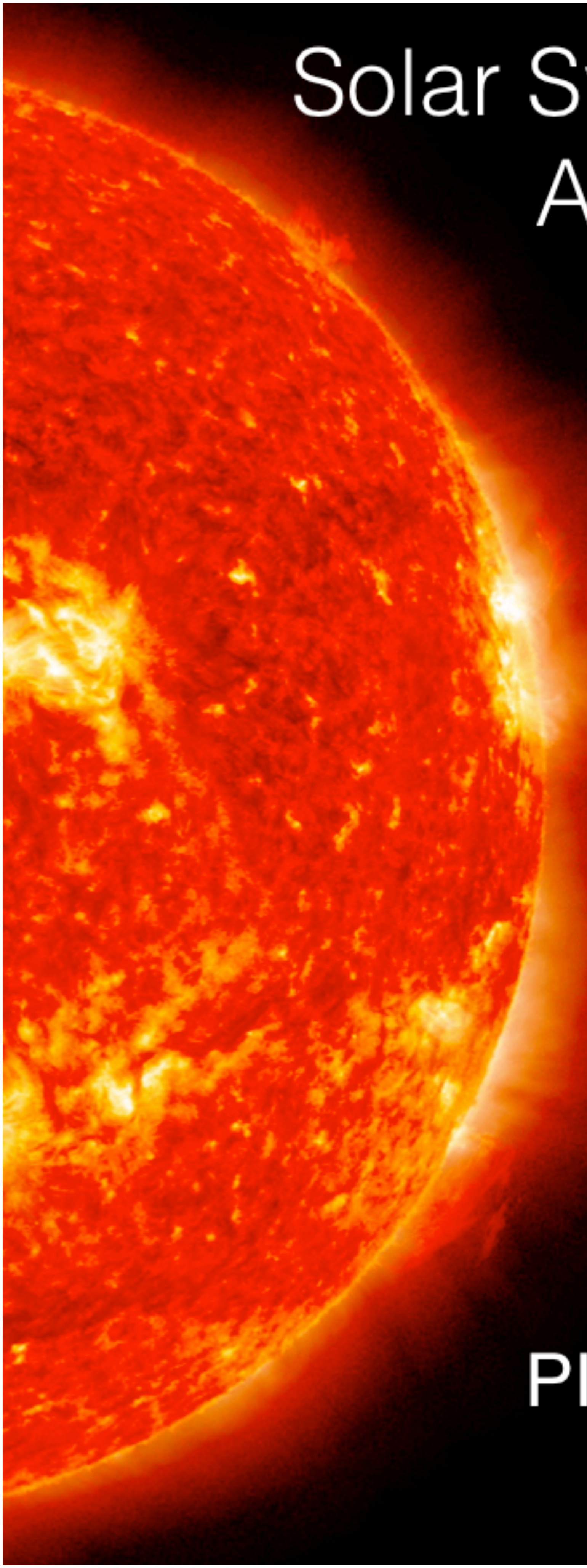

# Solar System Escape Architecture for Revolutionary Science

*Phase 1 Final Report*


Jet Propulsion Laboratory

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**Acknowledgement**

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*The cost information contained in this document is of a budgetary and planning nature and is intended for informational purposes only. It does not constitute a commitment on the part of JPL and Caltech.*

Nathan Gerald Nosanov

Owen Patrick Trease

The work described in this report is dedicated to Nathan Gerald Nosanov, born May 4, 2012, and Owen Patrick Trease, born April 19, 2013. May they live to see such things take flight.

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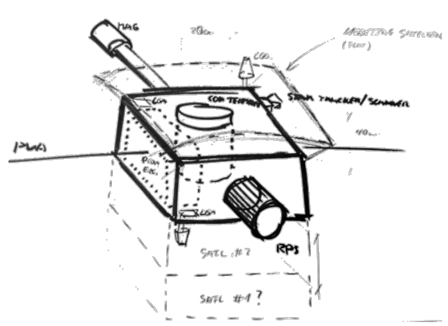
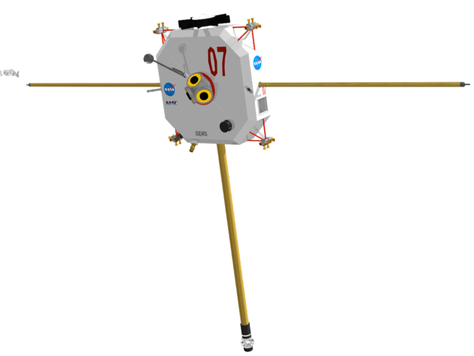
## SYNOPSIS

Our most distance spacecraft, Voyager 1 and Voyager 2, recently reached the boundary of the solar system known as the Heliopause. Beyond this boundary lies interstellar space, and, at tremendous distance, the stars. The Voyager spacecraft have shown us that the Heliopause is a dynamic, rapidly changing environment, varying greatly as the fields and effects emanating from the sun are met by a host of forces pushing in from our nearby galactic neighborhood. This is the region we aim to explore.

Specifically we aim to develop a mission architecture and spacecraft concept capable of reaching the Heliopause (~100 AU) region in multiple directions of interest, with a variety of scientific instrument suites, within a reasonable timeframe (~15 years.) This report details our Phase 1 work including science goal definition, trajectory planning, technology research, mission planning, instrument selection, spacecraft design, and more.

**We present an architecture employing a 250m x 250m solar sail with a 175 kg spacecraft. This spacecraft could carry a variety of instrument suites depending on the destination, time, and other factors. The spin-stabilized spacecraft would powered by a small radioisotope power system (~20 watts), makes use of an optical communication system, and carries 3 extensible booms for instrument accommodation as visible in the sketch and rendering below. We conclude that it would be reasonable to implement the architecture described herein in the 2020-2030 timeframe based on likely near-term technological and material advances. We are confident that cruise times of ~15 years could be achieved to science-rich destinations.**





This report details the approach we used to come to these conclusions and identifies some of the steps along the way. We utilized a variety of methodologies involving our full capability at JPL – from small point studies to large collaborative engineering processes. This paper also describes potential industry partnerships and costing information, and includes the methods used to account for our project's significant deviation from any prior mission in the NASA costing tool database.

Outreach and public engagement is an important part of NASA activities. We engaged in several different forms of this outreach and describe them in this report. We conclude with some open questions and a summary of activities we hope to engage in with Phase 2 funding.

The entire team would like to thank the NIAC program for the opportunity to investigate this fascinating and challenging concept.

## ABSTRACT

The Voyager program gives us tantalizing clues as to the nature and behavior of the Heliopause – the boundary between the sun's influence and the interstellar medium. This information comes from forty-year old instruments designed to study the outer planets. A targeted Heliopause investigation would give insight into the formation of the solar system, the role of the sun in the local interstellar neighborhood, and contribute to human exploration planning by helping to predict periods of low galactic cosmic ray (GCR) penetration into the inner solar system.

## MOTIVATION / VOYAGER PROGRAM INTRODUCTION

The Voyager program arose out of a realization in the early 1960s that the mid-1970s would offer a once-in-170-year planetary alignment. This event would allow one spacecraft to fly by multiple outer planets. The primary Voyager Mission was to investigate the Outer Planets, and the Voyager Interstellar Mission (VIM) was confirmed in the late 1980s as it was clear that the spacecraft were functioning well and actually had a chance of reaching the Heliopause boundary intact. As of mid-2013 the VIM is still active and the two Voyager spacecraft continue to return data, and are expected to continue to do so until the mid 2020s when the Plutonium$^{238}$ -based power system no longer produces enough energy to power the communication systems. From that point the Voyager spacecraft will be silent monuments to humanity's technical achievements for roughly one million years until high-speed impacts with interstellar dust grains cumulatively abrade the spacecraft back to the stardust from which they were made.

The SSEARS project concept arose from a conversation with Dr. Ed Stone, former JPL director and Voyager project scientist since the inception of the project. At one point Dr. Stone was asked how he would continue the Voyager science. His answer was to send multiple spacecraft in multiple directions to study the 3d structure of the boundary. The aim of this project is to develop an architecture that enables this investigation.

## HELIOPAUSE SCIENCE

This section will discuss the science goals and rationales for the goal of returning to the Heliopause and traveling beyond. These goals were derived from conversations with Dr. Stone (Voyager Project Scientist), the Science Mission Directorate, the Human Exploration and Operations Mission Directorate, the Heliophysics Decadal Survey, and the Planetary Science Decadal Survey.

SSEARS would enable the first comprehensive measurements of plasma, neutrals, dust, magnetic fields, energetic particles, cosmic rays, and infrared emission from the outer solar system, though the boundaries of the Heliosphere, and on into the interstellar medium (ISM). This would allow the mission to address key questions about the distribution of matter in the outer solar system, the processes by which the Sun interacts with the galaxy, and the nature and properties of the nearby galactic medium.

The principal scientific objectives of such a mission would be to:

- Explore the nature of the interstellar medium and its implications for the origin and evolution of matter in our Galaxy and the Universe;

- Explore the influence of the interstellar medium on the solar system, its dynamics, and its evolution;
- Explore the impact of the solar system on the interstellar medium as an example of the interaction of a stellar system with its environment;
- Explore the outer solar system in search of clues to its origin, and to the nature of other planetary systems.
- Significantly reduce human radiation risk for future crewed missions by understanding mechanisms leading to variability of dangerous levels of radiation in the inner solar system

We describe below examples of the scientific issues that could be addressed.

**The Nearby Interstellar Medium**

Our Sun is thought to be located near the edge of a low-density interstellar cloud (~0.3 $cm^3$), often referred to as the local interstellar cloud (LIC), that is made up of material blowing from the direction of star-forming regions in Scorpius and Centaurus. Present knowledge of the ISM is based on astronomical observations that average over long lines of sight, measurements of sunlight resonantly scattered back by interstellar H and He, data returned from the Voyager and iBEX spacecraft, and in situ measurements of neutral gas and dust that penetrate the Heliosphere. Direct observations of our local cloud by SSEARS would provide a unique opportunity to derive the physical properties of a sample of interstellar material, free from uncertainties that plague the interpretation of data acquired over astronomical lines-of-sight, and from uncertainties arising from the exclusion of plasma, small dust particles and low energy cosmic rays from the Heliosphere. Direct measurements would be made of the elemental and isotopic composition of the ionized and neutral components of the interstellar gas and of low-energy particle components, and of the composition and size distribution of interstellar dust. These measurements would provide a benchmark for comparison with solar system abundances (representative of the pre-solar nebula) and with abundances from more distant galactic regions, thereby providing important constraints on theories of galactic chemical evolution.

SSEARS would also measure cosmic ray nuclei and electrons, free from the influence of the Heliosphere, and investigate astrophysical processes that include acceleration by supernova shock waves, interstellar radio and x-ray emission, recent nucleosynthesis, and the heating and dynamics of the interstellar medium. Little is known about the properties of magnetic field in the local cloud or in the region beyond the termination shock. SSEARS would enable the first in situ measurements of interstellar magnetic fields and of the density, temperature, and ionization state of the interstellar gas, including studies of their variations over a variety of spatial scales. The possibility of identifying organic matter in the outer solar system and ISM is also an exciting possibility that is under investigation.

**The Interaction between the Interstellar Medium and the Solar Wind**

The solar wind and the interstellar medium interact to create the global Heliosphere, shown schematically in Figure 1. The size of the Heliosphere is determined by the balance between the solar wind ram pressure and the interstellar pressure. There are presently no direct measurements of the size and structure of the Heliosphere and our present understanding is based on theory and modeling, constrained by a few key measurements. The Voyager spacecraft have detected radio emissions that are thought to be caused by interplanetary shock waves hitting the denser interstellar plasma. Voyager 1 should soon reach the termination shock, providing a first direct test of our current understanding of Heliospheric structure, although some of the Voyager instruments were not designed to explore the boundaries of the Heliosphere and interstellar medium. SSEARS' enhanced capabilities and lifetime would greatly extend Voyager's exploratory studies, answering questions relating to how the ISM influences the solar system and how the solar system influences the ISM.

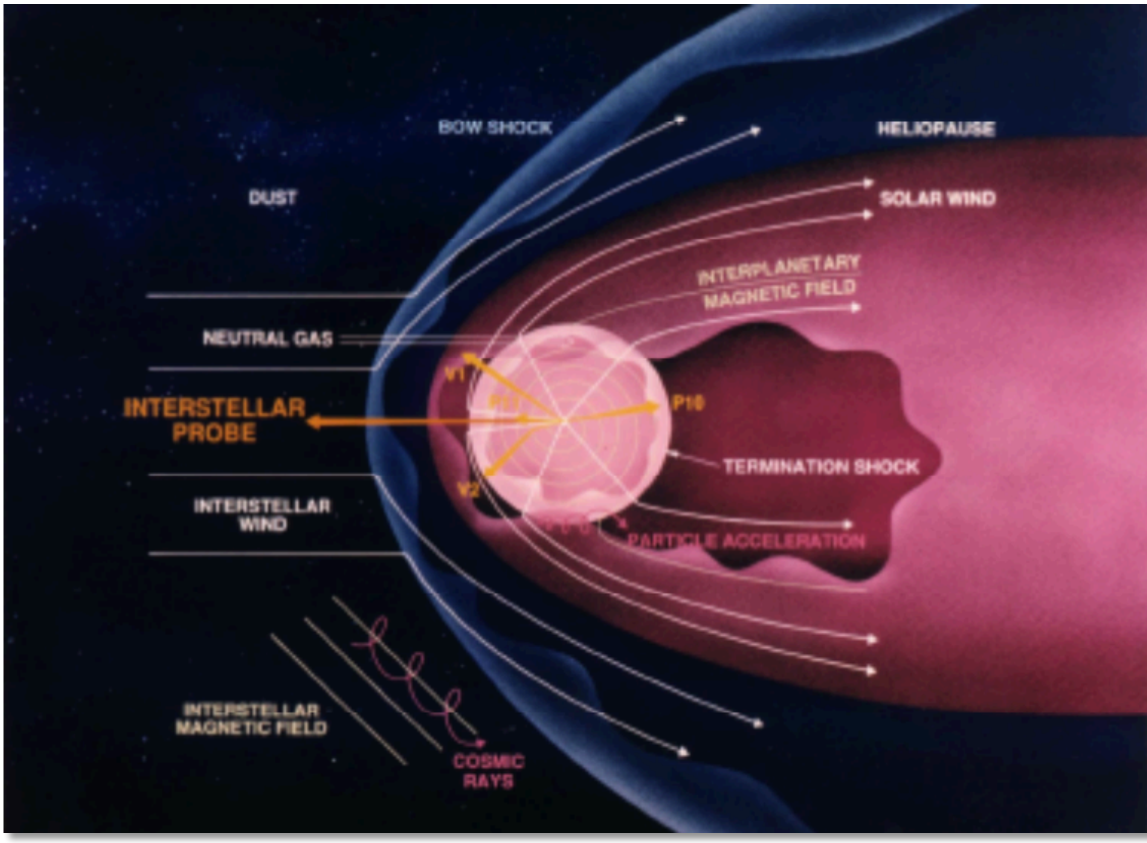



**Figure 1: Schematic of the Heliosphere created by the supersonic solar wind diverting the interstellar flow around the Sun. The interstellar ions and neutrals flow at 25 km/s relative to the Sun. The solar wind, flowing outward at 400-800 km/s, makes a transition to subsonic flow at the termination shock. Beyond this, the solar wind is turned toward the Heliotail, carrying with it the spiraling interplanetary magnetic field. The Heliopause separates solar material and magnetic fields from interstellar material and fields. There may or may not be a bow shock in the interstellar medium in front of the Heliosphere.**

The termination shock is a powerful accelerator that accelerates particles to energies as high as 1 GeV. In situ studies of shock structure, plasma heating, and acceleration processes at the termination shock will serve as a model for other astrophysical shocks. Past the termination shock, in the region called the Heliosheath, the solar wind flow is turned to match the flow of the diverted interstellar plasma, as illustrated Fig. 1. The spiraling solar magnetic field, frozen into the solar wind, is swept back with this flow. Depending on the unknown interstellar magnetic field strength, there may or may not be a bow shock created in the interstellar medium ahead of the nose of the Heliosphere. Energetic ions created by charge exchange in the Heliosheath can be used to provide an image of the 3D structure of the Heliosphere. Charge-exchange collisions lead to a weak coupling between the neutral and ionized hydrogen in the interstellar medium causing a pile-up of neutral hydrogen at the Heliosphere nose, referred to as the "hydrogen wall." The SSEARS spacecraft would pass through these boundary regions and make in situ measurements of the dust, plasma, fields and flows to answer questions regarding the size, structure and dynamics of the Heliosphere and the processes occurring at the boundaries. Our Heliosphere would serve as an example of how a star interacts with its environment. [i]

## SOLAR SYSTEM ORIENTATION

The plane of the Solar System is inclined at roughly 60 degrees to the plane of the galaxy, as seen in Figure 2. This helps us identify regions of interest because we can investigate the Heliopause along the solar system plane to avoid interactions with the galactic plane, and specifically study those interactions by sending the spacecraft along the galactic plane.

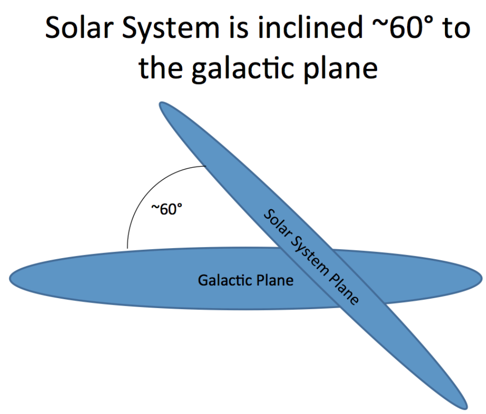



**Figure 2: The plane of the solar system and the plane of the galaxy are 60 degrees apart.**

## HELIOPAUSE/INTERSTELLAR MEDIUM REGIONS OF INTEREST

We began our thinking with the idea of sending spacecraft in the six cardinal directions from the Earth- up, down, left, right, in, and out relative to the sun and the galactic plane. The inclination of the solar system plane to the galactic plane increases the complexity of the ideal investigation by doubling the number of regions of interest. Figure 3 shows the directions of travel along the solar system ecliptic, and Figure 4 shows the directions of travel along the galactic ecliptic.

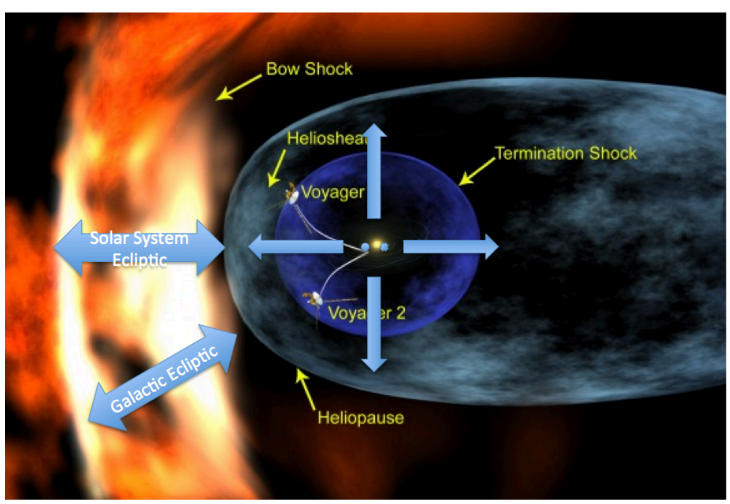



**Figure 3: Directions of travel/investigation along the Solar System Ecliptic**

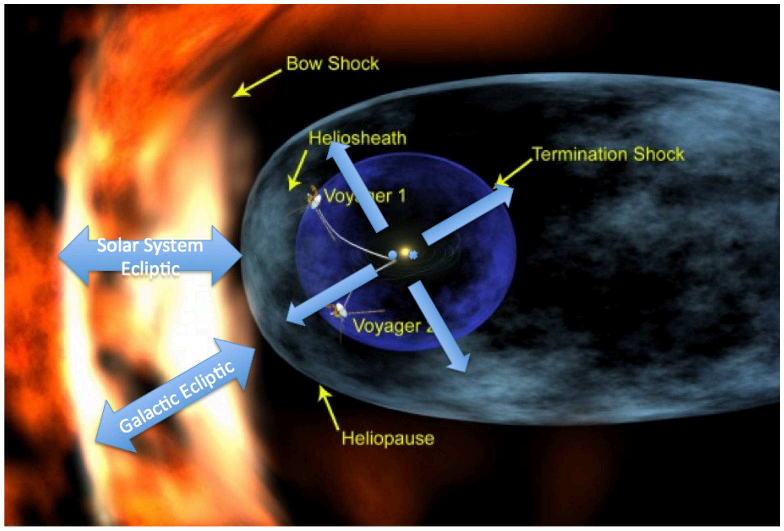



**Figure 4: Directions of travel/investigation along the Galactic Ecliptic**

Later in the project, after additional analysis, we revised our destinations to better account for the complexity of the sun's magnetic field. The sun's electric currents generate a complex magnetic field that extends far into space with the solar wind. The sun rotates as it emits the solar wind so the magnetic field is "wound" into a spiral known as the Parker Spiral.

