Phase I Final Report
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A Realistic Interstellar Explorer

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1.0 Introduction

For more than 20 years, an "Interstellar Precursor Mission" has been discussed as a high priority for our understanding (1) the interstellar medium and its implications for the origin and evolution of matter in the Galaxy, (2) the structure of the heliosphere and its interaction with the interstellar environment, and (3) fundamental astrophysical processes that can be sampled *in situ*. The chief difficulty with actually carrying out such a mission is the need for reaching significant penetration into the interstellar medium (~1000 Astronomical Units (AU[1])) within the working lifetime of the initiators (<50 years). During the last two years there has been renewed interest in actually sending a probe to another star system - a "grand challenge" for NASA - and the idea of a precursor mission has been renewed as a beginning step in a roadmap to achieve this goal.

Ongoing studies now being carried out by the Jet Propulsion Laboratory (JPL) in conjunction with a NASA-selected Interstellar Probe Science and Technology Definition Team (IPSTDT) are focused on the use of solar sails to achieve a more modest goal of ~400 AU in 20 years with a requirement of reaching at least 200 AU. This approach, as well as all of those previously considered, obtains significant solar-system escape speeds by "dropping" the probe into the Sun and then executing a ΔV maneuver at perihelion. In the solar sail approach, the sail is used first to remove the angular speed of the Earth and probe about the Sun; the sail is then maneuvered face-on into the Sun at 0.1 to 0.25 AU (set by sail thermal heating constraints), and solar radiation pressure accelerates the probe away from the Sun until the sail is jettisoned (at ~5 AU in the ongoing studies). The previous approaches, and that adopted for study here, initially send the probe out to Jupiter, using that planet's gravity to remove the probe angular momentum. The probe is then allowed to fall much closer to the Sun - in this case to 4 solar radii (R_S[1]) - where a large propulsive ΔV maneuver is required over a very short time - taken here to be ~15 minutes to minimize gravity losses. At this time all acceleration is over and the probe continues on a high-energy ballistic escape trajectory from the solar system.

In the scenario studied in our Phase I work, the use of a carbon-carbon thermal shield and an exotic propulsion system capable of delivering high specific impulse (I_{sp}) at high thrust is enabling. For the NASA/JPL case, the manufacture, deployment, and management of long-term low-thrust maneuvering with a solar sail is the enabling propulsive technology. Both implementations require low-mass, highly integrated spacecraft in order to make use of moderate (Delta-class) expendable launch vehicles (ELVs). The further from the Sun that operation is required, the more stressing are the reliability requirements for the spacecraft itself. Our Phase I concept is baselined to operate at least 2.5 times as far from the Sun as the goal for the JPL study (1000 AU versus 400 AU) and contains system-architecture elements required as next-roadmap-step after the sail mission - our probe design is largely independent of the propulsion means, whether those studied previously by us, solar sails, or others. In the Phase I work we have completed an initial scoping of the system requirements and have identified system drivers for actually implementing such a mission.

In this Phase I study we began the task of looking seriously at the spacecraft mechanical, propulsion, and thermal constraints involved in escaping the solar system at high speed by executing a ΔV maneuver close to the Sun. In particular, to fully realize the potential of this scenario, the required ΔV maneuver of ~10 to 15 km/s in the thermal environment of ~4 R_S (from the center of the Sun) remains challenging. Two possible techniques for achieving high thrust levels near the Sun are: (1) using solar heating of gas propellant, and (2) using a scaled-down

Orion (nuclear external combustion) approach. We investigated architectures that, combined with miniaturized avionics and miniaturized instruments, enable such a mission to be launched on a vehicle with characteristics not exceeding those of a Delta III. This systems approach for such an Interstellar Explorer has not been previously used to address all of these relevant engineering questions but is required to lead to (1) a probe concept that can be implemented following a successful Solar Probe mission (concluding around 2010), and (2) system components and approaches for autonomous operation of other deep-space probes within the solar system during the 2010 to 2050 time frame.

1.0.1 Advanced Concept Description

Two highly-ranked exploratory missions central to the discipline of Space Physics have eluded the grasp of NASA and the Space Science community for over 20 years. The first, Solar Probe (SP), now slated for a 2007 launch, would explore the inner boundary of our stellar environment; the corona of the Sun. The second, the Interstellar Probe (IP), would probe the outer boundary: the Very Local Interstellar Medium (VLISM).

