipated that at some point a discontinuity will occur, resulting in the radius of aphelion passage increasing to a much larger value before decreasing in a fashion very similar to that shown in Figs. 2 and 3. It is determined that the optimal launch date for an ideal sail with characteristic acceleration 1.5 mm/s^2 is 26 January, reaching 200 AU in 23.17 years; the mission timeline is detailed in Table 3.

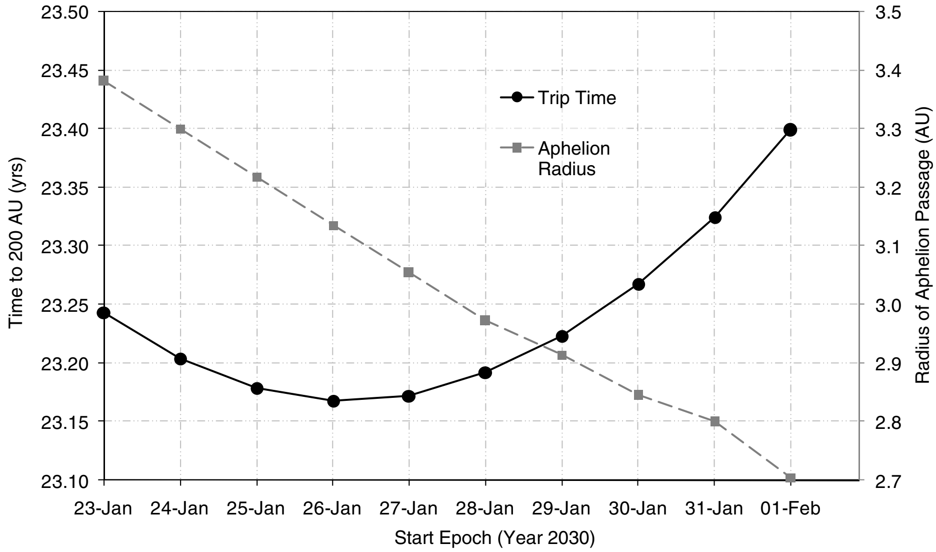

Fig. 2 Launch-date scan from 23 January until 01 February.

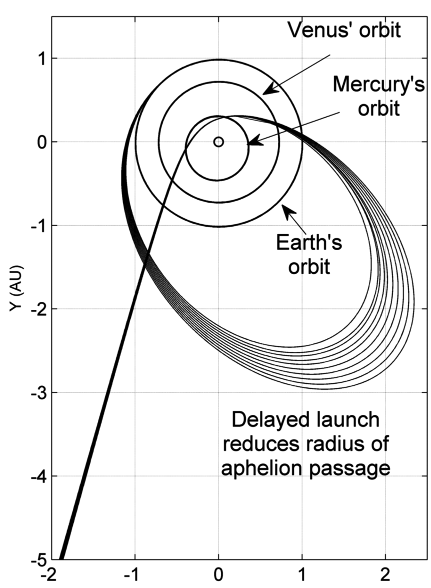

Fig. 3 Trajectory plot over launch-date scan from 23 January until 01 February, one trajectory per day.

V. System Design for the Interstellar Heliopause Probe

The focus of many previous solar sail IHP studies has been trajectory and sail design rather than system design. However, both the previous NASA-JPL [25–27] and ESA-ESTEC [28–30] studies have touched on the issue through traditional system design methodologies. At this stage, it should be noted that within this paper the solar sail is simply considered a subsystem of the spacecraft, whereas the spacecraft is traditionally termed a payload of the sail. To identify the critical technology drivers for the IHP mission, this study adopted a method of integrated statistical regression analysis allowing the system design of the sail and the probe to be based on technology trends and for system interdependencies to be modeled through iteratively solved interdependencies. The integrated nature of the developed methodology allows key technology drivers and requirements to be identified more accurately and novel sensitivity analysis to be performed: for example, solar sail size or nominal mass versus radioisotope thermoelectric generator (RTG) power density or data rate from 200 AU. It is this novel and integrated approach to system design that identified a critical turning point in required sail size for the SPO mission [32] and that is used to minimize the advancement degree of difficulty (AD2) [33] of the mission being considered. For example, the most significant technology requirement for the IHP mission is clearly the solar sail that has a high AD2; thus, if the solar sail technology requirements can be reduced by increasing the demand on other, more mature, technologies that have a lower AD2, then the overall mission AD2 can be reduced.

Throughout the spacecraft design, full redundancy is maintained wherever possible; however, systems such as the high-gain antenna and many solar sail components such as the sail film cannot reasonably be supported by a backup system, due to mass and/or volume constraints. The issue of systems redundancy on long-duration missions is of critical importance and some significant work has recently been done in this area [52,53]. An overview of the spacecraft mass budget, based on a minimum solar approach of 0.25 AU, is presented in Table 4, in which it is shown that the launch mass is 615 kg, providing a Soyuz-Fregat 2-1b launch-mass margin of 1404 kg. It should be noted that for a solar sail system design, unlike conventional system design, a large launch-mass margin does not mean that the technology demands of the mission can be reduced simply by allowing the use of heavier components to fill up the launch vehicle. As the launch-mass margin is reduced, the solar sail size will increase and hence the mission AD2 can actually be increased; therefore, the goal of solar sail system design is to minimize the mission AD2, which is typically achieved by minimizing the solar sail size. Table 4 gives the current best estimate (CBE) mass, which then has a design maturity margin (DMM) added to give the total subsystem mass allocation. The DMM is added at the equipment level, where greater than 5% is added for off-the-shelf items [European Cooperation for Space Standardisation (ECSS), category A/B], greater than 10% is added for off-the shelf items requiring minor modifications (ECSS category C), and greater than 20% is added for new design/development items, or items requiring major modifications or redesign (ECSS category D). Note in Table 4 that the added DMM can appear to be somewhat arbitrary. For