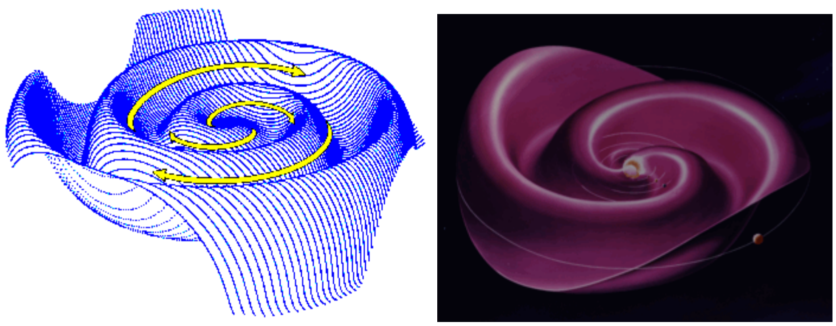


**Figure 5: Two visualizations[ii] of the sun's magnetic field. The spiral structure of the sun's magnetic field as the solar wind carries it past the planets.**

This magnetic field is primarily directed outward from the sun in one hemisphere, and inward towards the sun in another. The line between the two field directions is not exactly aligned on the solar equator, so the field becomes "wavy" as seen in Figure 5. As a result of this effect, radiation entering the solar system from the interstellar medium changes significantly with the solar cycle as the sign of the solar magnetic field switches. So, the north and south "hemispheres" of the Heliosphere should be investigated at different times corresponding to the amount of interstellar radiation present. This would necessitate at least two "waves" of spacecraft, one wave planned to reach the Heliopause in one hemisphere at the solar maximum, and another wave planned to reach the Heliopause in the other hemisphere at the solar minimum.

The SSEARS mission proposes to send multiple spacecraft out to the boundary of the Heliosphere to follow up on the Voyagers' recent discoveries and better define the interactions of the Heliosphere and the galactic media. The Voyagers were not targeted for specific locations to measure these interactions and so represent randomly placed snapshots of the Heliosphere—SSEARS would carry out a targeted analysis of the structure of the Heliosphere.

## SPECIFIC HELIOPAUSE DESTINATIONS

Specific targets for the SSEARS would be: the Heliosphere's nose (i.e., the point where the interaction is a head on collision due to the combined motion of the solar system and the galactic media), the anti-nose direction, and points perpendicular to the line connecting the nose and anti-nose (both "east and west" and "north and south" to determine the shape of the Heliosphere. Other potential targets are the anti-galactic center so that the center of the galaxy could be imaged using gravitational lensing or the source of neutral particles detected by iBEX (see Figure 6 below). We would also send spacecraft out at various times to measure these interactions during solar minimum and maximum for 2 solar cycles or a complete 22 year Hale solar magnetic cycle[iii,iv].

An unusual band of high energy neutral particles was detected by the iBEX and Cassini missions. A graphic for the iBEX data is shown in Figure 6—the band is apparently associated with a magnetic merging region where charge exchange between energetic charged particles from the interstellar medium are interacting with slow solar wind neutral particles. The nose is where the large arrow meets the Heliosphere.

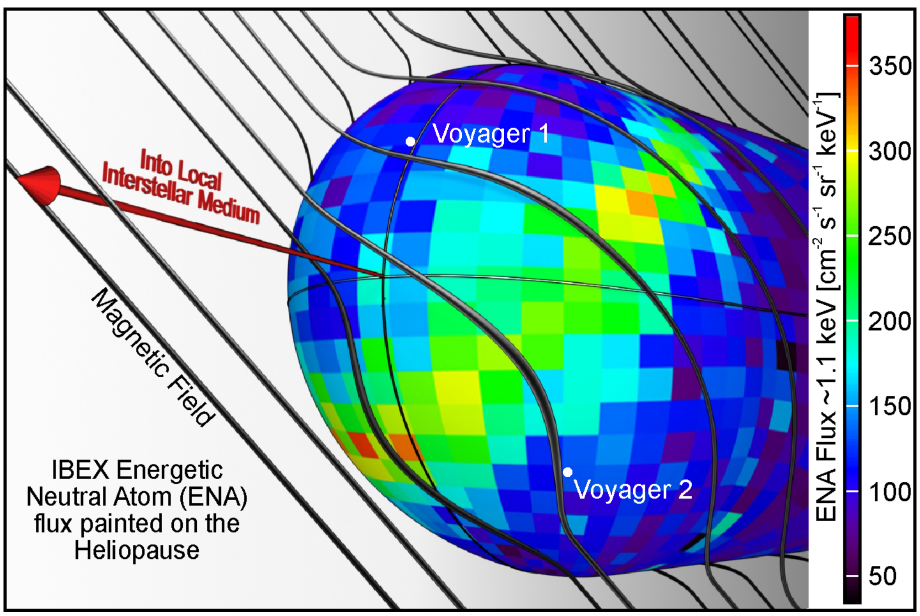
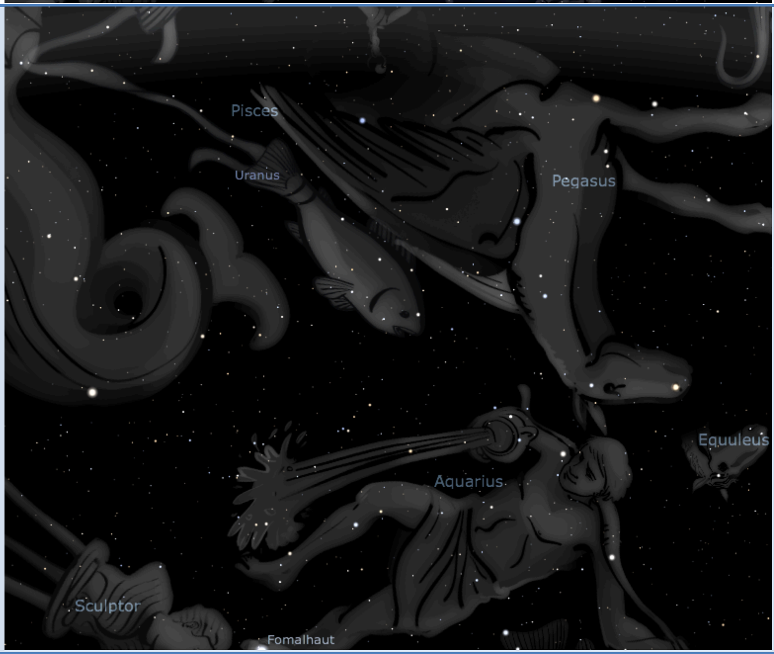
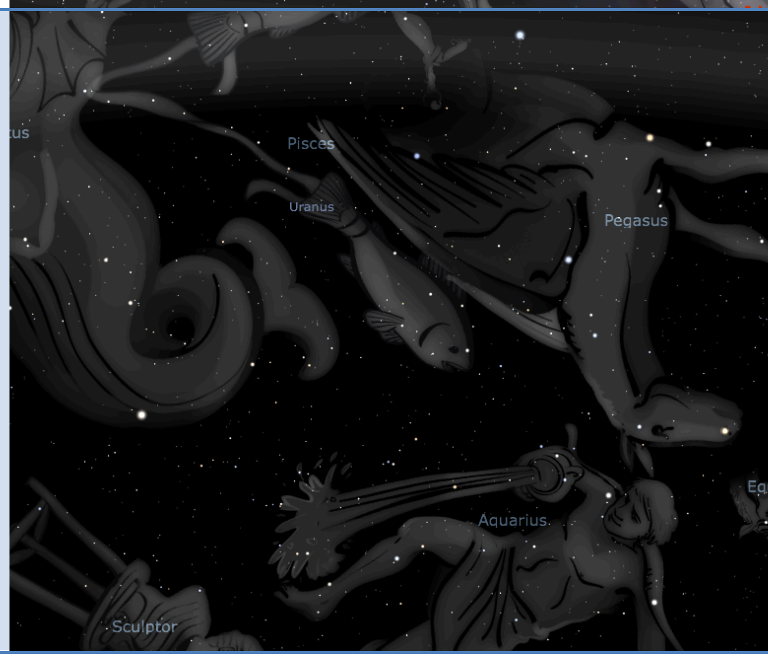



**Figure 6: Graphic presentation of the location of the Heliosphere's Nose and the iBEX source of high energy neutral particles. (IBEX Mission, NASA)**

Specific destinations along the Heliopause are displayed in two ways. First, a table is presented of approximate coordinates and the associated views from the surface of the earth. Constellations as visible from the surface of the Earth are rendered to aid recognition. The precise location along the Heliopause is at the center of each image. The table lists the destinations in Waves 1 and 2, temporally separated to allow investigation of the Heliopause at different times in the solar cycle. Second, a series of all sky-plots are included in a variety of coordinate systems.

Destination Region Table

| Name | Ra | Dec | Nearby Sky |
|---|---|---|---|
| Wave 1 | | | Reach destination regions at solar minimum |
| Apex | 30 | 270 |   |
| Anti apex | -30 | 90 |   |

| | | | |
|---|---|---|---|
| Nose | 5 | 260 |  |
| Anti nose | -5 | 80 |  |

<table>
<tr><td>90° from apex</td><td>-40</td><td>240</td><td>


</td></tr>
<tr><td>90° from nose</td><td>90</td><td>270</td><td>


</td></tr>
<tr><td>Wave 2</td><td></td><td></td><td>(Same destinations as Wave 1, but 11 years later) to study at different times in the solar cycle</td></tr>
</table>

## COORDINATE SYSTEMS

All coordinates presented here are referred to the J2000 epoch. The starting coordinate system is the standard Right Ascension (RA) and Declination (Dec) Earth-based Equatorial system used in most star maps. Right Ascension is a coordinate based on a sidereal day, the time it takes the Earth to make a 360 degree rotation inertially. It is classically broken up into 0 - 24 hours as in a solar day but the sidereal day is 1/365.25 (approximately 4 minutes) shorter than a solar day. The scale is 24 hours to 0 hours as you move from left to right on the map, as time marches on, stars that reach your north-south meridian are assigned a RA corresponding to the sidereal time. The stars are fixed in this coordinate system but the sun moves approximately 1 degree (2 apparent solar diameters) eastward each day in this coordinate system (right to left). These coordinates are dependent on an epoch such as J2000 because they are tied to the Earth and change as the Earth's axis precesses. Dec goes from 0 on the projection of the Earth equator to +/-90° at the poles.

Figure 7 is a segment of a star map showing the RA scale on the bottom. The blue line across the middle is the terrestrial equator, Dec = 0, so the star Regulus has positive Dec while the star Alphard has negative Dec. The diagonal line that passes through Regulus is the ecliptic, the path the sun moves along during the year.

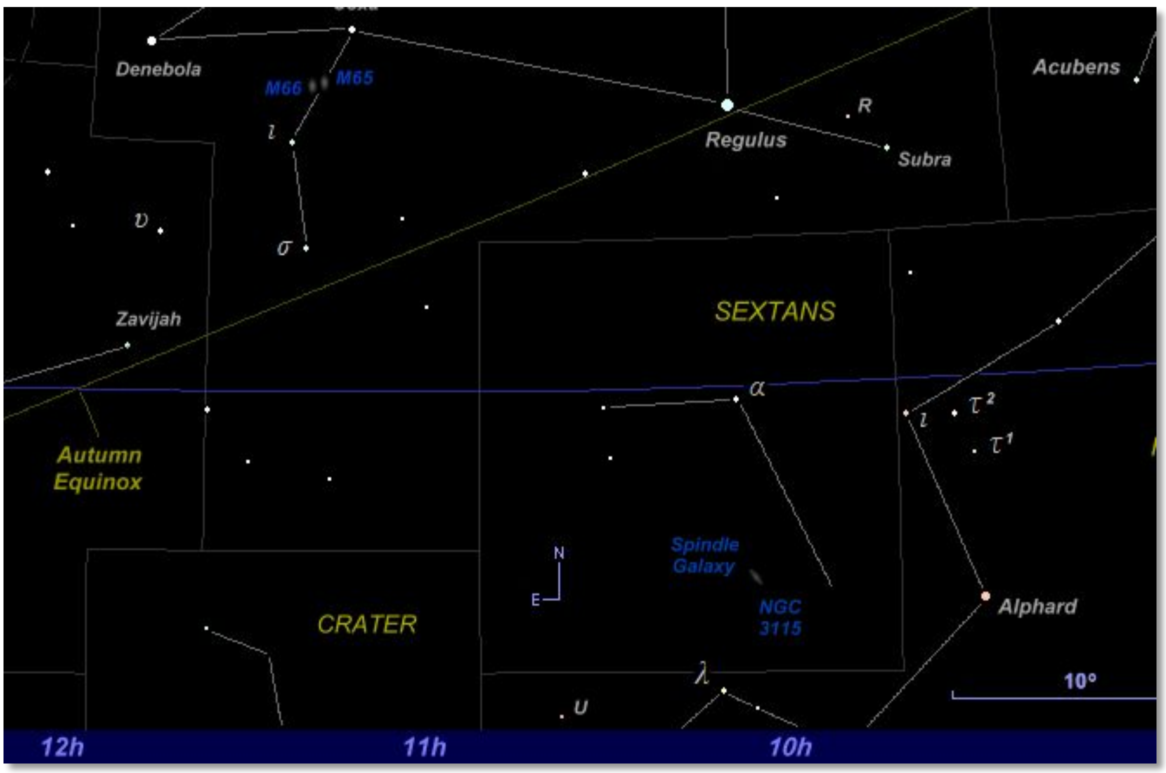



**Figure 7: Star Chart in Right Ascension and Declination Coordinates**

In our charts, the RA scale is changed to degrees (alpha) to correspond to other coordinate systems longitudes, i.e., RA corresponds to longitude and Dec corresponds to latitude (delta).

The maps are presented in a cylindrical projection, an x-y grid where longitude is x and latitude is y. In these projections, areas near the poles become distorted "beyond recognition". A projection that maps a sphere onto an elliptical area brings the polar areas more into what would be visualized in the real sky.

Figures 8-11 show the above maps with the points of interest for the project marked and a limited number of stars for orientation (dark blue). The Equator is in cyan. Also in cyan are the positions of spacecraft that will leave the solar system, Pioneer 10 and 11, Voyager 1 and 2, and New Horizons (P10, P11, V1, V2, and NH). The Galactic North pole (GNP), Galactic South pole (GSP) , Galactic center (GC), anti-Galactic center (AGC) and Galactic Equator are in red. The solar apex, the point in the sky that the sun is moving towards with respect to local stars, the anti-solar apex and apex equator are in black. The ecliptic is in grey. The Nose, Anti-Nose and Nose Equator are in green. The magenta dots are a rough reproduction of the iBEX ribbon[v]. Plots are made with Caltech's FORTRAN plotting package, pgplot[vi].

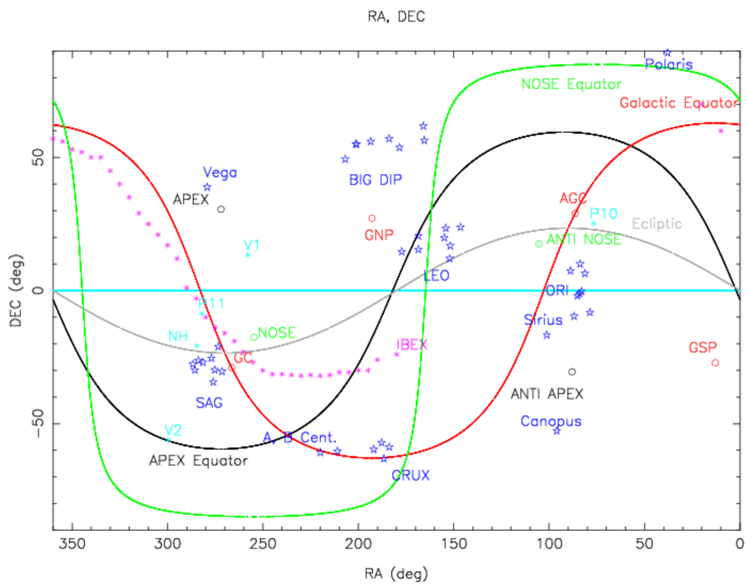



**Figure 8: Equatorial Projection of Sky**

The next coordinate system is Ecliptic coordinates, based on the ecliptic above. The coordinates are called Lambda and Beta corresponding to longitude and latitude. The 0 degree (beta) latitude is the ecliptic so the sun stays at 0 degrees latitude as it moves across the chart. The ecliptic North pole is at right ascension 18h 0m 0.0s (exact), and declination +66° 33′ 38.55″ Figure 9 shows the Ecliptic projection.