On a longer timescale there is the question of actually crossing the interstellar void itself. In this regard, IP is a precursor mission, i.e., it can answer fundamental science questions, assay our local neighborhood for conditions through which a starship must travel, and, most significantly, act as a testbed for technologies that must be developed on our way to reaching the stars.

The problem is, of course, distance. It is a long way to the stars and compressing the time for the journey into the scale of a human lifetime requires high speeds, and these, in turn, require very energetic means of propulsion [e.g., *Forward*, 1996]. The termination shock of the solar wind likely extends to ~100 AU while the loosely bound comets in the Oort cloud are thought to populate a spherical region out to ~1 LY, so an Interstellar Probe must be able to make some inroad into this distance, even for a precursor mission.

The basic equation that governs spaceflight is the rocket equation. Relativistic corrections were derived over 50 years ago [*Ackeret*, 1946; 1947], and these equations were soon applied to the idea of interstellar travel. Acceleration of 1 Earth gravity (1g) for ~1 year suffices to bring the vehicle to near light speed, producing significant time dilation. To attain high speeds efficiently, exhaust speeds also need to be near light speed; the large speeds required still translate into large mass ratios, requiring in turn, large mass flow rates. These constraints, in turn, translate into the need for fusion or photon rockets [*Shepherd*, 1952; *von Hoerner*, 1962; *Sänger*, 1961-2; *Sagan*, 1963; *Forward*, 1996].

To mitigate the phenomenal engineering challenges, suggestions have been made for locating the source of energy elsewhere, e.g., for beaming power to the vehicle from the solar system, or fusing the material in the interstellar medium itself [*Bussard*, 1960; *Forward*, 1996, and references therein]. While some wishful thinking has looked to "post-quantum" physics for answers (wormholes, reactionless drives, faster-than-light travel), today there is no evidence to suggest that such schemes are more than wishful thinking.

If we are to take seriously the notion of interstellar travel toward the middle of the next century, *and try to plan to make it happen*, then the best we can do is to rely on currently known physics, which can still push the limits of the doable even under the most optimistic of technological advances. A significant penetration of the VLISM would be ~10,000 AU in 50 years. This requirement translates into a speed requirement of 200 AU/yr ³ 940 km-s^{-1} (exceeding the escape speed from the Milky Way! [*Allen*, 1973]). An "optimized" rocket is characterized by a mass ratio of exp(1.59) = 4.90 [*von Hoerner*, 1962], so such a rocket reaching 200 AU/yr requires a specific impulse ~60,000 s. For a fusion rocket burning deuterium (D) and ^{3}He (so that the exhaust can be directed by magnetic fields, e.g., *Martin and Bond* [1978]), the converted mass fraction is 0.39%, and the effective exhaust speed is 26,450 km-s^{-1}, corresponding to I_{sp} ~ 2.7x10^6 s. Worked the other way, such a D-^{3}He rocket could attain 940 km-s^{-1} with a mass ratio of 1.036.

Physically, such a system may be realizable, but given the current status of fusion research, there is no way to package such a propulsive system into several hundred kilograms or less, and a ground-based test bed of a Daedalus-type system [*Bond et al*., 1978] is decades away. Hence, we believe that *realistically such a system is beyond the time frame of 10 to 40 years in the future, the time frame for this study*. However, probes that can attain one-tenth this speed (and still attain the same distance by traveling ten times as long) are, we believe, within grasp during this time frame if the appropriate studies and thrusts are begun now.

In this Phase I work we examined carefully the mission requirements for an Interstellar Probe to (1) enable serious consideration of actually launching this mission before the middle of the next century, and (2) minimize the required spacecraft mass and power, consistent with the required science measurements (the "sciencecraft" paradigm), in order to minimize the launch vehicle requirements and, therefore, a major cost driver. The design will have multiple applicabilities to outer solar system missions where mass will be at a premium, e.g., sample return missions from icy satellites or a Kuiper-belt object.