Table 3 IHP reference mission timeline for 26 January launch

Event	Time
Launch	T0
Aphelion passage	T0 + 1.5 yr
Perihelion passage	T0 + 2.8 yr
Sail jettison (5 AU)	T0 + 3.2 yr
Kuiper belt transit (40–55 AU)	T0 + 5.7–8.3 yr
100 AU	T0 + 12.9 yr
200 AU	T0 + 23.2 yr

Table 4 Mass budget, based on integrated statistical regression analysis

System	CBE mass, kg	DMM, %	CBE mass plus DMM, kg
Science instruments	25	0	25
AOCS	6	5	7
TT&C	55	8	62
Onboard data-handling system	1.5	10	2
Thermal and radiation	14	10	16
POWER	50	11	56
Mechanism and structure	27	15	31
Probe nominal dry mass	——	——	199
AOCS propellant, including sail separation allowance	2.7	5	2.8
Probe nominal wet mass	——	——	202
Solar sail flight components (see Table 5)	83	33	110
Solar sail jettisoned components, inc. Deployment module (see Table 5)	163	18	201
Solar sail nominal mass	——	——	311
Nominal spacecraft launch mass	——	——	513
System-level margin at 20%	——	——	102
Total mass at launch	——	——	614
Soyuz-Fregat 2-1b launch capacity	——	——	2020
Launch margin	——	——	1405 (70%)

example, the power subsystem DMM is 11%; however, this is simply a result of averaging the DMMs allocated at the equipment level.

A 25 kg allocation is provided for science instruments, which will most likely include a magnetometer. Hence, the spacecraft magnetic field must be low (preferably less than 10^{-9} T), known, and constant so that the science data can be corrected. From Table 4 it is shown that the probe wet mass is 202 kg; this is the portion of the spacecraft that will perform the science operations of the mission once the sail has been jettisoned.

The science instruments require a spinning spacecraft; thus, after sail jettison, the probe will continue to spin. The primary attitude and orbit control system (AOCS) driver is the Ka-band high-gain antenna (HGA), which (due to the high frequency of the Ka-band) suffers from high losses at pointing accuracy of worse than 180 arc seconds. The onboard telemetry, tracking, and command (TT&C) system contains two Ka-band medium-gain antennas (MGAs), with a required pointing accuracy of 5400 arc seconds. The MGAs are used for all communications during the period when the sail is attached to the IHP and for which a 10 m central cutout is provided in the sail surface. After sail jettison the HGA is used for primary communications. Therefore, during the mission phase when the IHP is attached to the solar sail, pointing requirements are defined as 5400 arc seconds, which is achieved through the use of sun sensors. Star sensors are subsequently used once the sail is jettisoned and a much higher pointing accuracy is required. Pointing accuracy is maintained by a small cold-gas system once the sail has been jettisoned and by the sail before this. Note that a low-gain S-band system is also included to provide robust communications before, and immediately after, sail deployment. The 2.5 m Ka-band HGA mass was found to be a key design driver due to a lack of sufficient data points on which to base the regression analysis. Many small HGA antennas have been flown, such that the HGA mass up to 2 m can be confidently projected. However, beyond this, only a limited data set exists from which to project. Using all the available data, an exponential mass curve was developed that allows projection with some confidence up to a limit of 2.5 m. This curve suggests an HGA mass of approximately 32 kg. A secondary curve was then derived from this to project the mass of a more advanced lightweight composite HGA. This lightweight HGA was projected at approximately 24 kg: a 25% mass savings. The development of such a lightweight composite HGA was determined to be a critical mission requirement; however, it should also be noted that the mass and power requirements of the Ka-band traveling-wave tube amplifier within the TT&C system are also of critical concern, with the current output power of 75 W toward the upper end of current technology.

The TT&C subsystem is a critical mission driver and further study is required. Specifically, the use of optical communications should be investigated, which would require a considerable improvement in pointing accuracy. Additionally, issues such as the acquisition strategy would have to be readdressed to account for the long slant ranges involved. The light travel time to 200 AU is 26 h; hence, using a beacon system, as employed currently, would not be suitable over such long slant ranges. This is an issue that must be addressed for all outer solar system missions when considering optical communications. In addition to acquisition strategy, lightweight components and long-lifetime lasers would need to be developed to make optical communications viable for the IHP mission. For these reasons, a radio frequency link was assumed within this study; however, a more in-depth communications analysis is required before final architecture selection.