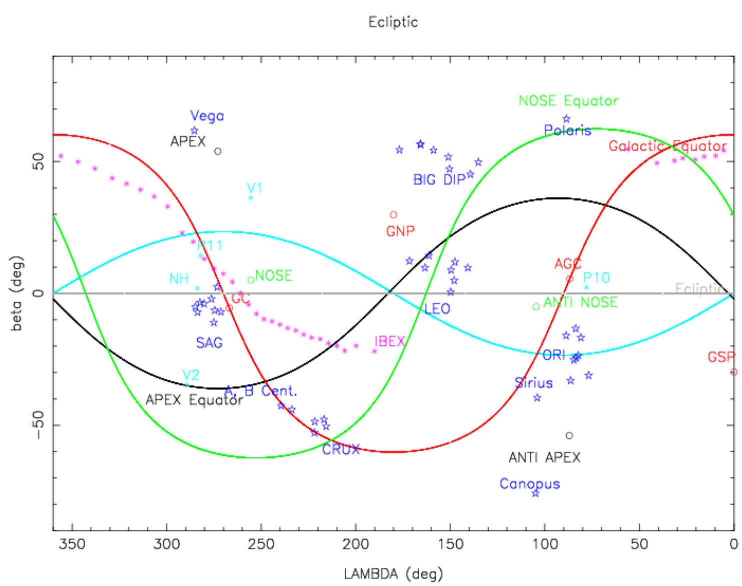



**Figure 9: Ecliptic Projection of Sky**

The next system is the Galactic coordinate system that is based on the Milky Way (our galaxy's center and poles). These coordinates are called l and b for longitude and latitude. The b=0 line is along the centerline of the Milky Way. The Galactic pole is at RA = 12h 51m.4, Dec = +27°.13 and the galactic center is at RA = 17h 45m.6, Dec = -28°.94. The galactic projection is shown in figure 10

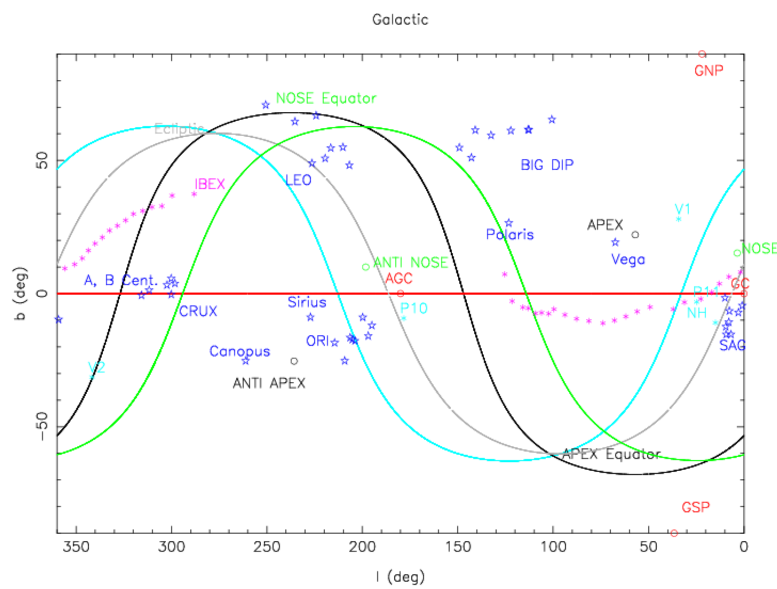



**Figure 10: Galactic Projection of Sky**

A particular mapping that preserves areas (equal area projection) is a Hammer map. It does not result in excessive crowding around the poles as severely as some other maps but does have a disadvantage that lines of constant latitude are not straight. There is another map (Mollwiede) that conserves areas and produces parallel lines of latitude but it is not solvable in closed form (requires iteration) and produces excessive crowding at the poles. The transformation of any of the above maps into a Hammer map is:

$$x = \frac{2\sqrt{2}\cos b\ \sin\frac{a}{2}}{\sqrt{1+\cos b\ \cos\frac{a}{2}}}$$

and

$$y = \frac{\sqrt{2}\sin b}{\sqrt{1+\cos b\cos\frac{a}{2}}}$$

where l is the longitude coordinate, b is the latitude coordinate and a is the difference between the longitude of a point and the center longitude for the map (180 degrees used). The calculated x and y coordinates reside within an ellipse in the resulting map.

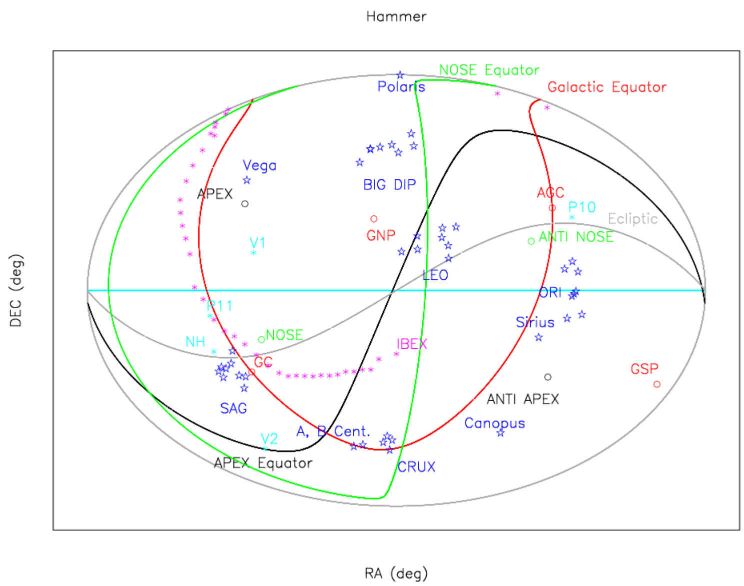



**Figure 11: Hammer Projection for Equatorial Sky**

### Conclusions

Programs to convert between different coordinate systems were developed for the SSEARS mission concept study. This was done to provide coordinates in a variety of systems to improve communication with scientists comfortable with each of the three systems. The table below list points of interest in the various coordinate systems.

**Points of Interest in 3 Coordinate Systems, J2000.**

| Point | Equatorial | | Ecliptic | | Galactic | |
|---|---|---|---|---|---|---|
| | RA | Dec | lambda | beta | l | b |
| Galactic North Pole | 192.85951 | 27.1283 | 180.023 | 29.8115 | 122.932 | 90 |
| Calactic Center | 266.4051 | -28.9362 | 266.84 | -5.53631 | 0.00011 | -0.0001 |
| Galactic South Pole | 12.859508 | -27.1283 | 0.0232 | -29.8115 | 302.798 | -90 |
| Anti Calactic Center | 86.4051 | 28.9362 | 86.8396 | 5.53631 | 180 | 0.0001 |
| Apex (visible) | 272 | 30.5 | 272.927 | 53.919 | 56.9845 | 22.0802 |
| Anti Apex | 92 | -30.5 | 92.9268 | -53.919 | 236.985 | -22.0802 |
| Nose | 254.73074 | -17.5687 | 255.4 | 5.1 | 3.53975 | 15.2112 |
| Anti Nose | 74.730744 | 17.5687 | 75.4 | -5.1 | 183.54 | -15.2112 |

## SSEARS SCIENCE CAPABILITIES

The SSEARS architecture is capable of enabling science investigation to the benefit of several major NASA stakeholders. For the Science Mission Directorate (SMD) SSEARS would enable the study of

1. the Interstellar Medium (IM) and its implications for the origin and evolution of matter

2. the influence of the IM on the Heliosphere
3. the influence of the Heliosphere on the IM
4. the dynamics of the coupling of the Heliosphere and the solar system as a model for other planetary systems

For the Human Exploration/Operations Mission Directorate (HEOMD), SSEARS could assist and possibly enable human interplanetary travel by enabling the *prediction of periods of low GCR penetration into the inner solar system*.

For the Heliophysics directorate, the SSEARS architecture would enable direct, in-situ investigation of two of the four goals of the 2012 Heliophysics Decadal Durvey:

1. Determine the interaction of the sun with the solar system and the interstellar medium
2. Discover and characterize fundamental processes that occur both within the Heliophere and throughout the universe.

## PHASE 1 GOALS

Our phase 1 NIAC proposal included the following goals for the phase 1 study period:

1. Identify science destination "directions/regions"
2. Develop trajectory modeling tool
3. Determine current material performance limitations
4. Project material trends to ~2030
5. Identify/design concept spacecraft for scaling estimates/modeling
6. Determine mission parameters for 2030 mission assuming current development trends
7. Propose for follow-up funds for further development

We have successfully completed our stated Phase 1 goals on schedule.

## PROPULSION METHODS

This section will describe the decisions made that led to the selection of the solar sail as a method of reaching the Heliopause.

Among our first thoughts when planning this project was a recognition of the need to reach the Heliopause much faster than the Voyager spacecraft did. Their 34 year journey was interpreted programmatically by NASA as the Voyager Primary mission (~1977- ~1989) and the Voyager Interstellar Mission (~1989-present). A single 35 year mission to return to the Heliopause strains the imagination as NASA has never in its history approved or directed a mission that, from its inception, was intended to spend over three decades en route. (Some science could be performed during cruise but the primary goals require proximity to the Heliopause.) Thus, we began our study with a review of available in-space propulsion methods.

### SOLAR ELECTRIC VS SOLAR SAIL

In-space propulsion has been extensively studied by various players throughout the space age. Specifically a study at JPL was done in the early 2000s and identified the best available in-space propulsion options at various distances and destinations from the sun. This study[vii] shows that solar sail propulsion is clearly superior to solar-electric propulsion for distances of many tens of AU from the sun or greater.

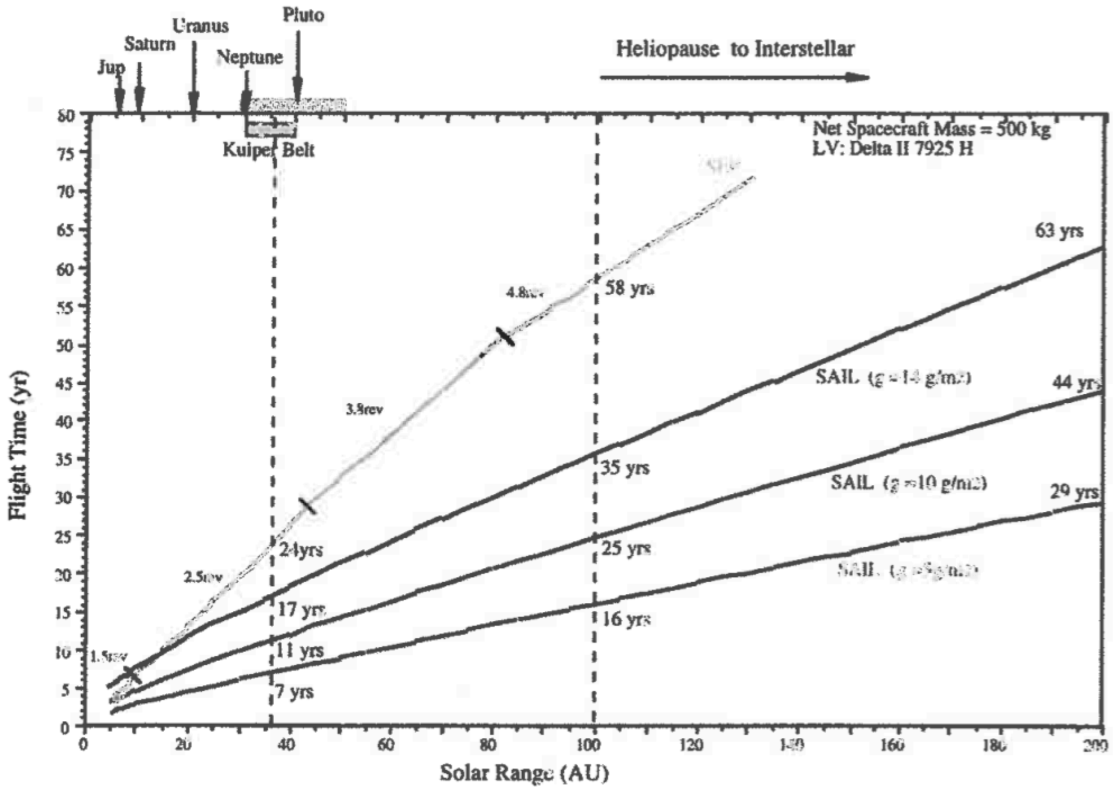



**Figure 12: Solar sails are significantly more effective than solar-electric propulsion when applied to solar system escape trajectories. This is largely due to the ability of the solar sail to come close to the sun without using fuel. Here three different sail densities are compared to SEP. (Image courtesy of Chen-Wan Yen)**

## NUCLEAR ELECTRIC PROPULSION

Nuclear Electric Propulsion (NEP) is not a practical solution to achieve solar system escape for most missions because it does not take advantage of the significant increase in power achieved by going close to the sun. As such, most NEP trajectories to the outer planets are relatively slow while requiring massive power systems[viii].

We concluded early on that a solar sail was the best option for this study at this time. This coincides well with current NASA investments in solar technology demonstrations in the 2014-2015 timeframe. We hope to convey that there are further benefits of that investment.

## SOLAR SAIL INTRODUCTION

A solar sail is a thin film sheet, usually with a very large area, that allows the use of light pressure to exert thrust on a spacecraft. Figure 13 shows the basic components of a solar sail vehicle.

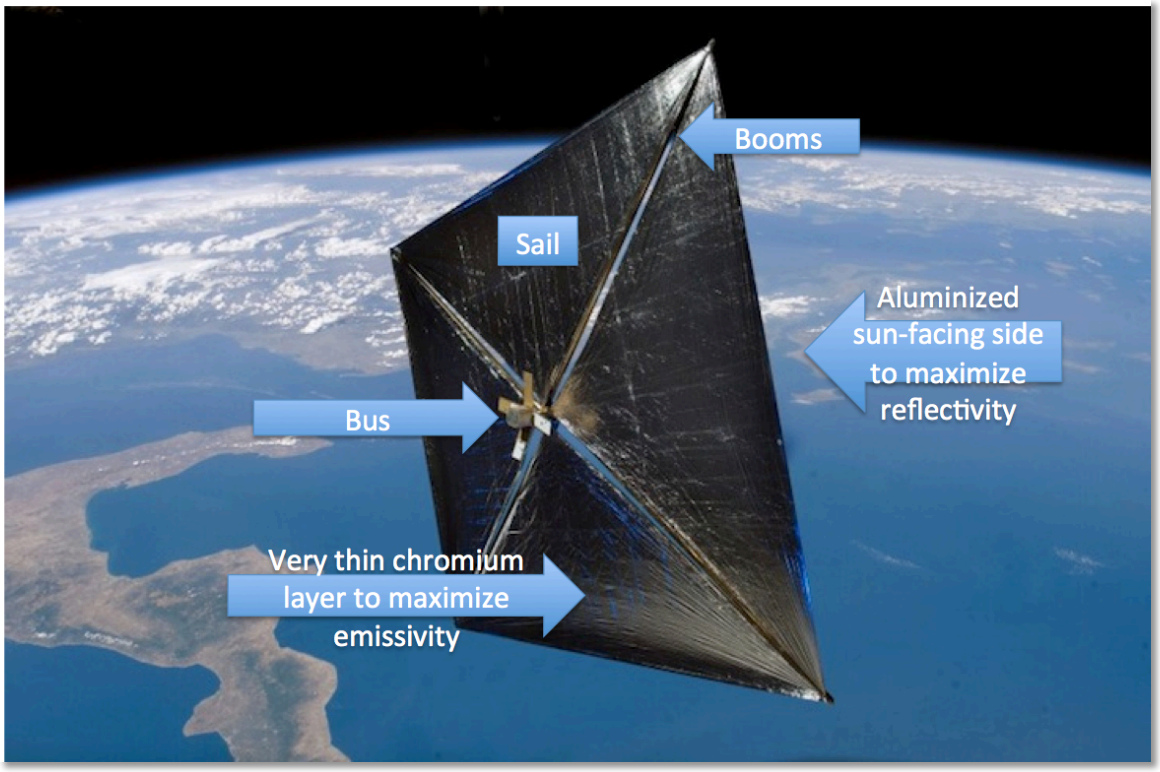



**Figure 13: Labeling the top-level relevant parts of a solar sail system.**

Key to the operation of a solar sail is to minimize areal density, minimize structural mass, and maximize area. The sun-facing side of the sail must be coated to maximize reflectivity and the opposite side must be coated to maximize emissivity. Usually the sail structural members "booms" extend radially outward from a central bus.

## TRAJECTORY MODELING

The key to using solar sails to escape the solar system is to reach the optimal distance from the sun. The amount of energy received from the sun decreases as $1/r^2$ and so thrust from the sail increases significantly as the sail approaches the sun. The trajectories involved are hyperbolic escape trajectories from the solar system. In order to achieve these trajectories there are several key design objectives:

1. Maximize sail area
2. Minimize spacecraft mass
3. Minimize perihelion (solar closest approach) distance
4. Maximize speed at perihelion

The following trajectories were developed using an instantaneously optimal steering law (for post solar-close approach). Assuming the Sun's gravity acts as a point-mass force on a sail, where the sail is modeled as a perfectly reflective flat plate, the equations of motion for the sailcraft are

$$\ddot{\bar{r}} = \frac{\mu}{r^2}(\beta \bar{a} - \hat{r}). \tag{1}$$

In Eq. (1), $\mu$ is the gravitational parameter of the Sun, $r$ is the distance from the Sun, $\beta$ (dimensionless) is the sail loading parameter, and $\hat{r}$ is a unit-vector directed from the Sun to the sailcraft. The sail acceleration is defined as

$$\bar{a} = (\hat{r} \cdot \hat{n})^2 \hat{n}, \tag{2}$$

where $\hat{n}$ is a unit-vector normal to the surface of the sail. Note that the sail loading parameter $\beta$, also called the sail 'lightness number', is the ratio of solar-radiation pressure (SRP) to solar gravity, i.e.,

$$\beta = \frac{2AW_E R_E^2}{c\mu M}. \tag{3}$$

In the above equation, $A$ is the total area of the sail, $c$ is the speed of light, $R_E$ is one astronomical unit (AU), and $W_E$ is the solar flux at one AU. The total spacecraft mass $M$ is the payload mass $m_p$ plus the mass of the sail, or $M = m_p + \rho A$, where $\rho$ is the density of the sail. It is sometimes convenient to describe the performance of the sail in terms of characteristic acceleration $\kappa$, or the acceleration provided by the sail at one AU. The characteristic acceleration is then written

$$\kappa = \beta \frac{\mu}{R_E^2}, \tag{4}$$

where $\kappa$ typically varies from 0.01 mm/s$^2$ (doable today) to 1.5 mm/s$^2$ (aggressive).

For fast departure from the solar system, the goal is to maximize the component of sail thrust along the spacecraft's Heliocentric velocity vector $\hat{v}$. This optimization problem is posed succinctly as

$$\begin{aligned} &\max J = \bar{a} \cdot \hat{v}, \\ &\text{subject to } \hat{n} \cdot \hat{r} > 0. \end{aligned} \tag{5}$$

The condition $\hat{n} \cdot \hat{r} > 0$ is present to ensure that the sail acceleration defined in Eq. (2) is always directed away from the Sun. To solve the optimization problem, we begin by defining a transverse vector $\hat{\theta}$ such that $\hat{\theta}$ is in the plane of motion and normal to $\hat{r}$ such that $\hat{\theta} \cdot \hat{v} > 0$. Then $\hat{n}$ can be resolved into radial and transverse components according to

$$\hat{n} = n_r \hat{r} + n_\theta \hat{\theta}. \tag{6}$$

Making use of $\hat{n} \cdot \hat{r} > 0$ in Eq. (5), we can write Eq. (6) only in terms of $n_\theta$

$$\hat{n} = +\sqrt{1 - n_\theta^2}\, \hat{r} + n_\theta \hat{\theta}. \tag{7}$$

Now substituting Eq. (7) into Eq. (2), we arrive at the following expression for thrust acceleration

$$\bar{a} = (1-n_\theta^2)\left[\sqrt{1-n_\theta^2}\,\hat{r} + n_\theta \hat{\theta}\right] \tag{8}$$

Note that with Eq. (8), the performance index $J$ in Eq. (5) is

$$J = \bar{a}\cdot\hat{v} = (1-n_\theta^2)\left[\sqrt{1-n_\theta^2}\,(\hat{v}\cdot\hat{r}) + n_\theta(\hat{v}\cdot\hat{\theta})\right] \tag{9}$$

Finally, from the expression provided in Eq. (9), it is easy to show that $J$ is a maximum when

$$n_\theta = \sqrt{\frac{3k^2 - k\sqrt{9k^2+8}+2}{6k^2+6}}, \quad \text{where } k = \frac{\hat{v}\cdot\hat{r}}{\hat{v}\cdot\hat{\theta}}. \tag{10}$$

This locally optimal steering law describes the best direction for orienting the sail normal for fastest departure from the solar system. Following the final perihelion passage, all trajectories are computed by simulating the spacecraft motion with Eq. (1) using the steering law provided in Eq. (10).

## BASIC TRAJECTORIES – UPPER STAGE FOR EARTH ESCAPE

We assumed an aggressive sail characteristic acceleration of 1.34 mm/s$^2$ (assumed because it matched with previous JPL work) and initially analyzed trajectories for four cases. In each figure the blue line represents the trajectory before solar close approach. The red vectors along the blue line show the sail normal angle to the direction of motion, and the green line shows the trajectory after solar close approach with the sail guided by the control law derived in the previous section.