We have not discussed instrument packages in detail. These have been discussed and debated many times over; however, the proposed architecture is in consonance with current thinking and requirements, and recently NASA's Interstellar Probe Science and Technology Definition Team (IPSTDT) has been examining the implementation of a more modest mission by making use of solar sail propulsion.

The work described here identifies a mission concept that can link the required science, desired instruments, spacecraft engineering and the realities of the fiscal and technological milieu in which NASA must operate if such missions are to reach fruition and lay the groundwork for eventually realizing true starships.

A "technically sweet" solution is to use the interstellar medium itself for fuel (the Bussard-ramjet approach, which still has problems with the magnetic intake [*Forward*, 1996]). The Interstellar Probe proposed here would assay the content and clumpiness of the Very Local Interstellar Medium (VLISM), assessing the possible implementation of a ramjet approach while pioneering technologies required for long-term autonomous cruise. Such a precursor mission could allow for the launch of such an advanced probe toward Epsilon Eridani or other stars late in the 21st century. At 200 AU/yr, such a second generation probe could make the first targeted interstellar crossing in ~3500 years, the approximate duration of the Egyptian Empire A more robust propulsion system that enabled a similar trajectory toward higher declination stars such as Alpha Centauri could make the corresponding shorter crossing in a correspondingly shorter time of ~1400 years, the time that some buildings have been maintained, e.g., Hagia Sophia in Constantinople and the Pantheon in Rome. Though far from ideal, the stars would be within our reach.

1.0.2 Science Rationale

Travel to the stars is the stuff that dreams and science fiction novels are made of. However, there is also a very scientifically compelling and serious side of the concept as well. A mission past the boundary of the heliosphere to only a fraction of a light year would yield a rich scientific harvest [*Jaffe and Ivie,* 1979; *Holzer et al.,* 1990].

The "fleet" of four interstellar spacecraft, Pioneer 10 and 11 and Voyager 1 and 2, all have speeds in excess of the escape speed from the Sun and will penetrate into interstellar space. Powered by Radioisotope Thermoelectric Generators (RTGs), the spacecraft all have a finite lifetime due to the half life of the Pu-238 fuel (89 years) as well as degradation of the Si-Ge convertors in the RTGs. Pioneer 10 and 11 have effectively been "switched off" as they have run out of sufficient power for sustained operation. The Voyagers now form the Voyager Interstellar Mission with the goal of at least penetrating the termination shock of the solar wind, which is thought to be located ~100 AU from the Sun.

Whether the Voyagers will actually reach the "undisturbed" interstellar medium prior to falling silent as well remains unknown today. What is clear is that there are fundamental sciencequestions that can only be addressed by instrumentation that actually penetrates outside of the heliosphere [*Holzer* et al., 1990; *Mewaldt et al.,* 1995]. The science goals include:

Explore the nature of the interstellar medium and its implications for the origin and evolution of matter in the Galaxy. We know amazingly little about the nature of the Very Local Interstellar Medium (VLISM). For example, measurements of rotation measures and dispersion measures of pulsars [*Rand and Lyne*, 1994] suggest a large scale magnetic field of ~1.4 μG, but we have no idea of the structure or its variations within 0.1 or even 0.01 LY. Similarly, the properties of the nonthermal portion of the medium (including the low-energy galactic cosmic rays) remain unknown.

Explore the structure of the heliosphere and its interaction with the interstellar medium. The Voyager Interstellar Mission may characterize the distance to the termination shock, but a farther-ranging probe is required to understand the dynamics of the interaction and how they are influenced by the conditions in the VLISM.

Explore fundamental astrophysical processes occurring in the heliosphere and the interstellar medium. Shock acceleration of particles has profound impacts upon many sub-branches of astrophysics. In addition, the structure of the solar wind interface with the VLISM has analogs in many other astrophysical settings.

Determine fundamental properties of the universe. Measurements of ^{3}He, D, and ^{7}Li would give constraints on big-bang nucleosynthesis and on how these key indicators have been processed in the interstellar medium [*Schramm and Turner*, 1998]. Also, by including an extremely accurate clock and a γ-ray detector, an extremely good baseline could be established for constraining the location of gamma-ray bursts (GRBs) in the plane of the sky. Extremely accurate tracking of the IP can be used to look for gravitational waves and a non-zero cosmological constant. Polarization measurements of the downlink carrier can be used to look for inherent anisotropies in the structure of space [*Nodland and Ralston*, 1997]. If real, these could be related to new physics that might lead to faster travel times (fulfillment of the "wishful thinking" mode of interstellar transport).