A. Solar Sail Subsystem Design

The primary function of the solar sail subsystem is to deploy, support, and control a large lightweight reflective surface that will impart a momentum onto the complete spacecraft assembly due to incident and reflected solar photons. Following optics-based reflectivity models, it has previously been assumed that the sail film must be held in high tension to ensure maximum reflectivity; such models assume symmetry about the surface normal beyond ± 6–$10°$ with a perturbed Lambertian distribution, seeking to achieve a specular reflection. Thus, scattered photons are shown as being of no use. A solar sail can, however, gain thrust from all reflected photons, diffusely scattered or not. Indeed, prior experimentation by NASA-JPL has shown that aluminum can provide a reflectivity of 0.88–0.90 for flatness ± 4.0, thus requiring only low film tension [54]. Furthermore, recent testing has shown that even heavily wrinkled (aluminum-coated) material retains 88% reflectivity under zero tension [55]. As such, if the sail film is held in high tension, it is highly likely that excess mass is being deployed without adequate compensation through increased reflectivity.

It was determined previously that a characteristic acceleration of 1.5 $\mathrm{mm/s^2}$ is required to attain the mission goals within the defined requirements. The mass fraction of the spacecraft that is not the solar sail subsystem is

$$m_{sf} = \left[1 - \left(\frac{a_s \sigma_s}{2\eta P}\right)\right] \tag{1}$$

Therefore, the required sail surface area for a given probe mass m_p can be determined from

$$A = \left[m_p\left(\frac{1}{m_{sf}} - 1\right)\right] \Big/ \sigma_s \tag{2}$$

It is shown from Eq. (1) that as the sail assembly loading is increased, the mass fraction of the spacecraft that is not the solar sail subsystem decreases. Therefore, from Eq. (2) it is noted that as the sail assembly

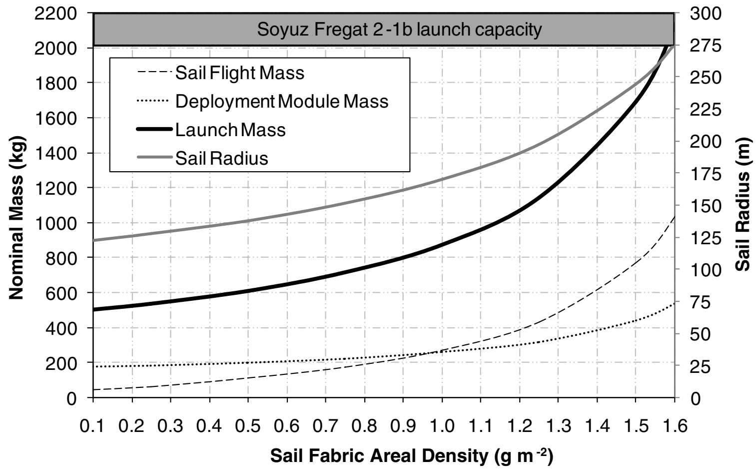

Fig. 4 Effect of variations in sail areal density on sail size and mission mass budget.

loading is increased, the required sail reflective-surface area will tend toward infinity, which occurs at a sail assembly loading of just over 5.168 g/m^2 in the scenario of this study. Clearly, a solar sail reflective-surface area of infinity is not feasible. Therefore, initially considering a three-axis-stabilized square solar, it is found that for a side length of less than 300 m, the sail assembly loading must be less than 3 g/m^2, which is very challenging [1] for a thermal limit of 0.25 AU. Because of the clear difficulty of developing a three-axis-stabilized square solar sail, a spinning-disc sail is considered, based on the NASA-JPL design [25–27,56]. The spinning-disc sail is scaled based on the other system masses, sail size, and moment of inertia, while maintaining a constant sail fabric tension of 6925 Pa ($\sim$1 psi). The absence of booms from the spinning-disc sail allows for a significant reduction in sail assembly loading while allowing the design to be generated without a significant level of prior technical data, enabling definition of technical requirements. The solar sail proposed for the NASA Interstellar Probe Mission is 410 m in diameter, with an assembly loading of 1 g/m^2. While there is nothing technically wrong with such a design, it is considered to be quite aggressive, due to the advanced nature of the required technologies along with the required sail size to achieve the mission goals in 15 years. Hence, design margins are added within this study, increasing the sail assembly loading to more attainable midterm values and reducing the overall mission AD2. A spinning-disc sail is not only tensioned by centripetal forces when fully deployed, but it must also be deployed using these spin-induced forces, due to the absence of structural elements that add mass, such as stiff booms. As such, a dedicated deployment module employing cold-gas thrusters mounted on 2.5 m booms to provide adequate torque for spinup is required, which is jettisoned following sail deployment. Control of the spinning-disc sail is accomplished by sliding the centrally mounted payload along short rails to offset the center of mass with respect to the center of pressure, causing the sail spin axis to precess; this could also be accomplished by using a gimbaled boom.

From Fig. 4, the sail areal density must be less than 1.6 g/m^2 to fit within the capacity of the launch vehicle; however, this leaves zero margin for growth within other systems. It is also noted that at an areal density of 1.0 g/m^2, the mass of the deployment system exceeds that of the sail. Thus, from this sail design, a technology requirement of a sail film areal density of 0.5 g/m^2 is derived to ensure a sufficient technology margin for other sail systems and to ensure that the sail size does not become a technology driver. The solar sail system in Tables 4 and 5 and Fig. 4 corresponds to a sail with radius 139 m, a center cutoff of 10 m to allow a 4π sr field of view from the spacecraft, and an assembly loading of 1.8 g/m^2. A detailed breakdown of the sail and its deployment system mass is given in Table 5, in which it is noted that the design point falls within the range of prior spinning-sail design studies.