1. using an upper stage rocket to escape earth orbit and escape along the ecliptic (Figure 14)
2. using an upper stage rocket to escape earth orbit and escape off the ecliptic using a Jupiter gravity assist to raise inclination (Figure 15)
3. using the sail to escape earth orbit and escape along the ecliptic (Figure 16)
4. using sail to escape earth orbit and escape off the ecliptic using a Jupiter gravity assist to raise inclination (Figure 17)

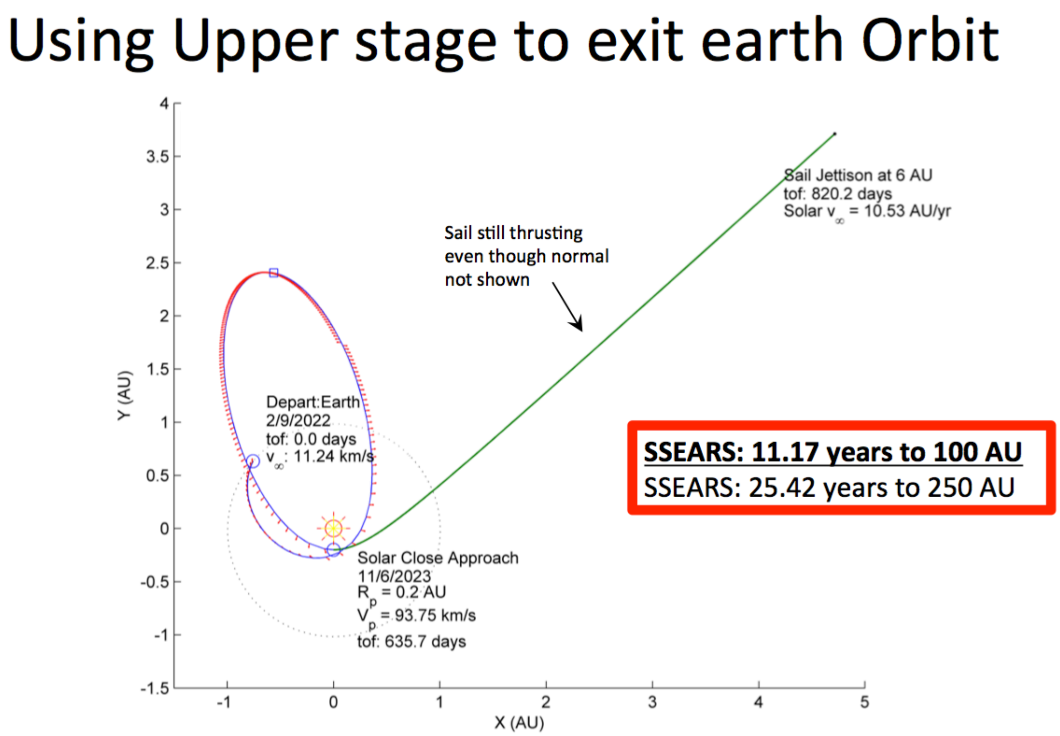



**Figure 14: This trajectory shows the spacecraft leaving earth, looping once around the sun to raise aphelion and lower perihelion, and then swinging by the sun again before proceeding radially away from the sun. This method reaches the vicinity of the Heliopause along the ecliptic in 11.1 years.**

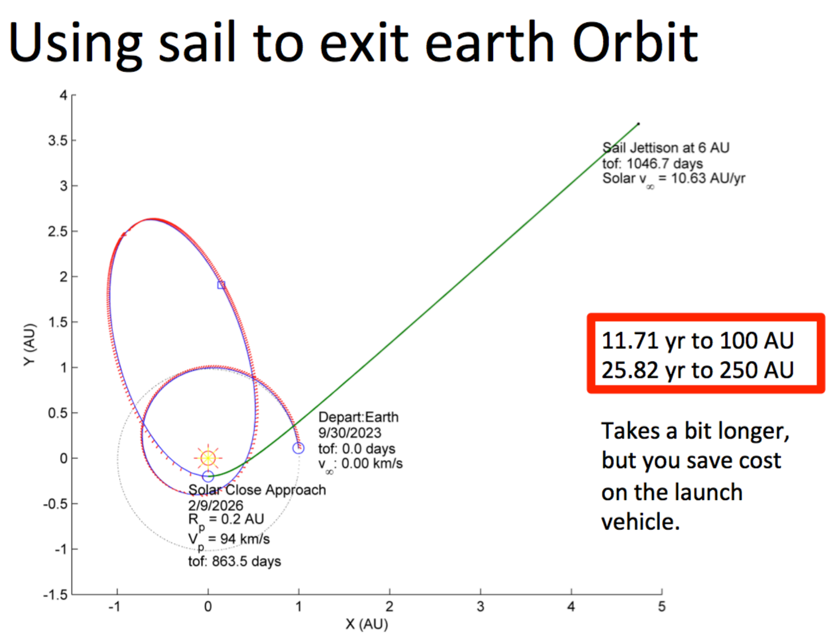



**Figure 15: This trajectory uses the sail itself to exit earth orbit and takes slightly longer - ~6 months – to reach the same distance along the ecliptic as that in Figure 14.**

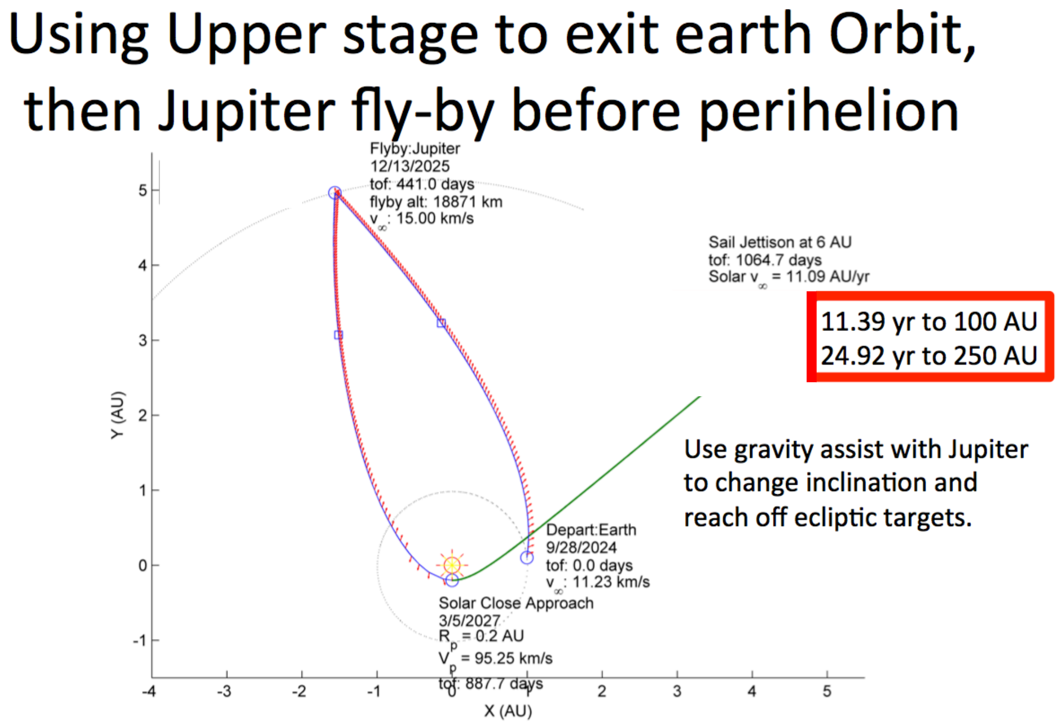



**Figure 16: The same distances can be reached at off-ecliptic inclinations by using a Jupiter gravity assist before perihelion. The transit time remains similar to the other trajectories. (3d nature of inclination change not displayed.)**

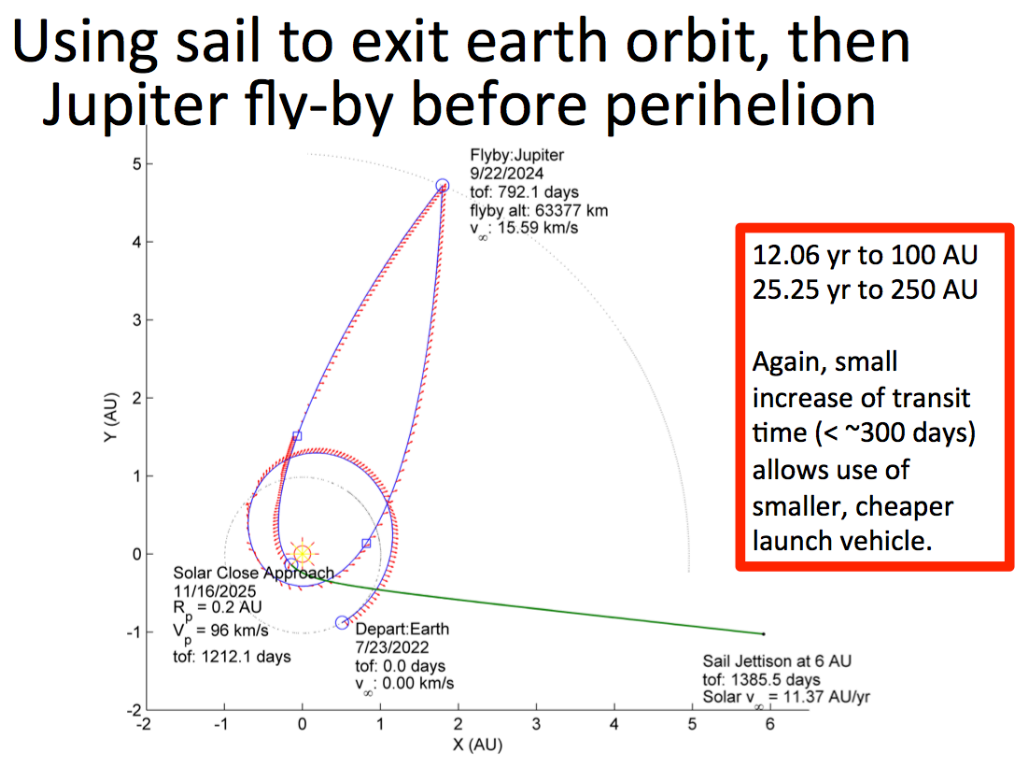



**Figure 17: The same technique as in Figure 15 applies to the off-ecliptic destinations, at a modest increase in transit time of ~ 1 year. (3d nature of inclination change is not displayed.)**

## SUNJAMMER

This section will discuss our heritage with the current NASA Technology Demonstration Mission "Sunjammer." Our proposed flight system is based on the Sunjammer spacecraft, due to launch in 2014. We studied the Sunjammer capabilities and decided to use it as a real-world foundation for the spacecraft design necessary to reach the Heliopause.

### PROGRAMMATIC BACKGROUND

Sunjammer is a technology demonstration project scheduled for launch in 2014. It is a sun observing mission with a 34 m x 34 m solar sail propulsion system. The sail is supported by 4 booms extending from the bus and ending in actuated vanes that enable steering of the sail.

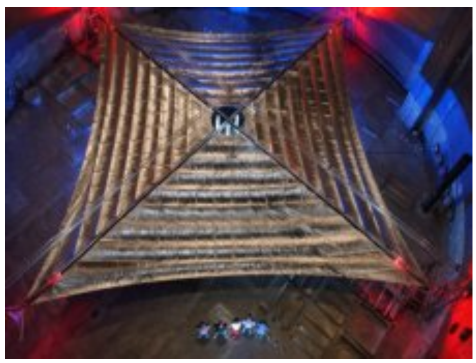


**Figure 18: An image of the 34m x 34m Sunjammer sail.**

Led by industry manufacturer L'Garde Inc. of Tustin, Calif., and including participation by the National Oceanic and Atmospheric Administration (NOAA), the Solar Sail Demonstration mission builds on two successful ground-deployment experiments conducted by L'Garde in 2005-2006 in a vacuum chamber at the Plum Brook Facility in Sandusky, Ohio, a research laboratory managed by NASA's Glenn Research Center in Cleveland. It also leverages the successful deployment of the NanoSail-D sail, a 100-square-foot test article NASA launched to Earth orbit in early 2011 to validate sail deployment techniques[ix].

### MISSION SUMMARY

The Sunjammer spacecraft was designed to perform a solar weather monitoring mission while "parked" at Sun-Earth L1. The sail will be used for station keeping as the instruments monitor solar activity.  Sunjammer should give NOAA and NASA an early warning for certain solar phenomena that might necessitate the proactive safing (switching spacecraft electrical systems into a "safe mode" to prevent interactions with solar phenomena) of certain assets in Earth orbit.

### SCALING SUNJAMMER

We decided to investigate the capabilities of a Sunjammer-like sail that was 250m x 250m. This is roughly the scaling limit of the Sunjammer sail manufacture, deployment, and control methods.

## SAIL MATERIAL CHOICES

### PRESENT SAIL MATERIALS

The Sunjammer spacecraft will use 5 micron thick Kapton film as a sail material. We performed an investigation into the materials available and determined that 5 micron Kapton and .9 micron Mylar are the two leading materials for this application. Mylar however does not survive radiation environments well and further study is required to determine whether this fully precludes the use of Mylar as a solar sail material.

### FUTURE SAIL MATERIALS

We contacted several manufacturers of Kapton (DuPont, 3M) and were told that 2 micron thick Kapton film is within manufacturing capability. We will see in a later section how this change affects Heliopause cruise times. The .5 micron thick Kapton film is theoretically possible but has not been significantly investigated because even 2 micron Kapton has yet to find a commercial application significant enough to justify the necessary modification to manufacturing facilities.

In parallel to the evaluation of DuPont's Kapton® films for the proposed application, we have identified other candidate materials to be studied in phase 2. This study will include CP-1, a potential successor to CP-1 known as CORIN™ XLS, and Thermalbright®[x]. CP-1 is a space-durable material developed at NASA Langley (LaRC) and exhibits a high resistance to UV radiation. It has currently been fabricated in large sheets and rolls to as thin as 1.5 micron. CP-1 has flown on Hughes HS-702 geosynchronous communications satellite. CORIN™ XLS is a potential next-generation CP-1. Thermalbright® polyimide is a high temperature highly reflective white polyimide film which is expected to be particularly beneficial for thermal control while maintaining good UV and VUV durability.

## SSEARS TRAJECTORIES

With our sail materials in mind, and an assumed spacecraft mass of 110kg, we modeled the transit time for sails with the different materials to 100AU. We also moved to a more realistic characteristic acceleration of .5 mm/s$^2$. We chose two sail sizes to compare, the 250m x 250m size that is the limit of current techniques, and a hypothetical 500m x 500m sail. Material choices are represented by the line color. We also estimated a scaling factor for the boom masses. We assumed two methods of boom mass scaling – linear (here called aggressive) and geometric (here called conservative). This factor is represented by the line style. Solid lines represent a conservative boom mass scaling factor, and dashed lines represent an aggressive boom mass scaling factor. The curve plots in figures 19 and 20 represent these results. Finally, vertical black lines represent the distance to the sun below which the listed material begins to degrade.

The below charts were derived from the initial spacecraft mass assumption of 110kg. When we completed our detailed spacecraft configuration, we concluded that a more realistic spacecraft mass is 175kg. The transit times were relatively insensitive to this change in spacecraft mass.

## SSEARS MODEL PAYLOAD TRAJECTORIES WITH 250X250M SAIL

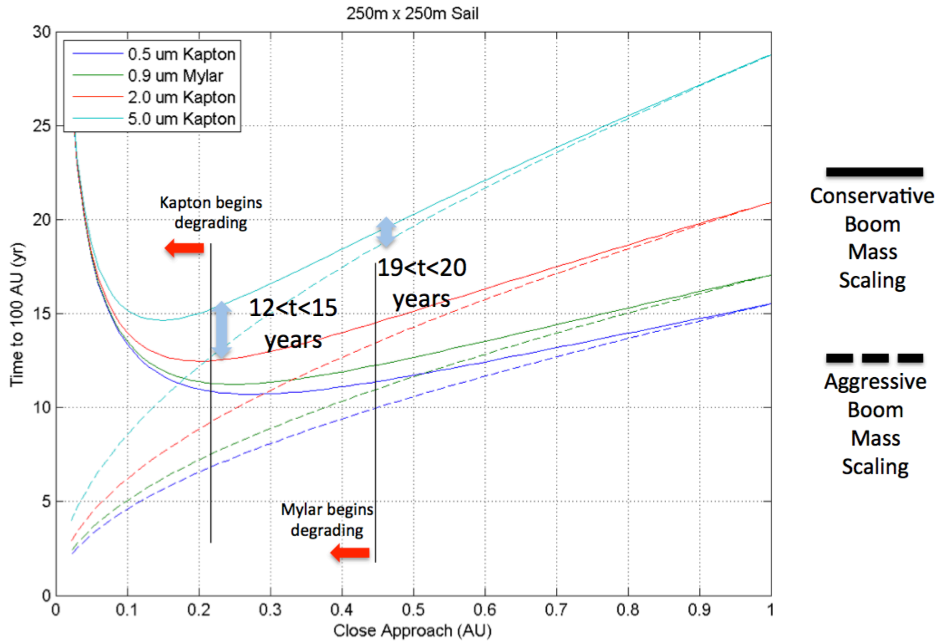



**Figure 19: We can see that 5 micron kapton, the material used by Sunjammer, provides the slowest velocity to the Heliopause. It cannot get closer than ~.22 AU from the sun and so the theoretical cruise time to 100AU is between 12 and 15 years depending on the boom scaling factor. The .9 micron thick mylar, which cannot get closer than .45 AU from the sun and is a difficult material to use from a radiation tolerance perspective, delivers the same spacecraft to the Heliopause in between 10 and 11 years.**

## SSEARS MODEL PAYLOAD TRAJECTORIES WITH 500X500M SAIL

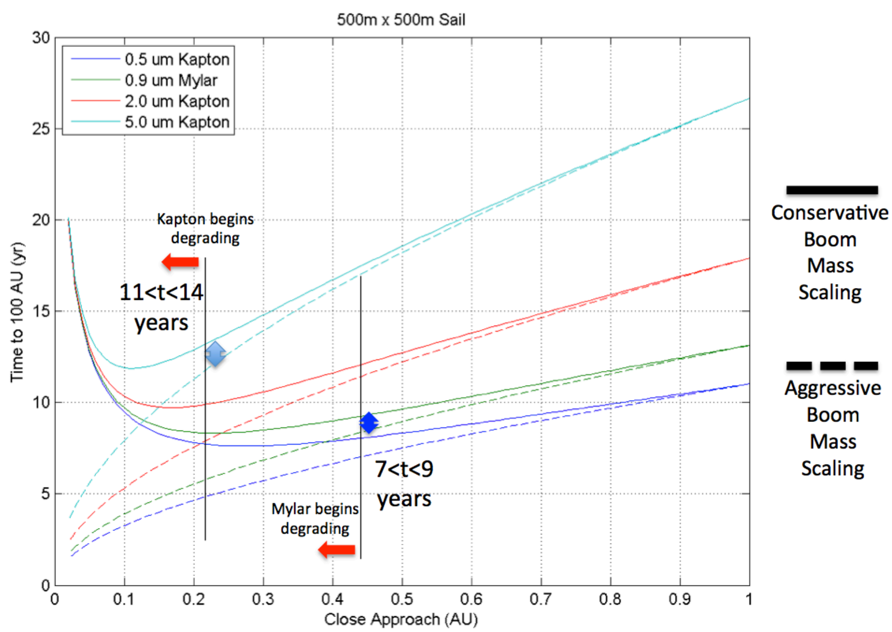



**Figure 20: With an even larger sail size we can see the trip times reduced by about a year, each. This result was very insightful because it shows that the additional complexity and cost of a 500m x 500m sail is probably not justified as little is gained by increasing the size of the sail beyond 250m x 250m.**

## OPTIMAL SAIL SIZE

Our investigation into the sail material showed us that a significant increase in sail size beyond 250m x 250m does not provide significant velocity improvement. Our flight system design study, then, assumes a 250m x 250m sail size.