These science goals are all in accord with formulations being pursued by the Sun-Earth Connections Roadmap committee as part of the current NASA strategic planning exercise.

1.0.3 Relevance to Office of Space Science Programs

The idea of a dedicated interstellar "precursor" mission first surfaced at the conference "Missions Beyond the Solar System" held at the Jet Propulsion Laboratory (JPL) in August 1976. The baseline mission was to reach 370 AU in 20 years after launch and 1030 AU in 50 years after launch using a fission-based Nuclear Electric Propulsion (NEP) system [*Jaffe and Norton*, 1980*; Jones and Sauer, 1984]*. Key measurements were *in situ* measurements of the solar wind, interaction region and the interstellar medium with the goal of characterizing near interstellar space [*Jaffe and Ivie*, 1979].

In March of 1990, the mission concept was revived [*Holzer et al*., 1990], as one of three "frontier probes," to explore the global heliosphere and local interstellar space. Again the prime focus was *in situ* fields and particles measurements. The goal was scaled back to reaching ~200 AU within ~25 years with 13 science instruments. A "powered solar flyby" was advocated as a means of accomplishing a rapid escape from the solar system without requiring NEP. The 1990 Workshop concept emphasis is in contrast with the TAU (Thousand AU) probe using a 1 MW NEP system to reach 1000 AU in ~50 yrs (terminal speed of 105 km-s^{-1}) [*Forward*, 1996].

In the *Space Physics Strategy-Implementation Study* [1991] a "fields and particles" Interstellar Probe mission was endorsed and called out for a launch shortly after 2010 "to reach a minimum distance of 200 AU within 25 years. The main enabling technology problem was viewed as the wedding of a large ÐV maneuver near the Sun (~4 R_S - the distance proposed for a near-solar probe) or otherwise providing the high velocity required. The mission has also been advocated by both (1) the solar and space physics panel and (2) the astronomy and astrophysics panel of the NAS/NRC study: *Space Physics in the 21st Century - Imperatives for the Decades 1995 to 2015*, as well as previous NAS/NRC reports.

Most recently, in early 1999, NASA solicited membership in a committee to examine interstellar probe requirements. This IPSTDT has had three meetings discussing the definition of a science payload and its incorporation into a mission based upon a solar sail. In this case the solar sail is the enabling technology but gives rise to similar instrumentation, power, and spacecraft requirements that are described here; results of the our work in the areas of spacecraft architecture, communication, and instrumentation accommodation are also applicable to this other mission concept.

In our Phase I work we have (1) married the perihelion burn concept with advanced propulsion concepts and studied packaging these in a realistic system architecture, (2) considered the science and engineering data that must be returned and used the requirements to design a telecommunications system that fits within spacecraft resources, (3) assembled a system design, subsystem by subsystem that can be launched on a Delta-III- class launch vehicle, and (4) investigated requirements for autonomous operations, reprogramming, and repairs. Our Phase I results have come a long way toward assembling such a self-consistent systems design.

To have a design that might fit the launch resources, we have set the probe mass to 50 kg; this has in turn decreased the payload mass (and power) and led us to an architecture that uses common ultra-low power (ULP) processor modules. By relying on these, significant effort must be expended to move traditional hardware functions into software to conserve mass and support an overall self-healing architecture. Although these cuts are drastic with respect to the initial design, they indicate the direction that the architecture and implementing technologies are driven in order to implement this mission in this fashion.