B. Power

The design maximum average power load was found to be during telecommunication downlink periods, although science acquisition continues. The end-of-life (EOL) design value was found to be 266 W, including design maturity margin and a 20% system-level margin. To generate sufficient power, a RTG-based system was specified. The EOL power design value corresponds to a beginning-of-life (BOL) requirement of 325 W, assuming a plutonium-238 half-life of 86.4 years. Using this BOL power load, the effect of RTG power density on IHP mass, sail size, sail mass, and total launch mass can be analyzed. The system design assumes that excess heat from the nuclear source can be used for thermal control. From Fig. 5 it is

Table 5 Mass budget of solar sail based on integrated statistical regression analysis

System	CBE mass, kg	DMM, %	CBE mass plus DMM, kg
Sail fabric mass (areal density of 0.5 g/m^2)	30	30	39
Seaming tapes, gore reinforcements (20% of sail fabric mass)	6	30	8
Cylinders, brackets, braces, rollers (25% of sail fabric mass)	8	30	10
Rim and tip masses	12	30	15
Yoke plates, tether locks (fixed mass of 6 yoke plates and 6 locks)	9	30	12
Gore deployment tethers (Kevlar tether radius of 0.5 mm)	3	40	4
Sail c.g./c.p. offset system	5	30	7
Misc. structure (10% of total sail mass)	10	40	15
Solar sail flight components	——	——	110
2.5 m booms for reaction control system thrusters moment arm	4.5	15	5
Discarded sail structure (areal density of 1 g/m^2)	60	40	85
Cold-gas spinup system	33	10	37
Primary structure	30	15	34
Launch release mechanism	15	15	17
Launch vehicle interface (4% of nominal launch mass)	20	15	23
Solar sail jettisoned components	——	——	201

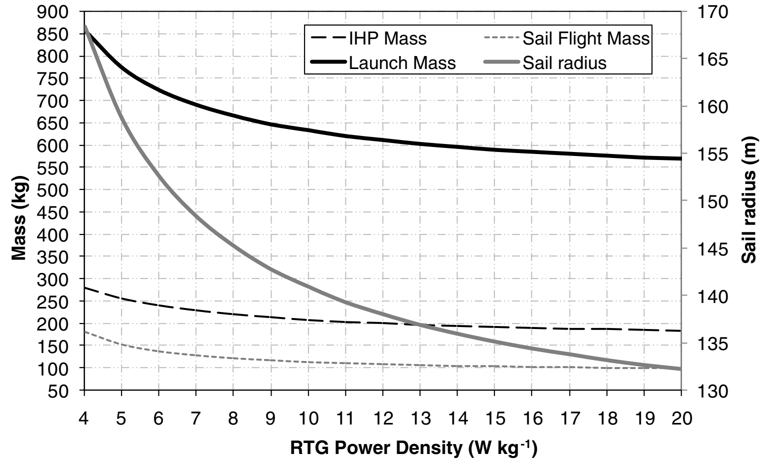

Fig. 5 Effect of variations in RTG power density on sail size and mission mass budget.

shown that RTG power density does not have a notable impact on sail size. For a conventional RTG of the type used on Cassini or Galileo, the sail radius is relatively modest, at only 160 m for a 5 W/kg RTG. However, the required IHP mass is 255 kg, which is considerably above the goal mass. A power density of approximately 12 W/kg is required is to reduce the IHP mass toward the goal value, requiring a sail radius of approximately 138 m. It is noted, however, that between 10–15 W/kg the gradient of sail radius decreases such that it can be stated for the design power load that a power density of above 12 W/kg is not required, as it will provide only modest benefits for the sail sizing. Current RTGs are approximately 7%-efficient, delivering about 5 W/kg [57]. The required power density corresponds to an advanced radioisotope power source based on ~21%-efficient alkali–metal thermoelectric converter cells using a two-brick general-purpose heat source.

VI. Potential Mission Architecture Adaptations

Originally, the mission architecture was divided into four options; however, on completion of the mission study it is prudent to review these architectures and consider possible adaptations. While noting the limited availability of such opportunities, a Jupiter gravity assist (JGA) was considered during the outgoing escape trajectory. It is immediately noted that the orbit inclination of Jupiter is 1.3 deg [58]; as such, the 7.5° latitude of the Heliopause nose needs to be attained entirely through the gravity assist to ensure compliance with requirement 2: i.e., that the probe travel through the nose of the Heliopause at 254.5° longitude. There are, of course, additional operational risks associated with a close Jupiter swingby, as well as the radiation issue of passing so close to Jupiter. Radiation exposure should be minimal, however, since the spacecraft will be traveling at high speed. The gravity assist was modeled as a simple patched-conic, assuming two-body dynamics, with Jupiter at 5.2 AU from the sun. The attainable Δv for pre-JGA velocities of between 5 and 15 AU/yr and for a range of perijove radii covering the mean distances of the Galilean moons was considered. It was found that using a JGA for fast Heliopause missions with characteristic accelerations greater than 1.5 mm/s^2 does not offer any significant reduction in trip time. However, low-performance-sail trajectories, which would ordinarily reach the Heliopause in over 25 years, would gain a modest benefit from a JGA. It is thus concluded that a JGA should not be considered further at this stage of design; however, an opportunistic flyby should not be ruled out.