## SAIL JETTISON

The sail would not receive much energy from the sun by the time it crosses Jupiter's orbit. At that time the sail would be turned and then jettisoned (in order to prevent the sail's trajectory from blocking communication from the spacecraft.) The spacecraft would continue alone. A series of renderings follow that demonstrate this process.

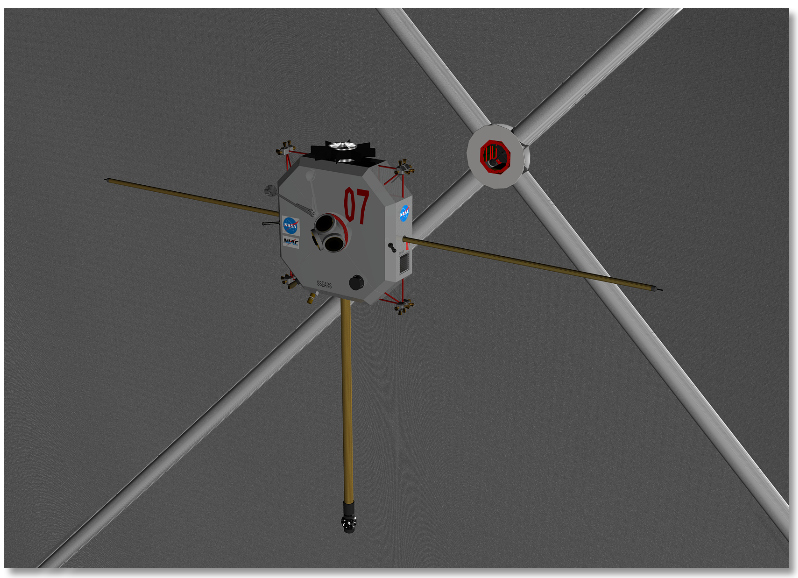



**Figure 21: The spacecraft separates from the sail and continues.**

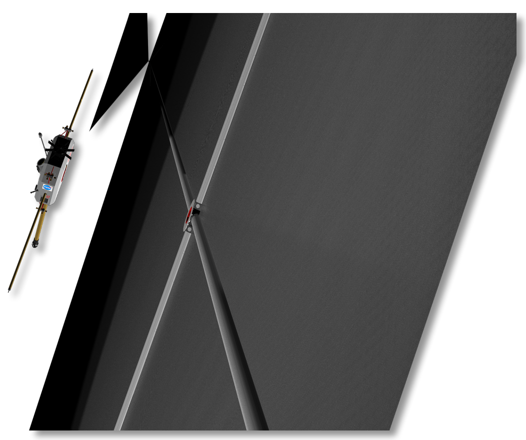


**Figure 22: Side view of the spacecraft following separation.**

# SPACECRAFT / FLIGHT SYSTEM

This section describes the spacecraft architecture that we designed to meet the requirements described up to this point in this report. We being with a discussion of the design approach and show several configuration graphics to demonstrate the stages of the mission.

## SPACECRAFT DESIGN

The guiding principle of the spacecraft design process was to leverage existing technology as much as possible. This involved scaling up the nominal Sunjammer design concept and using high heritage instruments. The sail would be folded and packed into a modular container. The spacecraft concept itself is visible on the foreground-facing side of the sailcraft container.

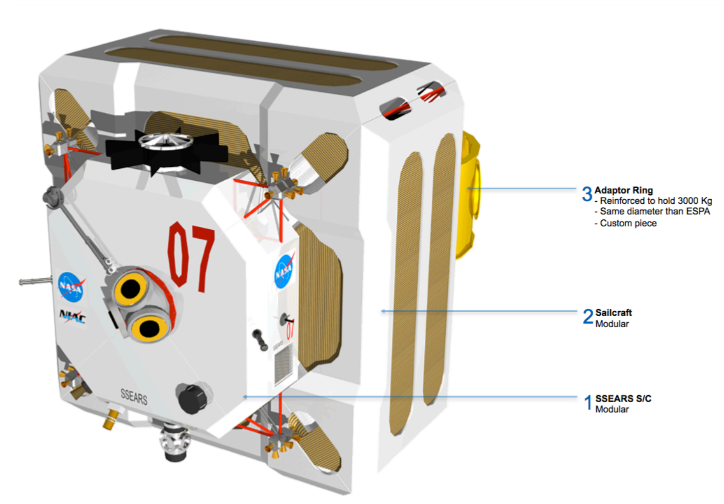



**Figure 23: Here the two main components of the module can be seen: the module itself and the sail craft. In the stowed mode 3 complete spacecraft can be stored inside the fairing of an Atlas V launch vehicle (5m in diameter).**

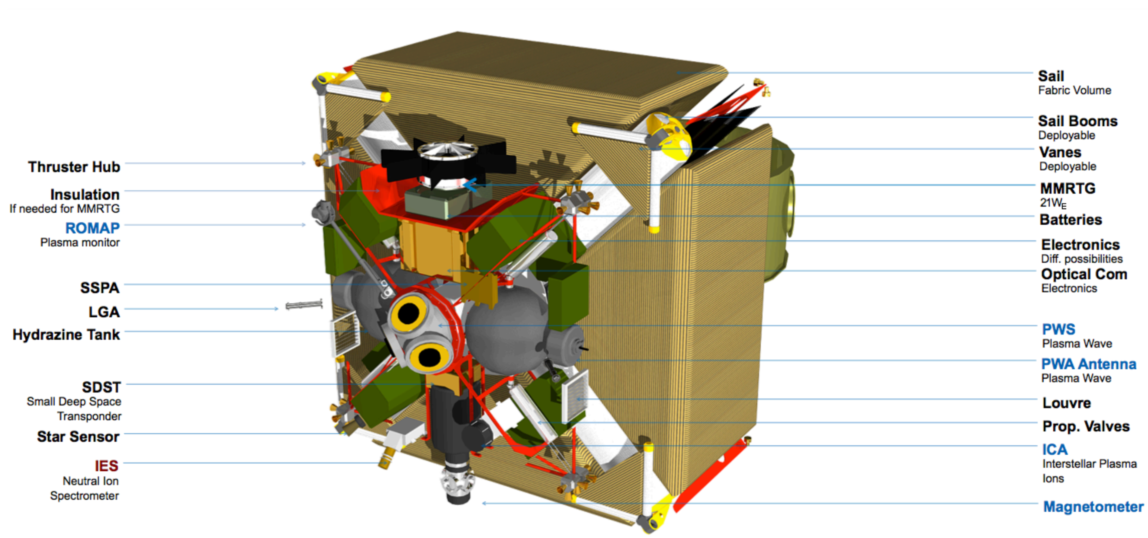



**Figure 24: Here the main selection of instruments can be seen as well as the internal structure and configuration. Some of the green volumes represent extra electronic boxes to either support current instruments or to be substituted by extra payload using a better packing ratio.**

The sail container deploys in this fashion:

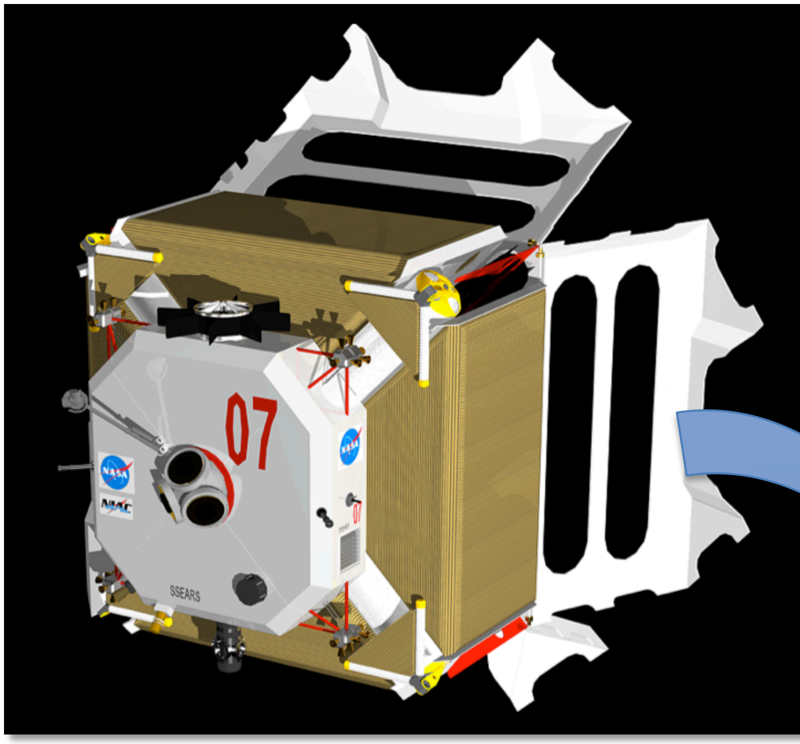
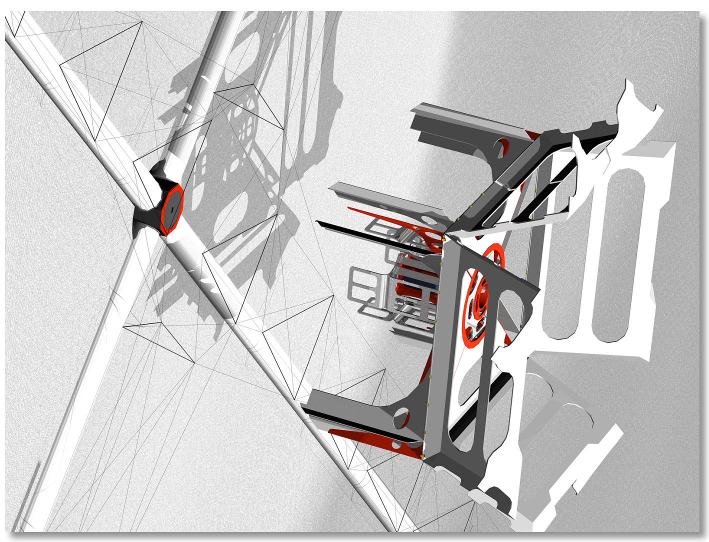



**Figure 25: Once SSEARS is detached from the launcher the sail protection covers open and the vanes are deployed. After that the sail is deployed. The next step is to deploy the booms for antennas and instruments.**

This component then jettisons after sail deployment leaving the spacecraft and sail separated.



The three major systems are visualized side-by-side for scale comparison below. The left side of the image shows a side view of the spacecraft, sail structure, and mounting hardware (left-to-right) and the right side of the image shows the spacecraft with the booms deployed.

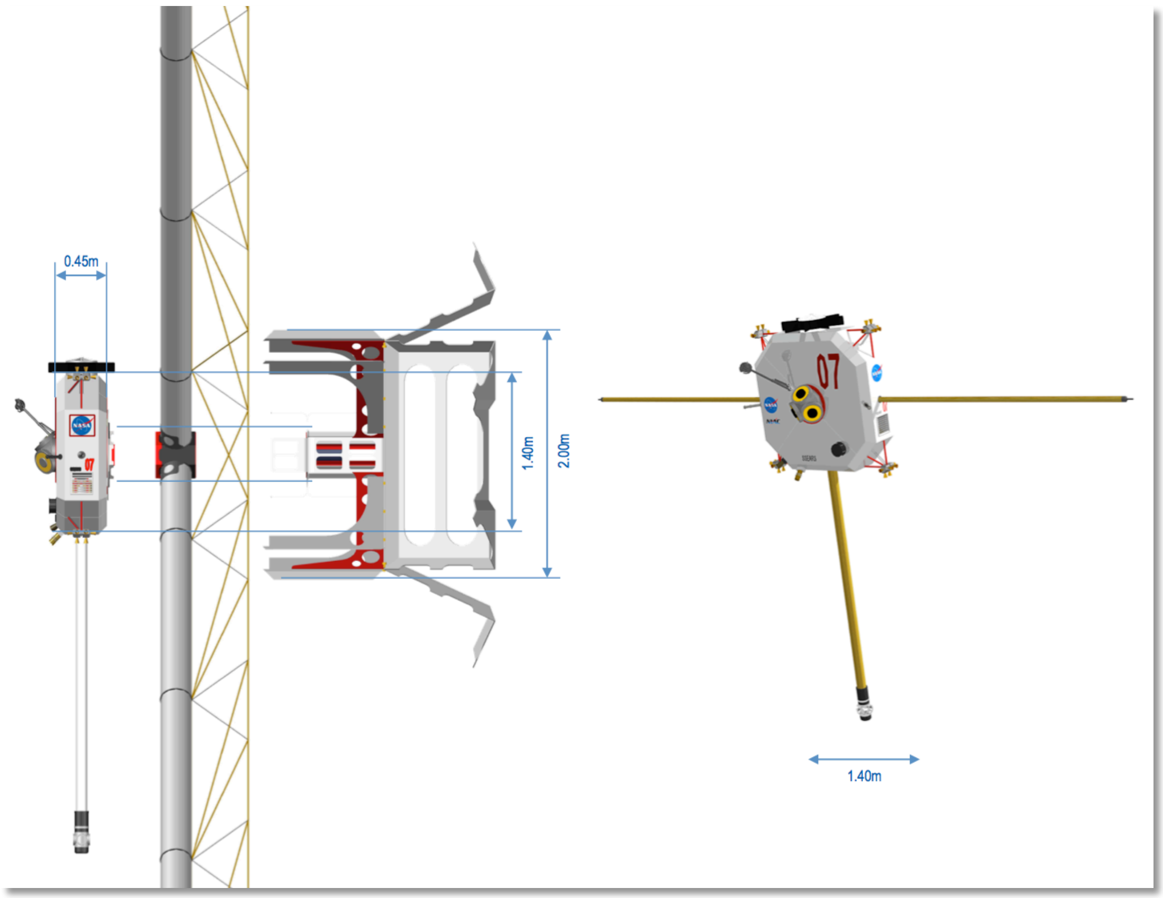



Figure 26: The three spacecraft elements are displayed side by side to provide a sense of relative scale.

Launch vehicle compatibility with Atlas 5 501 was shown for 3 units per Atlas 5.

## COMMUNICATION

We determined early in our study that a large dish antenna communication system would be so massive (as necessary to function from 100AU) that it would severely constrain the design of the rest of the spacecraft. We determined that optical communication can meet our needs.

Downlink

Laser beams at optical frequencies can be transmitted with angular beam-widths of a few microradians (μrads). Coupled with the ability to accurately point narrow laser beams to a fraction of the beam-width, signal power densities required for communication can be delivered over huge distances. To illustrate this further a point design is presented below.

A 20 watt average power 1550 nm laser beam transmitted through a 50 cm diameter telescope will result in an angular beam-width of approximately 3.6 μrad. Pointing jitter control of ~ 0.3 μrad will result in losses relative to the on-axis peak of ~ 1.5 dB. If such a beam with the indicated pointing control were transmitted over a distance of 100 AU with a 12 m diameter collector at the receiving end a 500+/-100 bits/second communication link could be established. This accounts for reasonable transmitter and receiver optical transmission and implementation losses and 3-dB of link margin. Such a link is enabled by the availability of high peak-to-average power ratio (160:1 to 320:1) lasers that permit the use of near-capacity achieving serially concatenated pulse-position modulation (SCPPM) error-correcting coding together with a single-photon-counting sensitivity receiver.

**Phase II Plans**

A high level summary of the studies for maturing the concept of Optical Communication from a distance of 100 AU to Earth are tabulated in Table 1 below.

| | Study Topic | Approach |
|---|---|---|
| 1 | Acquisition and tracking strategy | Sun as beacon combined with knowledge generated on-board of Earth position and phase |
| 2 | Concept of operations | Strategies needed to operate with > 1 day round-trip light time |
| 3 | Terminal Architecture Requirements | Emphasis on special environmental and lifetime requirements |

Link acquisition and pointing strategy based on the remote spacecraft terminal utilizing the Sun as a reference to off-point to Earth using on-board generated position and phase knowledge, will be part of our phase 2 study. Note that off-pointing is mandatory since the Sun-Earth angular separation of 10 mrad from 100 AU is many laser beam diameters. Pointing must also account for point-ahead angles of 100-200 μrad or 30-40 laser beam widths where a beam-width of 3-6 μrad is being considered. Residual 1-sigma errors of 0.3-0.4 μrads must be achieved and maintained for the duration of the link. The objective of these studies will be to show through modeling and simulation that: (a) spacecraft disturbance rejection through passive and/or active means can be achieved and; (b) adequate control bandwidth is available with the sun beacon to maintain offset pointing to Earth within the allocated error.

The field-of-view (FOV) of the acquisition sensor used on the spacecraft would be relatively large in order to overlap the angular separation between the Sun and Earth. Roughly ±10 mrad will be required. At the same time the instantaneous field-of-view (IFOV) should be approximately of the order of the transmitted laser beam-width i.e. close to 10 μrad. This suggests a 1000 x 1000 focal plane array, however, the viability of implementing such a sensor needs further study. Even at 100 AU the sun angular diameter is ~ 90 μrad and this will have to be factored into achieving the targeted residual 0.3-0.4 μrad pointing error. The feasibility and benefits of using other celestial objects will be explored.

On the ground the beam footprint from 100 AU would be approximately 4 Earth diameters, however, the irradiance would be a few femtowatts per square meter. A 12 m or larger collector or an array of large collectors would be required to gather sufficient photons to overcome background and shot noise. In our preliminary analysis a 256 pixel superconducting nanowire array behind a 12 m collector supported 500±100 bits/sec. Ground aperture would very likely utilize a conical scan to acquire and lock on the downlink signal. The use of adaptive optics and aggressive filtering to discriminate the faint signal against prevalent background light will be studied. The performance achieved with the latter would determine how close to the Sun the ground terminal can point and this would also determine the duration of possible outages. For comparison the capability achievable with an orbiting large aperture collector/receiver will be determined where shot-noise limited performance can be achieved.

The concept of operations for a 100 AU optical communication link needs further study. From ranges comparable to where Voyager II spacecraft is at present, round-trip light times of approximately 27 hours (> 1 day) will be encountered. Since bi-directional acknowledgement and re-transmission requests will not be viable with such long ranges, optical communication from 100 AU will involve repetitive data transmission over a sufficiently long duration to ensure signal detection and data reception at Earth. While similar to the strategy used by Voyagers downlinking data to DSN, optical communications would require additional considerations related to cloud-cover, number of available ground stations and their distribution around the globe. In our proposed study these considerations will be addressed and the autonomous capability required by the spacecraft to implement the concept of operations will be evaluated. Emphasis on spacecraft resources needed to support optical communications will also be assessed.

The suitability of space terminal architectures pursued for deep-space optical communication within the solar system will be re-examined for the expected environment at 100 AU, as well as, expected longer lifetime demands. The enhanced acquisition field-of-view mentioned earlier is an example of a needed modification. The spectral characteristics of the sun beacon will also result in optical design modifications. Depending on the concept of operations the command receiving architecture and data-buffering requirements will very likely require modifications. Lifetime demands may impose added redundancy and in some cases re-engineering of components or assemblies. The radiation environment and shielding requirements must also be factored in.

Uplink

For our baseline configuration we assume that the spacecraft would perform autonomously until it fails. In the case of our 100 AU mission this makes sense because there are no encounters, there are no maneuvers, and navigation is moot once the spacecraft has passed the sun. The spacecraft does have a small low-gain antenna for near-Earth communication. We will further investigate the uplink strategy in Phase 2.