1.1 Mission Overview

An interstellar precursor will only penetrate into the nearby interstellar medium. However, there are a variety of science goals to be accomplished, including taking the measure of the size of the heliosphere and investigating the properties of the undisturbed medium itself. The first constraint suggests that a mission should be sent near the "nose" of the heliosphere. This region, centered on the stagnation line in the flow of the ionized component of interstellar material as it interacts with the solar wind, is the closest that the undisturbed interstellar medium comes to the Sun. At the same time, there is general interest in exploring the properties of the medium, e.g., its clumpiness, in the direction of nearby stars. By choosing a direction toward a nearby star, the probe can report on conditions along its line of travel that is also in the direction along which integrated measurements can be made by employing the star's spectrum. For example, the Ly-a line profile yields information on both the hydrogen and deuterium column abundances between the Sun and target star [*Frisch*, 1993*; Linsky and Wood*, 1996]. Targeting nearer stars can thus yield important information on the near (~few parsecs) interstellar medium. As the ultimate goal of the probe roadmap is a nearby Sun-like star, a K, G, or F star nearby is a good candidate. Finally, orbital mechanics constraints favor targets near the plane of the ecliptic (as discussed below). Taken together, these constraints have led us to baseline a mission toward Epsilon Eridani (HD22049).

Epsilon Eridani is a K2V dwarf main sequence star 10.7 light years from Earth. The star was used by Drake in his Project Ozma search for intelligent radio emissions (Tau Ceti was also observed during the same time period in late 1960 [*Sagan and Shklovskii*, 1966]) and has been debated as a possible site for an Earth-like planet [*Lawton and Wright*, 1990; *Dole*, 1964]. Epsilon Eridani is nominally a single star but has a dust cloud surrounding it at a mean radius of ~23 AU [*Lawton and Wright*, 1990]. The star also appears to have a hot corona similar to that of the Sun [*Schmitt et al.,* 1996; *Laming et al.*, 1996]. It has an apparent rotation period of 11.1 days and may exhibit a magnetic activity cycle of ~5 years similar to the 11-year solar cycle; such variations are less pronounced in the other Sun-similar near stars τ Cet and σ Dra [*Gray and Baliunas*, 1995].

1.1.1 Mission Requirements

For this work, we adopt the use of a near-Sun ΔV maneuver to attain a high-speed escape from the solar system. This requires a Jupiter flyby to eliminate the heliocentric angular momentum of the probe. Increased performance for a given launch vehicle is possible by using an Earth flyby, but elimination of any Earth flybys is desirable for cutting down flight time with intensive operations, eliminating a deep-space burn, and eliminating the return of nuclear materials to the close proximity of Earth. It must be kept in mind that part of the increased throw mass must be used up in providing for a required deep-space maneuver, required to line up the probe for the Earth gravity assist flyby.

For the baseline mission, we required that the probe should be launched with a Delta III-class launch vehicle and approach within four solar radii of the Sun. With no other constraints, energy and momentum conservation set the required injection energy (C_3) to effect the near-Sun passage. The need to line up the outgoing asymptote of the trajectory toward the target direction while having Jupiter in the correct position to effect the gravity assist means that the required planetary alignment occurs roughly once every 13 years. The details of Jupiter's orbit cause the launch energy requirements to change each launch window and return roughly to an optimum roughly once every 80 years for a launch toward ε Eridani.

The idea of using a high-I_{sp}, high-thrust maneuver close to the Sun was first identified in 1929 by rocket pioneer Hermann Oberth [*Ehricke*, 1972]. Measuring all speeds in km/s the asymptotic escape speed from the solar system is approximately

$$v_{escape} = (\Delta V)^{1/2} \frac{35.147}{r_p^{1/4}}$$

This equation gives an approximation to the required propulsive maneuver for a given asymptotic speed. Note that for $r_p = 4\ R_S$, the "crossover" point is roughly DV = v_{escape} ~ 620 km-s^{-1} = 130 AU/yr. In other words, for escapes speeds below 130 AU/yr a perihelion maneuver offers an advantage, providing that it can be implemented technically. Above this speed, there is no major advantage, and a direct departure outward from Earth's orbit offers the simplest implementation.

As a baseline target we have chosen ΔV = 15 km-s^{-1} for the propulsion "target" to enable an asymptotic solar system escape speed of roughly 20 AU/yr. A 50-yr flight time will reach 1000 AU, a distance well outside of the projected influence of the heliosphere/interstellar medium interaction region. A probe lifetime of 500 years will reach 10,000 AU, a significant penetration into local interstellar space.