From Table 4, a considerable launch-mass margin is available. Therefore, an investigation of the effect of using the associated considerable excess C_3 capacity was conducted. It can be anticipated that an excess C_3 would offer only a small benefit to IHP missions, enabling more rapid arrival at the close solar approach. For low-performance sails, for which multiple spirals are needed to reach the final close solar approach, it is anticipated that the excess C_3 may be used to reach the close solar approach in fewer solar revolutions. A launch excess energy of 20 km^2/s^2 was considered. It was found, as expected, that for low-performance sails the excess C_3 may be used to reach the close solar approach in fewer solar revolutions. However, the actual time savings before sail jettison was minimal in these cases, in which the primary benefit was actually a reduction in accumulated thermal loads. At the 5 AU sail jettison the cruise velocity for a 0.85 mm/s^2 solar sail was found to be reduced by 0.83% against the optimal zero-excess-energy case. Thus, as a result of the reduced time to sail jettison and the reduced cruise velocity, the complete mission duration was reduced by only 2%. Increasing the sail performance to a zero-excess-launch-energy single-revolution-level (i.e., a 1.5 mm/s^2) sail, it was found that the trajectory optimizer was unable to gain a close solar approach of 0.25 AU, instead using a close solar approach of 0.29 AU. Once again, the initial spiral down time was reduced; however, due to the increased close solar approach and inability of the sail (due to reduction this time) to increase the instantaneous radius of aphelion, the cruise velocity at sail jettison was found to be reduced by 29%, with the resultant trip time to 200 AU increased by 42%. This clearly illustrates the importance of minimizing the radius of close solar approach. It thus seems clear that once the sail performance reaches a level at which a zero-excess-launch-energy single-revolution trajectory is enabled, any additional energy must be sufficiently large to enable both a close solar approach and the instantaneous radius of aphelion to be sufficiently increased. Therefore, a terrestrial planet gravity assist would also not provide sufficient additional energy to benefit the mission architecture.

To confirm this result, a 3.0 mm/s^2 sail was considered using the available excess launch energy; it was found that at sail jettison, which occurred earlier than in the zero-excess-launch-energy scenario, the cruise velocity was reduced by almost 10%, with the trip time to 200 AU reduced by less than 5%. It is thus clear that excess launch energy is of little or no benefit within the mission scenario detailed within this paper; however, it can be of modest benefit at extremes of sail performance.

VII. Kuiper Belt Fly-Through Mission

Having considered a Heliopause mission, the less challenging scenario of a Kuiper belt (or Edgeworth–Kuiper belt) mission is briefly considered. Targeting a solar distance of between 30 and 55 AU, the Kuiper belt, in addition to being a scientifically worthwhile target in its own right, can reasonably be considered a technology pathfinder toward a Heliopause mission. For this mission, as previously, a system design using a method of integrated statistical regression analysis was performed that detailed a wet spacecraft mass, excluding solar sail, of 152.6 kg. It was found that to reach the orbit of Pluto within 15 years, a sail characteristic acceleration of 0.5 mm/s^2 was required. Thus, directly adapting the sail defined for the IHP mission, the required sail radius is 63.2 m, with a sail assembly loading of 3.3 g/m^2. The solar sail flight component mass was found to be 40.7 kg with a jettisonable mass of 121.4 kg, giving a total launch mass, including a final 20% system-level margin, of 377.6 kg. The mission launch mass is relatively low, enabling the low-cost Rockot launch vehicle to be used, further reducing mission cost from conventional missions that require much larger and more expensive launch vehicles. It should be noted that this mission can be performed by a three-axis-stabilized square sail of side length 130 m and assembly loading of 6.5 g/m^2; however, such a mission could not easily act as a technology pathfinder toward a Heliopause mission, due to the required sail performance of the later mission. The Kuiper belt mission timeline is detailed in Table 6.

VIII. Oort Cloud Fly-Through Mission

Beyond the Heliopause, the next scientifically notable destination is the solar gravitational lens at 550 AU. However, due to the need to maintain spacecraft position within 100 m of the gravitational lens in order to exploit it and the inability of a solar sail propulsion system to slow down a spacecraft within the outer solar system, this scientifically notable destination is not considered suitable for outer solar system exploration using solar radiation pressure as the sole means of propulsion. The next scientifically notable destination is the Oort cloud (alternatively, the Öpik–Oort cloud), although it is noted that no direct observations have been made to validate its existence. The Oort cloud is thought to extend from approximately 50–200,000 AU, or 0.79–3.16 light years, although an inner Oort cloud is also thought to possibly exist from 2–20,000 AU. The spherical shell of the Oort cloud is the ultimate boundary of the solar system: the edge of the sun's gravitational influence. Hence, the outer regions of the Oort cloud are only loosely bound to the solar system and are thus easily affected by the gravitational pull of passing stars and their own Oort cloud, occasionally dislodging comets from their orbits within the cloud and sending them toward the inner solar system as long-period comets. Assuming an isotropic and homogeneous cloud with a mean number density of between 1 and 10 AU^{-3}, the number of Oort cloud objects encountered was considered for a 30-year cruise (considered to be the upper reasonable limit) or a 100-year cruise (considered to be the absolute upper limit) to 50 000 AU. As the spacecraft travels through the Oort cloud, N objects would be detected within a radius R of the spacecraft, in time t. The volume of this cylinder through the Oort cloud is $[\pi R^2 v_\infty t]$, so the detection rate N/t can be obtained from