## INSTRUMENT SUITE FOR MODEL SPACECRAFT

The following table shows the baseline instrument suite selected for this spacecraft design. Other instrument suites will be developed and configured as part of the Phase 2 work.

| Instrument | Heritage |
|---|---|
| Magnetometer | Cassini |
| Plasma Monitor | ROMAP - Rosseta |
| Plasma Wave | PWS - Voyayer |
| Neutral Ion Spectrometer | IES - Rosetta |
| Interstellar Plasma Ion Detector | ICA - Rosetta |

This suite primarily focuses on the interactions between the solar wind and the local interstellar medium. Other spacecraft would carry instruments intended for study of the galactic wind, interstellar dust, and galactic cosmic rays as described in the next section. One of the advantages of the spacecraft architecture is that duplicate spacecraft could be built with different instruments. We expect to employ the following instruments in other instrument suites (to be further defined in phase 2:

1. Energetic particle monitor (>500kev)
2. Anomlaous cosmic rays
3. Dust detector
4. Galactic cosmic ray detector (<10 MeV/nuc, 10-100, MeV, and >100 MeV)

## MASS EQUIPMENT LIST (MEL)

The initial estimate of spacecraft mass of 150 kg was close to the mass computed by the detailed spacecraft design: ~150kg. Accounting for margin, a spacecraft mass of 175 kg is used for our trajectory calculations and our spacecraft mass came to less than 170 kg even when fueled and with 10% system margin.

## SSEARS Mass Equipment List (MEL)

| | Mass (kg) | Units | CBE (kg) | Contin-gency | Expected Mass (kg) | Heritage | Comments |
|---|---|---|---|---|---|---|---|
| **Spacecraft** | | | | | | | |
| **Science Instruments** | | | 6.30 | 30% | 8.19 | | |
| Magnetometer | 0.70 | 1 | 0.70 | 30% | 0.91 | Cassini | |
| Plasma Monitor | 1.00 | 1 | 1.00 | 30% | 1.30 | Rosseta | ROMAP |
| Plasma Wave | 1.40 | 1 | 1.40 | 30% | 1.82 | Voyager | PWS |
| Neutral Ion Spectrometer | 1.00 | 1 | 1.00 | 30% | 1.30 | Rosseta | IES |
| Interstellar Plasma Ions | 2.20 | 1 | 2.20 | 30% | 2.86 | Rosseta | ICA |
| **Structure and Mechanisms** | | | 33.00 | 28% | 42.30 | | |
| Bus structure | 15.00 | 1 | 15.00 | 30% | 19.50 | | Composite honeycomb panels |
| Secondary structure (incl. booms) | 7.00 | 1 | 7.00 | 30% | 9.10 | | |
| Cabling harness | 5.00 | 1 | 5.00 | 30% | 6.50 | | |
| System Assembly Hdw. | 4.00 | 1 | 4.00 | 30% | 5.20 | | |
| Ballast | 2.00 | 1 | 2.00 | | 2.00 | | |
| **C&DH** | | | 8.50 | 10% | 9.38 | | |
| Spacecraft Flight Computer (SFC) | 0.60 | 1 | 0.60 | 5% | 0.63 | SMAP | |
| Non-Volatile Memory (NVM) | 0.70 | 1 | 0.70 | 5% | 0.74 | SMAP | |
| MSAP Telemetry Interface (MTIF) | 0.70 | 1 | 0.70 | 5% | 0.74 | SMAP | |
| MSAP System Interface Assy (MSIA) | 0.70 | 1 | 0.70 | 5% | 0.74 | SMAP | |
| Local Eng. Unit Digital Card (LEU-D) | 0.70 | 1 | 0.70 | 5% | 0.74 | SMAP | |
| Local Eng. Unit Analog Card (LEU-A) | 0.70 | 1 | 0.70 | 5% | 0.74 | SMAP | |
| CDH Elect. Power Conv. Unit (CEPCU1) | 1.10 | 1 | 1.10 | 5% | 1.16 | SMAP | |
| Critical Relay Controller Card (CRCC) | 0.30 | 1 | 0.30 | 5% | 0.32 | SMAP | |
| Chassis and backplane (incl. Power cards) | 3.00 | 1 | 3.00 | 20% | 3.60 | SMAP | Holds both C&DH and Power |
| **ACS** | | | 3.65 | 16% | 4.25 | | |
| Star Tracker/Scanner | 2.70 | 1 | 2.70 | 20% | 3.24 | SMAP | |
| Sun Sensor | 0.10 | 2 | 0.20 | 10% | 0.22 | SMAP | |
| LN-200S IMU | 0.75 | 1 | 0.75 | 5% | 0.79 | MSL | |
| **Power** | | | 17.62 | 20% | 21.18 | | |
| Small MMRTG | 10.32 | 1 | 10.32 | 30% | 13.42 | | 21 $W_E$ BOL, 250 $W_t$ |
| Battery | 3.00 | 1 | 3.00 | 5% | 3.15 | MER | 150 Wh |
| Power Switch Slice (PSS) | 1.90 | 1 | 1.90 | 10% | 2.09 | SMAP | |
| Housekeeping Power Cond. Unit (HPCU) | 1.00 | 1 | 1.00 | 5% | 1.05 | SMAP | |
| Power Bus Controller (PBC) | 1.40 | 1 | 1.40 | 5% | 1.47 | SMAP | |
| **Telecom** | | | 30.68 | 26% | 38.61 | | |
| Small Deep Space Transponer (SDST) | 2.65 | 1 | 2.65 | 5% | 2.78 | MSL | |
| X-Band SSPA | 1.35 | 1 | 1.35 | 5% | 1.42 | MSL | 17 $W_{RF}$ |
| Coax Transfer Switch (CXS) | 0.10 | 1 | 0.10 | 10% | 0.11 | | |
| Filter | 0.28 | 1 | 0.28 | 10% | 0.31 | | |
| Coax Cable | 0.30 | 1 | 0.30 | 30% | 0.39 | | |
| Low Gain Antenna (LGA) | 0.50 | 2 | 1.00 | 10% | 1.10 | SMAP | |
| Optical Com System | 25.00 | 1 | 25.00 | 30% | 32.50 | | 75 $W_E$ input power |
| **Propulsion** | | | 8.90 | 14% | 10.18 | | |
| Hydrazine tank | 0.80 | 2 | 1.60 | 30% | 2.08 | ATK custom | 10 liter tank, assume ½ press. |
| Thrusters | 0.40 | 8 | 3.20 | 5% | 3.36 | | 4 N monoprop thruster |
| Valves | 0.30 | 4 | 1.20 | 10% | 1.32 | | |
| Filter | 0.50 | 1 | 0.50 | 5% | 0.53 | | |
| Propellant lines and components | 1.50 | 1 | 1.50 | 30% | 1.95 | | |
| Sensors | 0.30 | 3 | 0.90 | 5% | 0.95 | | |
| **Thermal Control** | | | 7.70 | 30% | 10.01 | | |
| Multilayer Insulation (MLI) | 0.38 | 10 | 3.80 | 30% | 4.94 | | |
| Heaters | 0.05 | 20 | 1.00 | 30% | 1.30 | SMAP | |
| Sensors | 0.01 | 30 | 0.30 | 30% | 0.39 | SMAP | |
| Louvers | 1.30 | 2 | 2.60 | 30% | 3.38 | | |
| **S/C Dry Mass** | | | 116.35 | **24%** | 144.09 | | |
| Hydrazine Propellant | 10.00 | 1 | 10.00 | | 10.00 | | |
| **S/C Wet Mass** | | | **126.35** | 22% | **154.09** | | |
| **S/C Wet Mass with 10% system margin** | | | | | **168.50** | | |
| **Sail Module** | 625.00 | 1 | 625.00 | 30% | 812.50 | Sunjammer | σ=10 g/m², A=62,500 m² |
| **Inflation Module** | 50.00 | 1 | 50.00 | 30% | 65.00 | Sunjammer | Incl. adapter & sep'n. hdw. |
| **TOTAL LAUNCH CBE+Contingecy** | | | **801.35** | **50%** | **1,200.08** | | |

With that spacecraft mass the transit times for different sail materials are provided in Figure 27. Boom mass scaling figures apply as in previous graphs (solid line is conservative scaling, dashed line is aggressive scaling.) 2 micron thick kapton is the material of choice both from a near-mid term (~2020s) feasibility standpoint and a cruise time standpoint. With this material cruise times range between 15 and 18 years. However 2 micron Kapton has yet to find a commercial application sufficient to motive manufacturers to develop it. With enough time .5 micron thick Kapton will emerge as the better choice, further reducing cruise times to between 12 and 15 years.

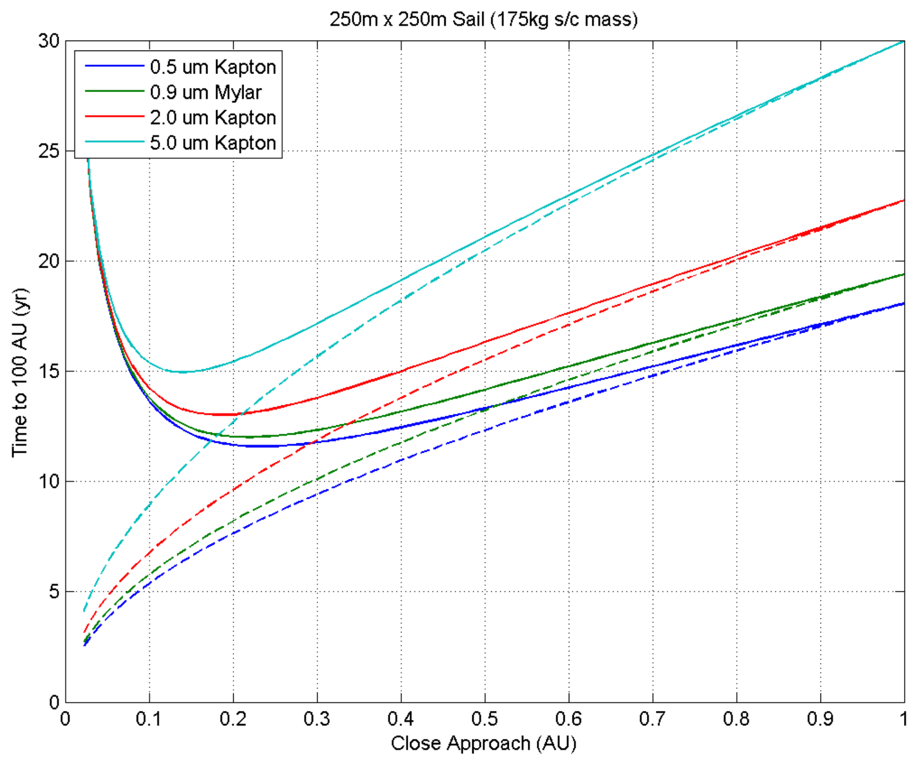



**Figure 27: Our estimated spacecraft mass of 175kg raises the transit time somewhat, and shows that 2 micron Kapton is a clear choice for sail material. A phase two task will be to discuss the manufacturing of 2 micron thick Kapton with various manufacturers.**

## THERMAL CONTROL CHALLENGES

The spacecraft would face thermal control challenges both at perihelion and in deep space. The spacecraft concept uses louvers, shielding and insulation as accounted for in the MEL. There is an external wall of aluminum between the sailcraft and the spacecraft itself that would ensure there is a shadow on the spacecraft when the sun is behind the spacecraft. Regarding thermal tolerance as the spacecraft approaches the sun, it is unclear how close the spacecraft can get to the sun, and how long the spacecraft can stay there. Part of our phase 2 study will be to determine specific trajectories, their time spent in the high heat and radiation environment near the sun, and the extent to which the short duration spent at those distances may, to some degree, alleviate thermal and radiation exposure concerns. As a baseline power source, the small RPS would be mounted to the spacecraft near a central position and the waste heat would be used to heat the spacecraft in deep space.

# COST

The SSEARS cost estimate for a solar-sail propelled spacecraft, designed to perform fields and particles research at the Heliopause region, was performed using a variety of tools and methodologies. Overall, two distinct flight system cost scenarios were developed, a ten-spacecraft mission concept and a smaller three-spacecraft mission concept, which is considered the descope option. JPL's Parametric Mission Cost Model (PMCM), which uses statistically derived cost estimating relationships in order to estimate the full breadth of the JPL mission work breakdown structure (WBS), was utilized in the estimate. These costs were supplemented with instrument costs, developed using analogous instruments flown on previous missions. L'Garde Inc. provided a ROM (rough order of magnitude cost) for the 250m x 250m solar sail. The launch vehicle cost was determined using ROM costs provided by the Launch Services Program at Kennedy Space Center (KSC). Subsequently, a cost risk analysis was performed using ACEIT to assess a confidence level in the estimate. All estimates were performed in FY13 constant year dollars.

Two mission sizes were considered – the original concept of 10 spacecraft and a descoped concept with three spacecraft.

**Cost Estimate Input Assumptions**

**Overall Mission:** The SSEARS mission is assumed to be Class A mission with dual cold redundancy. Consistent with JPL reserve policy, 30% reserves are assumed through Phase D, and 15% reserves are assumed in Phase E. A team of 6.5 (FTE) scientists is required through launch; 2.5 (FTE) scientists are assumed to be adequate post-launch. The formulation phase (Phases A/B) is assumed to be 18 months; Phases C/D are assumed to be 40 months, followed by a 15 year voyage to the Heliopause.

**Spacecraft and Solar Sail:** For both the three-spacecraft and ten-spacecraft scenarios, a MMRTG power source and optical communication are the baseline design selections; solar sails and a monoprop hydrazine blowdown configuration serve as the primary propulsion subsystem. Subcontracting of the spacecraft is also assumed in each case. A discount of 50% is taken into account after the first spacecraft unit is produced, as the additional spacecraft are exact replicas of the first unit.

L'Garde Inc provided a solar sail ROM[xi] cost based on their experience with costs incurred while building the Sunjammer 1200m$^2$ solar sail and sailcraft. Their proven expertise with Sunjammer gives the team a high level of confidence in L'Garde's estimate. The $53M cost provided by L'Garde includes the separation and inflation modules, solar sail membrane and structure, as well as their integration. An additional 17% JPL fee is assessed in addition to the quote from L'Garde. A discount of 20% is taken after the first unit.

**Instruments**: A core payload with five instruments is assumed for both mission scenarios. Cost estimates were developed for the core payload. A Voyager-based magnetometer, plasma monitor, plasma wave, and neutral ion spectrometer are used as analogous instruments for the cost basis. The TIDE instrument on the POLAR mission is used as an analogy-base for the interstellar plasma ion instrument. It is assumed that the effort, and thus the cost, will decrease by a factor of 65% after the first unit of each instrument produced, as they are exact replicas of the first instrument units. In the case of the 10 spacecraft scenario, there are 50 instruments in total. There are 15 instruments in total for the 3 spacecraft scenario.

Cost estimates were also developed for two optional instruments, a Channeltron, which will complement the Plasma Monitor, and a Cosmic Ray instrument.

**Launch Vehicle:** For the ten-spacecraft scenario, an Atlas V 551 throughput cost is used which includes launch services, payload processing, launch vehicle integration mission, unique launch site ground support and tracking, data and telemetry services, as provided in the payload planner's guide from the Launch Services program at KSC. For the three-spacecraft scenario, an Atlas V 501 cost was used.

### Cost Estimate Output

The cost output is based on the input assumptions discussed previously. The relative magnitudes of the ten-spacecraft cost scenario versus the three-spacecraft cost scenario are in alignment. Due to the large amount of fixed cost, one would not expect a direct linear relationship with the number of spacecraft.

**Project Total (FY 2013 $M)**

| | | | PMCM Model Output: 10 Spacecraft | PMCM Model Output: 3 Spacecraft |
|---|---|---|---|---|
| **Total Project Costs ($M)** | | | **$3447.0M** | **$1796.7M** |
| 1.0 | Project Management | | $67.2M | $51.2M |
| 2.0 | Project System Engineering | | $17.1M | $17.1M |
| 3.0 | Mission Assurance | | $81.1M | $35.5M |
| 4.0 | Science Team | | $29.8M | $29.8M |
| 5.0 | Instruments | | $94.7M | $42.3M |
| 6.0 | Flight System | | $1426.2M | $515.7M |
| 6.1 | | Flight System Management | $9.7M | $9.7M |
| 6.2 | | Flight System Engineering | $15.5M | $14.9M |
| 6A | | Starjammer | $876.0M | $318.5M |
| 6B | | L'Garde Sailcraft | $516.1M | $163.6M |
| 6.13 | | Materials and Processes | $0.6M | $0.6M |
| 6.14 | | Flight System Testbeds | $8.4M | $8.4M |
| 7.0 | Mission Operations System | | $519.8M | $519.8M |
| 8.0 | Launch System | | $255.6M | $172.9M |
| 9.0 | Ground Data System (Currently included under 7.0) | | | |
| 10.0 | Project Systems I&T | | $245.9M | $74.1M |
| 10A | | ATLO - Starjammer | $194.8M | $58.1M |
| 10B | | ATLO - L'Garde Sailcraft | $51.1M | $16.0M |
| 11.0 | Education & Public Outreach | | $24.9M | $12.9M |
| 12.0 | Mission Design | | $8.6M | $8.6M |
| | Reserves | | $676.1M | $316.7M |

**Figure 28: Cost breakdown by Work Breakdown Structure**

**Instrument Options:** Two instruments were evaluated for inclusion in the SSEARS mission as options, a Channeltron and a Cosmic Ray instrument. The marginal cost of including the Channeltron and the Cosmic Ray instrument on the ten-spacecraft mission scenario is $20.9M and $15.3M, respectively. These are not included in the estimates above; they are merely options at this point.

**Cost-Risk Analysis:** Uncertainty and risk exist in the estimate, as expected with such a unique mission. The order quantity discounts are aggressive, but fair, given the number of exact replicas of each spacecraft, sail, and instruments required. Additionally, the assumed schedules for both the ten-spacecraft and three-spacecraft scenarios are the same, drawing heavily on the learning curve/repetition assumptions inherent in building multiple units of the same system.

To understand the risk posture of the mission, a mission-level output based s-curve was generated for each scenario in ACEIT. A lognormal cost distribution was assumed, as it has been proven that space system cost typically follows a lognormal distribution. The PMCM total output without reserves was used as the most likely point estimate. A coefficient of variation of 0.4 and 0.45 was used for the three spacecraft and ten spacecraft scenarios respectively, which is higher than the space systems industry standard of 0.35, so that the higher level of uniqueness and risk in the SSEARS mission is captured. Both scenarios attained approximately a 50% cost confidence level, meaning that the probability that the mission actual cost will be less than the cost estimate with reserves is 50%. The risk analysis for the ten spacecraft scenario follows; the three spacecraft risk analysis may be found in the appendix.