$$N = \pi R^2 v_\infty t \rho \tag{3}$$

Conversely, the search distance from the spacecraft to enable a desired detection rate is given by

$$R = \sqrt{\left(\frac{N}{t}\right)\frac{1}{\pi v_\infty \rho}} \tag{4}$$

The results of this analysis for a 100-year cruise to the Oort cloud show that one object per year may be detected within 200–600 Earth radii of the spacecraft, whereas for a faster 30-year cruise, one object per year may be detected within a closer range of 100–300 Earth radii of the spacecraft. As such, an Oort cloud mission can be expected to encounter a sufficient number of objects to be scientifically interesting.

Table 6 Kuiper belt mission timeline

Event	Time
Launch	T0
First perihelion passage	T0 + 0.5 yr
Second perihelion passage	T0 + 1.75 yr
Third perihelion passage	T0 + 4.6 yr
Arrival at Kuiper belt	T0 + 14.1 yr
End of primary mission at 50 AU	T0 + 20 yr
End of secondary mission at 100 AU	T0 + 36 yr

Oort cloud exploration is clearly an extremely challenging goal, especially when using solar radiation pressure alone to propel the spacecraft. Transit to such a large heliocentric distance necessitates either a long-duration mission on the order of 100 years or more or extremely high cruise speeds to provide reasonable mission durations. Considering a 100-year cruise to the Oort cloud requires the spacecraft to maintain a velocity of 500 AU/year, and a 30-year cruise requires the cruise speed to be increased to approximately 1666 AU/year (2.5% of light speed). These are clearly immense requirements to be imposed on the propulsion system.

Returning to the concept of the close solar photonic assist, the closer the perihelion of the trajectory, the greater the photon flux and thus thrust produced by the solar sail. However, the minimum distance from the sun, which from the IHP study was shown to be critical, is dependent on the thermal properties of the sail. The use of aluminum as a reflective layer limits the close-approach radius. Hence, alternative approaches are sought such as the use of beryllium or an all-carbon-mesh sail. The use of carbon reduces the sail mass, due to the low density of carbon, while extending the temperature envelope of the sail. It should be further noted that a carbon sail would be immune to ultraviolet degradation and does not cling or wrinkle, allowing simple elastic deployment from a rolled stowage configuration. Compared to polymer substrate sails, or an all-metal sail, carbon-mesh sails would be thick and highly porous. All-carbon sails could be manufactured from fibers or tubes, i.e., carbon nanotubes. Carbon nanotubes can be generated using chemical-vapor deposition, in which the decomposing of an organic gas over a substrate covered with metal catalyst particles produces nanotubes. The chemical-vapor deposition technique allows for the production of nanotubes on a much larger scale and is potentially a key to their use for solar sail applications. Furthermore, the nanotube configuration could potentially be finely tuned to allow a carbon nanotube sail to be optimized for specific wavelengths corresponding to peaks in the solar spectrum. Thermally, carbon-mesh sails could survive temperatures up to $\sim$3000 K (the melting point of carbon is 3800 K), before outgassing becomes excessive and sail degradation becomes an issue. Note that an element of outgassing could be used to provide a propulsive push at perihelion, although this would degrade sail performance for later multifunctional duties such as a parabolic antenna. If all-carbon sails were used, then high-performance carbon sails could approach as close as four solar radii to the sun's surface.

Bulk carbon has a reflectivity of 0.27 at a wavelength of 1000 nm,[§] which is beyond the visible wavelength range. Noting that as wavelength is reduced, the reflectivity typically reduces [for example, bulk molybdenum has reflectivity of 0.5737 at wavelength 1000 nm and reflectivity of 0.5667 at 546 nm (visible light) [59]], an all-carbon solar sail is estimated to have a reflectivity on the order of 0.25, which is significantly lower than an aluminum sail film. However, the increase in melting point allows a much closer solar approach radius and hence a much increased solar photonic assist. Use of doped-carbon nanotubes or a reflective coating can enhance reflectivity. However, carbon has the highest melting point of all the elements, and hence any added reflective coating would reduce the sail close-approach radius and would hence require a complex trade to be performed. For example, tungsten has the second-highest melting point of all elements, with the highest strength (of metals) at temperatures over 1900 K, and has a reflectivity of 0.46. Hence, a tungsten coating could notably increase sail reflectivity; however, tungsten has a density of $19,250\ kg/m^3$, and even a very thin tungsten layer would significantly increase sail mass, offsetting any benefits of increased reflectivity. Furthermore, tungsten does not sublime.

The simplest way of reaching four solar radii is a direct launch to a JGA that can be used to target a low perihelion. However, the required launch C_3 for a direct-to-Jupiter trajectory is on the order of 100 km^2/s^2; this could possibly be reduced to as low as 8 km^2/s^2 by using multiple inner-planet gravity assists at the cost of extended mission duration. Assuming a direct-to-JGA launch, the minimum solar approach is achieved in 3.7 years. The all-carbon sail will be

[§]Data available online at www.webelements.com [retrieved 8 March 2010].

stowed until the minimum solar approach to minimize sail performance degradation.

An analysis of the cruise speed attained as a function of the solar sail lightness number was performed. A lightness number of 1 corresponds to a characteristic acceleration of 5.93 mm/s^2. If a lightness number of 1 is used and the sail is pitched face-on to the sun, then solar gravity is effectively switched off and the sail follows a rectilinear path, with the perihelion velocity conserved. For the Jupiter–solar perihelion transfer, an ellipse with an aphelion at 5.2 AU (Jupiter) and a perihelion at 0.0186 AU (four solar radii) was considered. For a lightness number of 1, the rectilinear postperihelion trajectory will be at 90 deg to the incoming ellipse. The perihelion velocity can be determined as 65.04 AU/year, with the spacecraft reaching 200 AU in 3.08 years postperihelion, 1000 AU in 15.4 years, and the Oort cloud in 768 years. This mission duration is clearly unacceptable and so the characteristic acceleration needs to be increased to far in excess of 5.93 mm/s^2. For lightness numbers greater than 1, the trajectory is an inverted hyperbola (net force of the sail is outward). Considering the conservation of specific energy, from perihelion to the cruise speed v_∞, the cruise speed can be obtained by as

$$v_\infty = 2\pi\sqrt{\frac{2r_a}{r_p^2 + r_p r_a} - \frac{2(1-\beta)}{r_p}}\ \text{AU/year} \tag{5}$$

From Eq. (5) it is determined that a 100-year cruise to the Oort cloud is enabled with a characteristic acceleration of 343 mm/s^2, whereas a faster 30-year cruise would be possible using a characteristic acceleration of 3874 mm/s^2.

Selecting the fast mission as the baseline and with the required sail performance defined, the sail loading (total spacecraft mass per unit area) required for a sail characteristic acceleration of 3874 mm/s^2 is 0.59×10^{-3} g/m^2. Designing for a launch C_3 of 100 km^2/s^2, the Ariane 5 ECB with multiple ignition (Ariane V Versatile) can deliver approximately 750 kg to this launch energy; the total mission launch mass is thus defined as 675 kg, allowing a 10% launch-mass margin. The launch mass will consist of flight and nonflight mass (deployer mass), therefore assuming that the nonflight mass is approximately 200% of the flight mass, and then the flight mass is 250 kg, which requires a vast disc sail radius of 11,629 m.

The available nonsail mass fraction was assumed to be 10% of the flight mass. Therefore, the nonsail mass will have an allocation on the order of 25 kg, which will be evenly distributed over the entire sail surface and not as a centrally located bus, necessitating a bus areal density of less than 59 $\mu g/m^2$. From the allocated 25 kg, all the normal subsystems such as telecommunications, command and data handling, and power must be provided. Additionally, a remote sensing capability such as a radar or lidar system is required, with molecular electronics and micromachining providing early indications of how such systems could be developed. The bus systems should be incorporated into the sail film, with small micronodes placed at the intersections of the sail film grid network. The nodes will communicate with each other using the quantum conductance of the carbon nanotube sail, which will also generate the onboard power by using the inductive effect of the motion of the conducting sail through the interstellar magnetic field with power storage achieved by tuning some of the carbon nanotubes to act as semiconductor storage devices. The sail will be able to generate almost 10^{-7} W/m^2 by such means, assuming an interstellar magnetic field strength on the order of 10^{-10} T.

Communications at path lengths beyond 1000 AU necessitate the use of non-RF systems due to the significant space losses incurred; as such, an optical system is required. Noting that the divergence of a Gaussian beam (laser) is larger for longer wavelengths, a laser wavelength of 890 nm (near infrared) was assumed. Considering a distributed antenna diameter matching that of the sail, a phased-array laser transmitter working as a coherent light source was assumed. The optical system will communicate with a near-Earth relay spacecraft, with terminal antenna of 6.5 m, which is the preferred option, as it eliminates atmospheric losses in the signal. Assuming that approximately 10% of the available power is used for telecommunications purposes, an upper limit of 10^{-8} W/m^2 of power is available. It was thus found that the vast aperture provides a data rate on the order of 4 million bps at 200 000 AU.

The Oort cloud mission timelines for 30-year (fast) and 100-year (slow) cruises are detailed in Table 7.

IX. Discussion of Solar Sail Technology Development

Technology can be developed following two primary models: either technology push or applications pull. For space systems engineering, a technology is typically developed and adopted once a clear enabling capability has been identified to justify the development expenditure. The endpoint of solar sailing applications pull was investigated through consideration of exploration beyond the Kuiper belt. To realize the endpoint of solar sailing applications pull, the identified technology requirements of that endpoint application must be enabled through prior applications, with each prior mission acting as a stepping stone to the next. It is of note that the first solar sail mission application will likely be required to overreach the required technology for that application in isolation; however, the mission would need to be considered with a larger roadmap context [31,34].