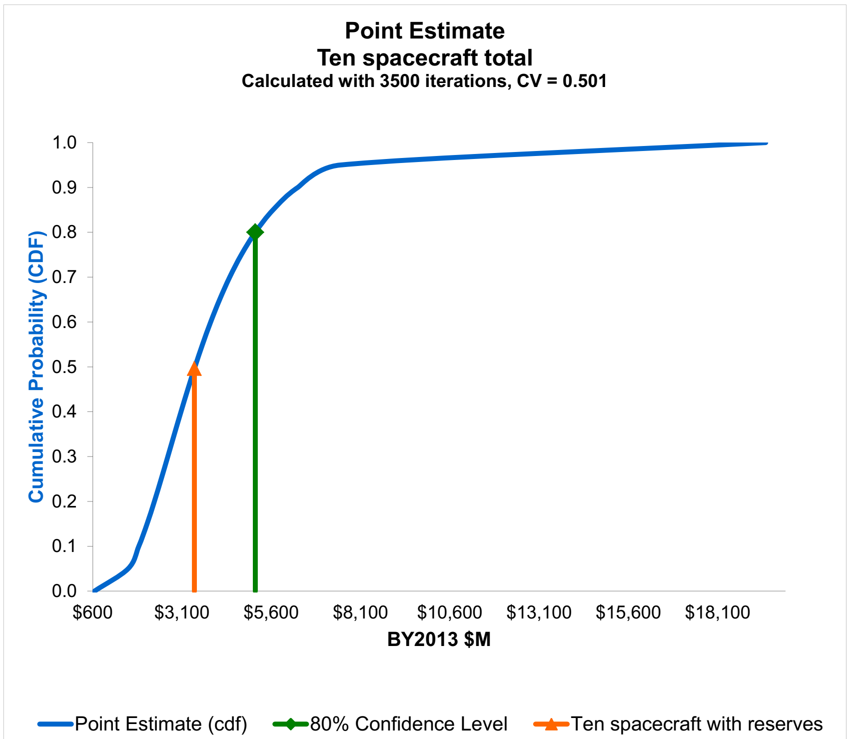



**Figure 29: mission-level output based s-curve**

| Markers | Costs | Confidence |
|---|---|---|
| 80% Confidence Level | $5,153.189 | 80.0% |
| Ten spacecraft with reserves | $3,447.000 | 49.6% |

The cost estimate of $3447M, for a ten spacecraft mission, accounts for a substantial amount of uncertainty and risk, consistent with a space mission at this stage in its definition. The estimate will continue to be refined and risk reassessed as the mission evolves.

Final cost outputs are as follows for both the ten-spacecraft mission and the three-spacecraft mission:

Project Summary (JPL WBS)

| | | Formulation (Phase A/B) ($M) | Implementation (Phase C/D) ($M) | Development Total ($M) | Operations (Phase E) ($M) | Project Total (FY 2013 $M) | Notes/Models |
|---|---|---|---|---|---|---|---|
| Total Project Costs ($M) | | $109.5M | $935.8M | $1218.2M | $578.5M | $1796.7M | |
| 1.0 | Project Management | $6.0M | $31.3M | $37.3M | $13.9M | $51.2M | |
| 1.1 | Project Manager & Staff | $1.6M | $4.2M | $5.9M | $9.0M | $14.9M | |
| 1.2 | Business Management | $1.6M | $4.7M | $6.4M | $4.4M | $10.7M | |
| 1.4 | Project Reviews | $0.3M | $0.7M | $0.9M | $0.5M | $1.4M | |
| 1.6 | Launch Approval | $2.4M | $21.7M | $24.1M | | $24.1M | |
| 2.0 | Project System Engineering | $3.2M | $13.9M | $17.1M | $0.0M | $17.1M | |
| 2.1 | Project Systems Engineering | $1.0M | $3.4M | $4.4M | | $4.4M | |
| 2.2 | Project Software Engineering | $0.8M | $1.8M | $2.6M | | $2.6M | |
| 2.3 | End-To-End Information System | $0.1M | $1.1M | $1.2M | | $1.2M | |
| 2.4 | Information Systems Engineering & Communications | $0.4M | $1.2M | $1.6M | | $1.6M | |
| 2.5 | Configuration Management | $0.3M | $1.2M | $1.5M | | $1.5M | |
| 2.6 | Planetary Protection | $0.0M | $0.1M | $0.1M | | $0.1M | |
| 2.7 | Contamination Control | $0.1M | $0.8M | $0.9M | | $0.9M | |
| 2.9 | Launch System Integration | $0.2M | $1.0M | $1.2M | | $1.2M | |
| 2.10 | Project V & V | $0.1M | $3.2M | $3.4M | | $3.4M | |
| 2.11 | Risk Management | $0.1M | $0.2M | $0.3M | | $0.3M | |
| 3.0 | Mission Assurance | $3.1M | $27.7M | $30.8M | $4.7M | $35.5M | Msn Assurance wrap excludes reserves, LV, and itself |
| 4.0 | Science Team | $1.3M | $11.5M | $12.8M | $17.0M | $29.8M | |
| 5.0 | Instruments | $4.2M | $38.1M | $42.3M | | $42.3M | |
| 6.0 | Flight System | $51.6M | $464.1M | $515.7M | $0.0M | $515.7M | |
| 6.1 | Flight System Management | $1.0M | $8.7M | $9.7M | | $9.7M | |
| 6.2 | Flight System Engineering | $1.5M | $13.4M | $14.9M | | $14.9M | |
| 6A | Starjammer- 3 copies | $31.9M | $286.7M | $318.5M | | $318.5M | |
| 6B | L'Garde Sailcraft - 3 copies | $16.4M | $147.3M | $163.6M | | $163.6M | |
| 6C | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 6D | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 6E | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 6.13 | Materials and Processes | $0.1M | $0.5M | $0.6M | | $0.6M | |
| 6.14 | Flight System Testbeds | $0.8M | $7.5M | $8.4M | | $8.4M | |
| 7.0 | Mission Operations System | $6.2M | $55.8M | $62.0M | $457.8M | $519.8M | |
| 8.0 | Launch System | | | $172.9M | | $172.9M | |
| 8.1 | Launch Vehicle | | | $172.9M | | $172.9M | |
| 8.2 | Upper Stage / SRM | | | $0.0M | | $0.0M | |
| 9.0 | Ground Data System (Currently included under 7.0) | | | | | | |
| 10.0 | Project Systems I&T | $7.4M | $66.7M | $74.1M | | $74.1M | |
| 10A | ATLO - Starjammer- 3 copies | $5.8M | $52.3M | $58.1M | | $58.1M | |
| 10B | ATLO - L'Garde Sailcraft - 3 copies | $1.6M | $14.4M | $16.0M | | $16.0M | |
| 10C | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 10D | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 10E | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 11.0 | Education & Public Outreach | $0.3M | $2.9M | $3.2M | $9.7M | $12.9M | EPO wrap includes everything except reserves, LV, and itself |
| 12.0 | Mission Design | $0.9M | $7.8M | $8.6M | $0.0M | $8.6M | |
| #.# | Other | | | $0.0M | | $0.0M | |
| #.# | Reserves | $25.3M | $216.0M | $241.2M | $75.5M | $316.7M | 15% |

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**Figure 30: Final mission cost for 3 spacecraft: $1.796B**

Project Summary (JPL WBS)

| WBS | Element | Formulation (Phase A/B) ($M) | Implementation (Phase C/D) ($M) | Development Total ($M) | Operations (Phase E) ($M) | Project Total (FY 2013 $M) | Notes/Models |
|---|---|---|---|---|---|---|---|
| Total Project Costs ($M) | | $265.1M | $2324.2M | $2844.8M | $602.2M | $3447.0M | |
| 1.0 | Project Management | $7.4M | $34.3M | $41.7M | $25.4M | $67.2M | |
| 1.1 | Project Manager & Staff | $2.5M | $5.1M | $7.6M | $14.5M | $22.0M | |
| 1.2 | Business Management | $1.9M | $6.5M | $8.4M | $10.0M | $18.4M | |
| 1.4 | Project Reviews | $0.5M | $1.1M | $1.6M | $1.0M | $2.6M | |
| 1.6 | Launch Approval | $2.4M | $21.7M | $24.1M | | $24.1M | |
| 2.0 | Project System Engineering | $3.2M | $13.9M | $17.1M | $0.0M | $17.1M | |
| 2.1 | Project Systems Engineering | $1.0M | $3.4M | $4.4M | | $4.4M | |
| 2.2 | Project Software Engineering | $0.8M | $1.8M | $2.6M | | $2.6M | |
| 2.3 | End-To-End Information System | $0.1M | $1.1M | $1.2M | | $1.2M | |
| 2.4 | Information Systems Engineering & Communications | $0.4M | $1.2M | $1.6M | | $1.6M | |
| 2.5 | Configuration Management | $0.3M | $1.2M | $1.5M | | $1.5M | |
| 2.6 | Planetary Protection | $0.0M | $0.1M | $0.1M | | $0.1M | |
| 2.7 | Contamination Control | $0.1M | $0.8M | $0.9M | | $0.9M | |
| 2.9 | Launch System Integration | $0.2M | $1.0M | $1.2M | | $1.2M | |
| 2.10 | Project V & V | $0.1M | $3.2M | $3.4M | | $3.4M | |
| 2.11 | Risk Management | $0.1M | $0.2M | $0.3M | | $0.3M | |
| 3.0 | Mission Assurance | $7.6M | $68.7M | $76.4M | $4.7M | $81.1M | Msn Assurance wrap excludes reserves, LV, and itself |
| 4.0 | Science Team | $1.3M | $11.5M | $12.8M | $17.0M | $29.8M | |
| 5.0 | Instruments | $9.5M | $85.2M | $94.7M | | $94.7M | |
| 6.0 | Flight System | $142.6M | $1283.6M | $1426.2M | $0.0M | $1426.2M | |
| 6.1 | Flight System Management | $1.0M | $8.7M | $9.7M | | $9.7M | |
| 6.2 | Flight System Engineering | $1.5M | $13.9M | $15.5M | | $15.5M | |
| 6A | Starjammer- 10 copies | $87.6M | $788.4M | $876.0M | | $876.0M | |
| 6B | L'Garde Sailcraft - 10 copies | $51.6M | $464.5M | $516.1M | | $516.1M | |
| 6C | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 6D | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 6E | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 6.13 | Materials and Processes | $0.1M | $0.5M | $0.6M | | $0.6M | |
| 6.14 | Flight System Testbeds | $0.8M | $7.5M | $8.4M | | $8.4M | |
| 7.0 | Mission Operations System | $6.2M | $55.8M | $62.0M | $457.8M | $519.8M | |
| 8.0 | Launch System | | | $255.6M | | $255.6M | |
| 8.1 | Launch Vehicle | | | $255.6M | | $255.6M | |
| 8.2 | Upper Stage / SRM | | | $0.0M | | $0.0M | |
| 9.0 | Ground Data System (Currently included under 7.0) | | | | | | |
| 10.0 | Project Systems I&T | $24.6M | $221.3M | $245.9M | | $245.9M | |
| 10A | ATLO - Starjammer- 10 copies | $19.5M | $175.3M | $194.8M | | $194.8M | |
| 10B | ATLO - L'Garde Sailcraft - 10 copies | $5.1M | $46.0M | $51.1M | | $51.1M | |
| 10C | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 10D | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 10E | | $0.0M | $0.0M | $0.0M | | $0.0M | |
| 11.0 | Education & Public Outreach | $0.6M | $5.6M | $6.2M | $18.7M | $24.9M | EPO wrap includes everything except reserves, LV, and itself |
| 12.0 | Mission Design | $0.9M | $7.8M | $8.6M | $0.0M | $8.6M | |
| #.# | Other | | | $0.0M | | $0.0M | |
| #.# | Reserves | $61.2M | $536.4M | $597.5M | $78.5M | $676.1M | 15% |

Page 1

**Figure 31: Final mission cost for 10 Spacecraft: $3.44B**

*The cost information contained in this document is of a budgetary and planning nature and is intended for informational purposes only. It does not constitute a commitment on the part of JPL and Caltech.*

In the beginning of our study we thought we could fit ten spacecraft on one launch vehicle. During our configuration study we learned that we can only fit three per vehicle using currently available vehicles. Future developments in the launch vehicle industry might change this, perhaps with the Falcon Heavy. However for our cost analysis we included a single Atlas 5 501 in both the three-spacecraft and ten-spacecraft scenarios. The three-spacecraft scenario indeed fits onto one launch vehicle. If launched today, the ten-spacecraft scenario would require the addition of roughly $.5B, the current approximate cost of 2 Atlas 5 launch vehicles. Rather than include those costs (nearly certain to be inaccurate by the time any such mission would be seriously considered) in the formal estimate, we left the acknowledged, under-estimated cost in the model for the ten-spacecraft scenario.

The 10-spacecraft mission could use combinations of launch vehicles at different times. Further several disruptive launch technologies (e.g. SpaceX "Grasshopper" reusable first stage rockets) are nearing the marketplace making accurate predictions more difficult.

## PHASE 2 ACTIVITIES

Our Phase 2 proposal (developed in parallel with this final report) focuses on three primary technical challenges: demonstrating sail material thermal tolerance for expected perihelion distances, showing that the L'Garde Sunjammer boom design can be scaled up to the size necessary for a 250m x 250m sail, and developing a sail control method suited to such a large structure.

| Material | Boom Design | Control |
|---|---|---|
| Thermo-mechanical testing of available sail materials at MSFC Solar Thermal Test Facility | ~170m design study with team Mechanical Engineer and L'Garde team | Study and simulation of large sail control rates |

We will also further develop the optical communications concept of operations and investigate other subsystem issues.

## FUTURE APPLICATIONS

The architecture we have described can be applied for other missions besides the Heliopause exploration mission that we have explored in detail. Very large solar sails can enable many types of missions to distant objects.

### OORT CLOUD OBJECT / OUTER PLANET / MOON IMPACTOR

An object discovered in 2010, called 2010WG9, is believed to have originated in the Oort cloud[xii]. It is thought to be a pristine sample of early solar system materiel, as its orbit does not come closer to the sun than the distance to Uranus. This object is a scientifically valuable object as its surface has not experienced the periodic melting and freezing that would occur with objects that come closer to the sun.

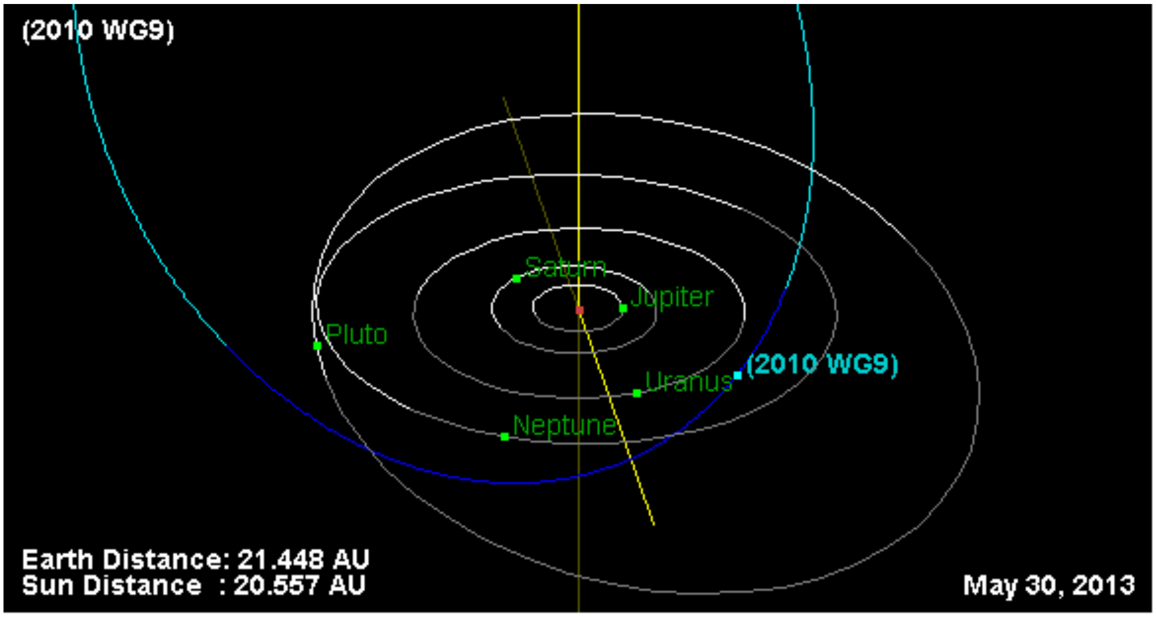



**Figure 32: The object 2010 WG9 does not get any closer to the sun than Uranus, protecting its early solar system materials from the cycle of heating and freezing that happens to objects that get closer to the sun.**

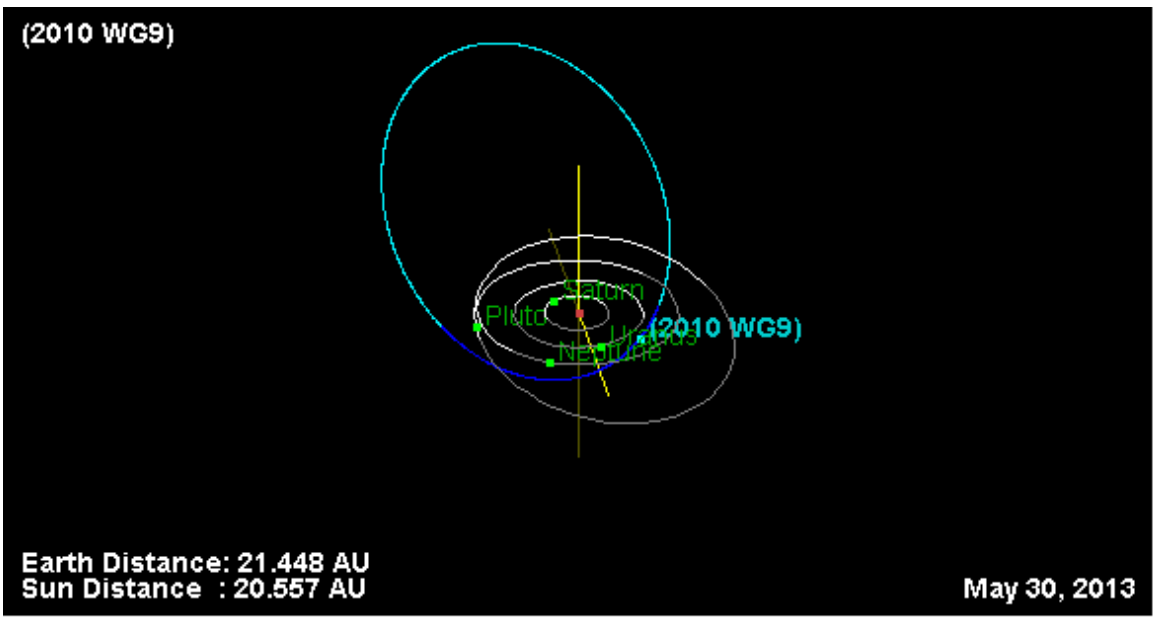



**Figure 33: The object has an unusual orbit and likely contains pristine samples of early solar system material.**

The SSEARS architecture could be used to deliver an impactor to the surface of this object at high relative velocity, with a second spacecraft following behind to study and pass through the impact plume. This architecture provides great flexibility in mission design and timing due to its use of a solar close approach to "aim" the outbound trajectory. This same concept could apply to Enceladus, Europa, or the other outer planet moons, with expected transit times as follows:

| Target Planet (or its moons) | Distance from Sun | Cruise Time |
|---|---|---|
| **Jupiter** | ~5 AU | ~2 years |

| | | |
|---|---|---|
| **Saturn** | ~10 AU | ~3 years |
| **Uranus** | ~20 AU | ~5 years |
| **Neptune** | ~30 AU | ~6 years |
| **Pluto** | ~32 AU for the next several decades | ~7 years |

These figures all include variability due to the different lengths of time possible to reach "maximum velocity" depending on the perihelion distance and whether one exits Earth's orbit with the sail itself or with an upper stage rocket. However, the figures represent a significant improvement over current transit times and results in significantly higher arrival velocities well suited for impactor missions.

## GRAVITATIONAL LENSING (~550 AU)

As early as 1978 it was recognized that a spacecraft at roughly a distance of 550 AU and beyond could take advantage of the sun's gravitational effect to magnify the hydrogen line at 1420 MHz, the so-called ideal frequency for interstellar communications[xiii]. This concept has been developed further by Frank Drake and Claudio Maccone[xiv] as well as a study in 1999 at JPL[xv].