Using solar radiation pressure alone to propel the spacecraft, no impossibility was identified when considering an Oort cloud mission. However, the practical difficulties of the mission, such as control of a nearly 12 km structure, means that alternative approaches are likely to be sought. An Oort cloud mission is thus classed as beyond far term for solar sail propulsion and is hence unsuitable as a solar sailing applications pull. With this, the IHP mission is considered to be the endpoint of solar sailing applications pull. Thus, the enabling solar sail technology requirements of the IHP mission application must be devolved to preceding missions such that the IHP mission is enabled through a coherent development roadmap. Future technology development of solar sailing is required to enable these technology requirements if a solar sail IHP mission application is to be enabled. As such, the current solar sail technology development focus on three-axis-stabilized square solar sails is not fully compatible with any future solar sail IHP mission application; however, a Kuiper belt mission does appear feasible. The development of solar sailing technology thus appears to be at an important juncture, where either current technology activities are revisited to enable an application such as the IHP mission, perhaps through a hybrid sail architecture (say, a spinning square sail), or it is accepted that solar sailing will likely never be used for such a mission. However, accepting the latter critically limits the number of applications-pulling solar sail technology development and hence

Table 7 Oort cloud mission timeline from sail deployment

Solar distance, AU	Milestone	Slow mission	Fast mission
30	Kuiper belt	3.2 weeks	6.6 days
50	——	1.2 months	11 days
100	——	2.4 months	3.1 weeks
200	Heliopause	4.8 months	1.4 month
1000	——	2.0 years	7.2 months
50,000	Inner edge of Oort cloud	100 years	30 years
200,000	Outer edge of Oort cloud	400 years	120 years

critically questions the sustainability of future solar sail technology development within the current three-axis-stabilized square sail architecture.

X. Conclusions

The use of solar sail propulsion has been considered for exploration of the Kuiper belt and beyond. A detailed analysis of a 25-year Interstellar Heliopause Probe mission was conducted using a method of integrated statistical regression analysis to identify critical enabling technology requirements. It was found that the 25-year mission duration necessitated a solar sail characteristic acceleration of 1.5 $\mathrm{mm/s^2}$. A very large three-axis-stabilized square solar sail, or a spinning-disc sail with assembly loading of 1.8 $\mathrm{g/m^2}$, which is within the range of prior spinning-sail design concepts, is thus required. However, significant sail technology development in areas such as sail film areal density reduction is required. Furthermore, non-solar-sail critical spacecraft systems were identified in order to minimize the mission advancement degree of difficulty by reducing the technology requirements on the solar sail. It was found that a radioisotope thermoelectric generator power density of approximately 12 W/kg was required to enable the mission. However, it was also found that a power density above this was not required, as it provided only modest benefits for the sail sizing. Similarly, an advanced lightweight composite high-gain antenna and a high-efficiency Ka-band traveling-wave tube amplifier were found to be critical enabling technologies. Furthermore, as the sail structure is likely to be jettisoned at 5 AU for science reasons, a reliable concept for sail separation is needed, as is a similar concept to separate the sail from its deployment module. Finally, autonomous operation beyond Jupiter and during the science phase is an important requirement to reduce ground segment costs for such a long-duration mission.

Following the Interstellar Heliopause Probe mission, a similar Kuiper belt mission study was briefly considered that would act as a precursor. It was found that the Kuiper belt mission could be conducted with a sail assembly loading of 3.3 $\mathrm{g/m^2}$ and a sail radius of 63 m. A low-cost Rockot launch vehicle was found to be suitable for this mission. It was also noted that the Kuiper belt mission could be performed by a three-axis-stabilized square sail of side length 130 m and assembly loading of 6.5 $\mathrm{g/m^2}$; however, such a mission could not easily act as a technology pathfinder toward a Heliopause mission, due to the required sail performance of the later mission.

Finally, an Oort cloud mission study was presented in which it was noted that beyond the Heliopause, the Oort cloud is the next scientifically interesting destination available by solar sail propulsion. The Oort cloud mission was found to require a significant increment in technology capabilities beyond the Interstellar Heliopause Probe mission, with a very close solar pass, an all-carbon-mesh sail, and a distributed spacecraft system across the sail surface generating power from the interstellar magnetic field. In conclusion, a solar sail Oort cloud mission is considered to be beyond far-term technology. As such, the Interstellar Heliopause Probe mission is considered to be the outermost mission scenario from our solar system that will likely use solar sail technology and the likely goal point of any solar sail technology development program. Therefore, contrary to previous studies, the conclusion that a spinning sail is required is critical for any reasonable Interstellar Heliopause Probe mission, as this technology requirement must be enabled through prior missions, with each mission acting as an enabling facilitator toward the next mission. Note that it is highly unlikely that, say, a three-axis-stabilized square sail mission to the Kuiper belt will provide much, if any, confidence to then progress to a spin-stabilized disc solar sail architecture for an Interstellar Heliopause Probe mission. As such, for a far-term solar sail mission such as an Interstellar Heliopause Probe to be enabled, all preceding missions must act as enabling facilitators to the next mission and must therefore develop the sail architecture required for such far-term missions. With this conclusion, the ongoing and future technology development of square solar sails should be carefully considered.

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