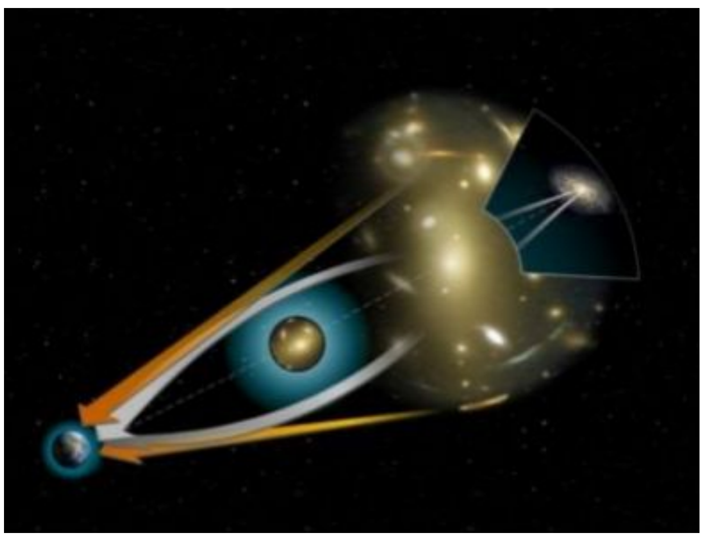


**Figure 34: Gravitational lensing at work. A space probe at 550 AU and beyond could exploit such effects to make detailed studies of other solar systems, among numerous other scientific targets.**

In 1999 JPL produced a rough design which estimated the mass of such a telescope at roughly 1000kg. In order to apply our architecture to this mass we conceived of using *nine* sails as a "Raft" as follows.

## NINE-SAIL "RAFT" FOR GRAVITATIONAL LENSING TELESCOPE DELIVERY

A concept was developed for nine-sails to be arranged in a tiled formation. The sails would not utilize the Sunjammer-type vanes, but the overall system would use the corner sails themselves as vanes, as shown below:

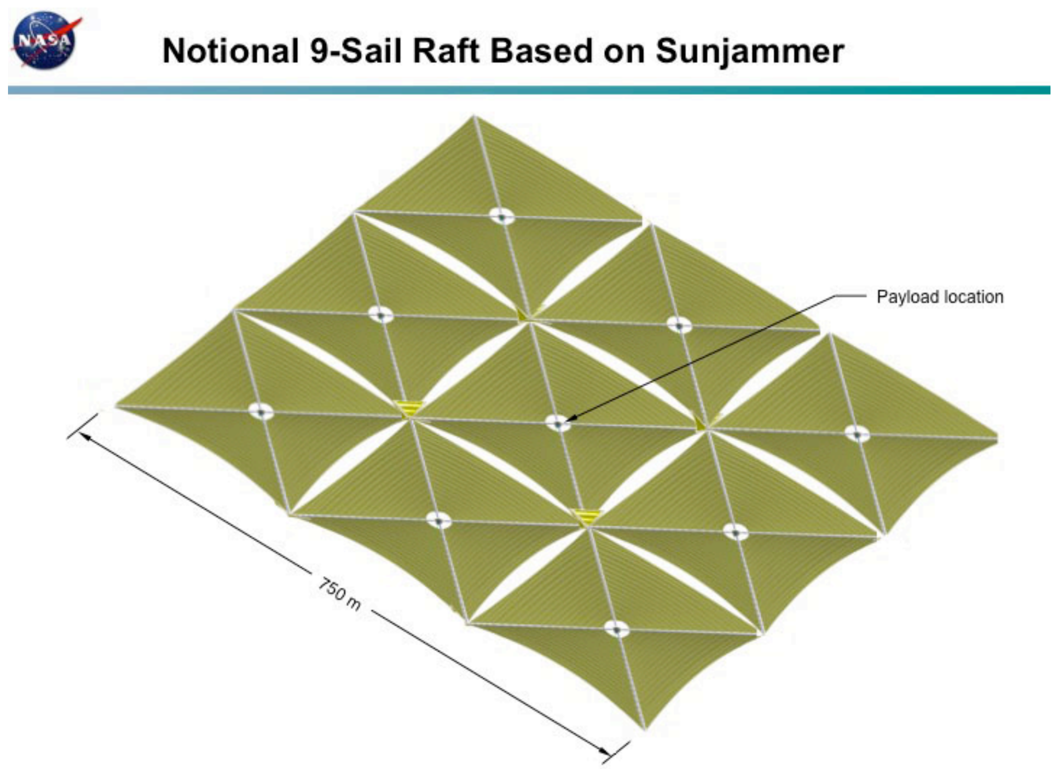



**Figure 35: 9-sail raft design to carry payload range between 500- and 1000kg gravity lens telescope.**

Analysis showed that we can transport 1000kg to 550 au in less than a century. Unfortunately, despite the imaginative appeal of this 9-sail approach, one cannot, with this architecture, increase the area of the sail by a factor of 9 without increasing the *mass* by the factor of 9. This results in essentially the same performance as with just one sail because the payload mass is so small. If the payload was significantly more massive then this "raft" concept would be a useful reduction in transit time over the single-sail version.

So, we conclude that the SSEARS architecture is a reasonable approach to reaching 550AU within a human lifetime, if not during a productive career (at least as such time frames are measured at the beginning of the 21st century.) We leave the feasibility of such biological and medical advances to the imagination.

## THOUSAND ASTRONOMICAL UNIT MISSION (TAU)

JPL developed a mission concept in 1987 that would send a spacecraft to a distance of 1000 AU using then-existing technology. The proposed spacecraft would measure the distance to other stars via stellar parallax, measure conditions in the interstellar medium, and perform tests of general relativity via communications with Earth[xvi]. This spacecraft was proposed to use a 1 MW fission reactor and ion drive to reach 1000 AU in 50 years. Given that much work remains to develop a flight-ready space fission reactor system, a SSEARS-like solar sail architecture bears further studying for this application.

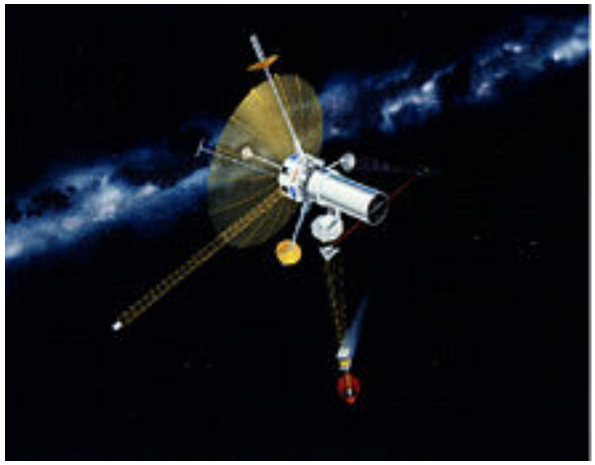


Figure 36: TAU spacecraft concept art

As shown in figures 19 and 20, useful solar sail sizes seem to reach their maximum at about 250x250m, at least with any currently conceivable support technology. Reducing the transit time to 1000 AU from the 120 years it would take the SSEARS architecture will require applying more energy to the sail rather than increasing the size of the sail. This could be achieved with a reflector or beamed energy system. The SSEARS team intends to propose such a study of such a system to a future call.

### OORT CLOUD (~50,000 AU/ 1 LY), EXOPLANET PROBE (<4 LY)

Our SSEARS concept is theoretically limited only by the amount of power/energy that can be applied to the sail and the ability of the material to withstand that energy. As such we can dream of energy transfer systems and materials capable of the performance necessary to reach another solar system in a human lifetime. We hope to explore this space in the far term and believe that NIAC is the ideal program to support such investigations once the "nearer-term" concepts have been explored.

## OUTREACH ACTIVITIES

### CONFERENCES

Due to the Federal Budget "Sequester" it was impossible to travel to many of the relevant conferences.

#### ATTENDED

| | |
|---|---|
| ***NIAC 2012 Fall Conference, Hampton Bays, VA*** | PI Jeffrey Nosanov attended this conference and presented a poster about the project. |
| ***NIAC 2013 Spring Conference, March 2013, Chicago, IL*** | PI Jeffrey Nosanov attended this conference and presented ongoing work and progress made so far. |

| | |
|---|---|
| ***Starship Century Symposium, May 2013, San Diego, CA*** | PI Jeffrey Nosanov attended this symposium and discussed the project with many leaders in the field. |

## ACCEPTED/INVITED BUT UNABLE TO ATTEND DUE TO NASA BUDGET SEQUESTRATION

| | |
|---|---|
| ***International Astronautical Congress 2013, Beijing, China*** | This final report was accepted but travel will not be permitted. |
| ***Starship Congress (Icarus Interstellar)*** | Abstract was accepted but travel will not be permitted. |
| ***100 Year Starship Conference*** | Abstract was accepted but travel will not be permitted. |

## POSTER FOR GENERAL PUBLIC

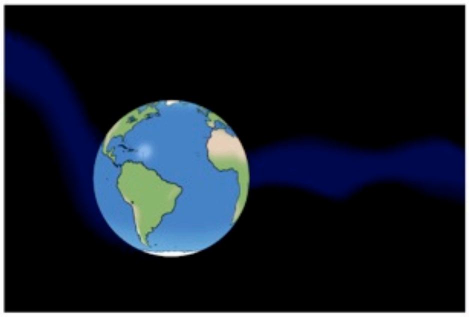
The poster graphic for public consumption will arrive separately in an email to program management.

## REDDIT AMA (ASK ME ANYTHING)

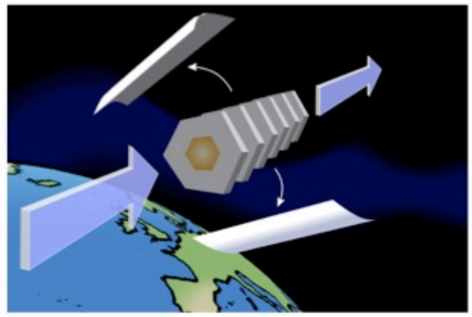
The popular forum website www.reddit.com hosts exchanges between users called "AMAs" This stands for "Ask Me Anything" and many popular or influential figures have hosted an AMA giving their fans and supporters a chance to post questions. Mr. Nosanov hosted an AMA under the title "I am a NASA Innovative Advanced Concepts Fellow, developing a mission concept to the edge of the solar system and nearby interstellar space." The post attracted dozens of viewers, many of whom asked insightful and interesting questions. I answered every single serious question and have concluded that an AMA is a valuable component to a broad outreach effort, but I could have expanded my audience significantly (compared to other AMA events) by advertising or posting about the AMA on several other websites beforehand. The entirety of the exchange (~25 pages of questions and answers) will be sent via email to program management.

## VIDEO

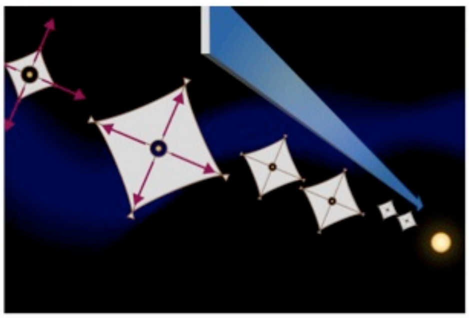
We initiated the development of a brief outreach video to convey the mission concept in a 60-second clip. We developed a storyboard concept for the video and hope to produce the video in Phase 2. The storyboards are reproduced below.



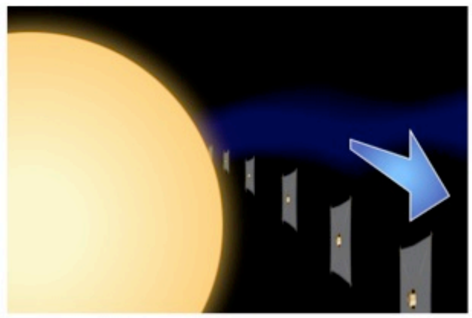
1) Establishing shot – Earth



2) Cut to close up – Payload enters frame left as half-shells separate to reveal 6 stacked spacecraft

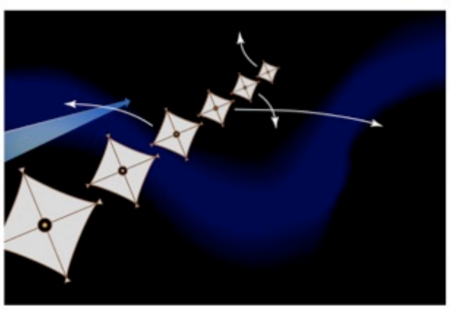


3) Dissolve to – As each spacecraft enters frame left, no solar sail is visible. Solar sails unfurl as each spacecraft travels into frame.

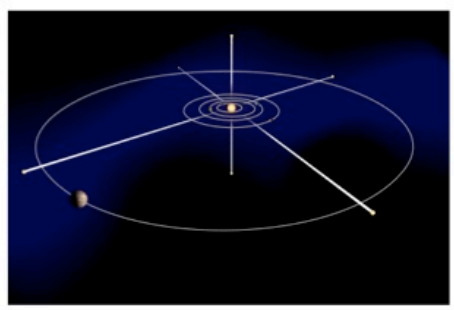


4) Dissolve to – Spacecraft emerge from behind the sun. Camera pans to follow the first craft… [shot continues with next frame]

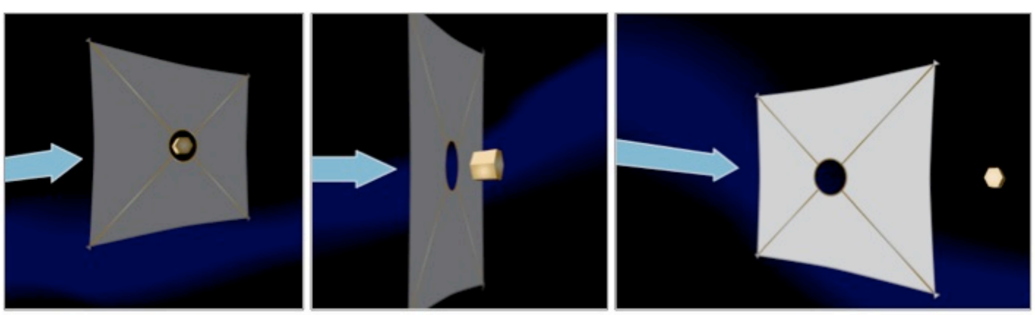
R. Barkus SSEARS Storyboard, draft 0501 1



4) [continued from previous frame] Camera continues pan as spacecraft head into distance. Each of the first four spacecraft then begin a graceful move into their respective trajectory.

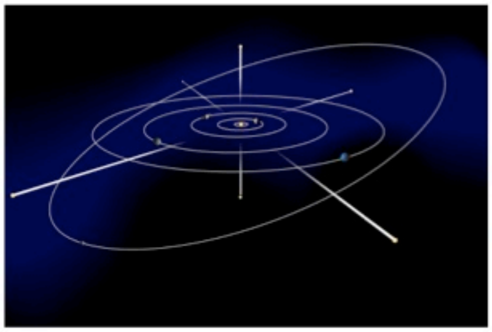


5) Cut to – Animated diagram showing each individual trajectory. The spacecraft, represented by a marker dot and a long fading trail, have just passed Jupiter's orbit.

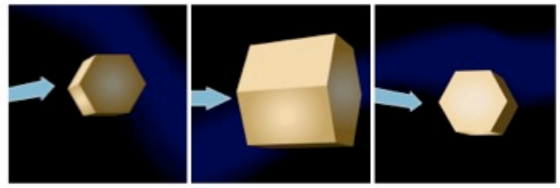


6) Cut to close up – A single spacecraft enters frame left as camera begins follow-pan. Solar sail is released causing a slight speed decrease in the sail and an increase in the spacecraft. Camera pan eases out as spacecraft continues into distance.

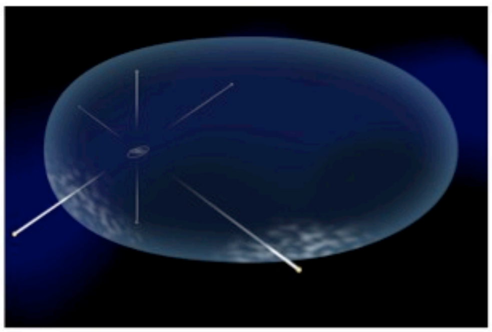
R. Barkus SSEARS Storyboard, draft 0501 2



7) Cut to – Animated diagram showing each individual trajectory. The spacecraft are now beyond Pluto's orbit.



8) Cut to close up – A single spacecraft, now without solar sail, enters frame left as camera begins follow-pan. Camera pan eases out as spacecraft continues into distance.



9) Dissolve to – Animated diagram showing spacecraft markers and long trails (but no visible Heliopause). As the first spacecraft passes the Heliopause boundary a bright wavering texture is illuminated. As the spacecraft travels beyond the boundary, the texture becomes still and somewhat dim. The process is repeated as each spacecraft passes the various boundary points. After all six craft have passed beyond the Heliopause boundary, the complete dimensions are visible.

R. Barkus SSEARS Storyboard, draft 0501 3

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[v] "Pick-Up Ions in the Outer Heliosheath: A Possible Mechanism for the Interstellar Boundary EXplorer Ribbon", J. Heerikhuisen, N. V. Pogorelov, G. P. Zank, G. B. Crew, P. C. Frisch, H. O. Funsten, P. H. Janzen, D. J. McComas, D. B. Reisenfeld, and N. A. Schwadron, 2010 ApJ 708 L126 doi:10.1088/2041-8205/708/2/L126

[vi] http://www.astro.caltech.edu/~tjp/pgplot/, Tim Pearson, California Institute of Technology

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[viii] Preliminary Design of Nuclear Electric Propulsion Missions to the Outer Planets, Chit Hong Yam, T. Troy MicConaghy, K. Joseph Chen, J Longuski
[ix] http://www.nasa.gov/mission_pages/tdm/solarsail/solarsail_overview.html
[x] http://www.mantechmaterials.com/. NeXolve, a ManTech subsidiary, uses FLEXcon to make the films

[xi]

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L'Garde, Inc.

20 March 2013

Jeffrey Nosanov
The Jet Propulsion Laboratory
4800 Oak Grove Dr.
Pasadena, CA 91109

Attention: Jeffrey Nosanov
Subject: ROM Cost Proposal LTP-1226 in Support of JPL's SSEARS Study

Dear Mr. Nosanov

L'Garde is pleased to provide a *Rough Order of Magnitude* (ROM) Cost proposal in support of JPL's SSEARS Study. The fully-burdened costs as a function of deployed solar sail size are summarized below. We used the 1200 $m^2$ Sunjammer Solar Sail Funding/Budget data as a point of departure. The cost numbers in the table below do not include launch costs.

**Table 1**. ROM Cost versus Deployed Solar Sail Size

| Solar Sail Size | Cost |
|---|---|
| 1,200 $m^2$ (34.6m x 34.6m) | $ 12,207,389 |
| 2,500 $m^2$ (50m x 50m) | $ 14,480,674 |
| 10,000 $m^2$ (100m x 100m) | $ 22,423,876 |
| 62,500 $m^2$ (250m x 250m) | $ 53,439,628 |

We are looking forward to working with the Jet Propulsion Laboratory on this exciting project.

I will be the administrative point of contact and Dr. Arthur L. Palisoc will be the technical point of contact. Dr. Palisoc's contact information is

Arthur L. Palisoc
VP Engineering
714-259-0771 ext 228
Email: art_palisoc@lgarde.com

Should you have any questions, please do not hesitate to give me a call at extension 261.

Sincerely,

Larry Beebe
CFO & Contracts Manager
714-259-0771 ext 261
Email: larry_beebe@lgarde.com

LTP-1223 Cost Proposal to JPL in Support of JPL's Asteroid Retrieval Mission Project

[xii] The Peculiar Photometric Properties of 2010 WG9: A Slowly-Rotating Trans-Neptunian Object from the Oort Cloud, David Rabinowitz, Megan E. Schwamb, Elena Hadjiyska, Suzanne Tourtellotte, Patricio Rojo, Astronomical Journal, 2013 Apr 20
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