Candidate targets for solar system escape direction include the “nose” of our Sun’s helio-sphere, with J2000 heliocentric ecliptic right ascension of 74.9° and declination of -7.8°, as well as nearby stars [*Allen*, 1973] with J2000 heliocentric declination within 33° of the ecliptic plane that are spectral types K, G, or F (near-Sun-like); Table 1.1.1 lists likely candidates for harboring Earth-like planets. The nearest stars [Section 114 of *Allen*, 1973] with Earth Mean Equator 1950 declinations less than ±32.7° + 23.5° (Earth obliquity) comprised the initial candidate list. This list decreased after an Earth Mean Equator of 1950-to-Earth Mean Orbit (ecliptic) of 2000 coordinate transformation indicated several stars with ecliptic declinations beyond ±32.7°. The initial screening of candidate stars (see Table 1.1.2) eliminated the potential of targeting well-known stars such as Alpha Centauri and Sirius, due to ecliptic declinations of -58.23° and -39.59°, respectively. The 32.7° limit comes from the turn angle (equation (1.1.1)) of an impulsive maneuver at a 4-solar radii perihelion (r_p) that propels the spacecraft to a 20 AU/year solar system escape velocity ($V_{.}$). This limiting case requires a 90°-inclination Jupiter-to-perihelion transfer, and assumes perihelion lies in the ecliptic plane.

$$\rho = \cos^{-1}(1/(1 + V^{2} r_p/\mu_{Sun})) \quad \text{where } \mu_{Sun} = 2.9591221\text{x}10^{-04}\ \text{AU}^3/\text{day}^2 \quad (1.1.1)$$

The ΔV required at perihelion (equation (1.1.2)) is the difference between the perihelion velocities of the hyperbolic solar system escape and elliptic Jupiter-to-perihelion transfer trajectories. At a typical Jupiter-Sun range of 5 AU (approximate aphelion distance for the Jupiter-to-perihelion transfer), a perihelion ΔV of 14.801 km/sec in the spacecraft velocity direction will deliver the spacecraft to solar system escape conditions of 20 AU/year and ≤ 32.7°

declination with respect to the ecliptic.

(1.1.2) $\Delta V = (V^{2} + 2\mu_{Sun}/ r_p)^{0.5} - (2\mu_{Sun}\, r_a/(r_p(r_a + r_p)))^{0.5}$

where r_a and r_p are Jupiter aphelion and perihelion range.

Star	Distance (Light Years)	Spectral class	Habitable planet probability
Alpha Centauri A	4.3	G4	0.054
Alpha Centauri B	4.3	K1	0.057
Epsilon Eridani	10.8	K2	0.033
Tau Ceti	11.8	G8	0.036
70 Ophiuchi A	16.4	K1	0.057
Eta Cassiopeiae A	18.0	K9	0.057
Sigma Draconis	18.2	G9	0.036
36 Ophiuchi A	18.2	K2	0.023
36 Ophiuchi B	18.2	K1	0.020
Delta Pavonis	19.2	G7	0.057
82 Eridani	20.9	G5	0.057
Beta Hydri	21.3	G1	0.037
Data is from Dole [1964]			
Designations A and B refer to components in multiple (bound) star systems; the effect of multiple systems on the formation of stable planetary orbits remains unknown			

Table 1.1.1 Candidate Stars for Launch Direction

Star	Spectral Class	α Right Ascension (deg)	δ Declination (deg)
Epsilon Eridani	K2	48.29	-27.76
Tau Ceti	G8	17.66	-24.77
Wolf 28	DG	13.28	0.19
Procyon	F5	115.87	-16.00
191408	K3	297.06	-15.68
131977 A	K5	228.26	-4.31
36 Ophiuchi	K1	259.96	-3.54

Table 1.1.2 Stars Within Declination Limits

1.1.2 Designing Fast, Low-Cost Trajectories to Four Solar Radii.

The mission requires a fast transfer to Jupiter, where an unpowered gravity assist reduces aphelion and perihelion. In addition, the gravity assist accomplishes a plane change that places the spacecraft on a trajectory where a perihelion ΔV will yield the desired solar system escape direction. A NASA-sponsored Interstellar Probe study in 1991 included teams at Jet Propulsion Laboratory and Science Applications International Corporation. This study's "200 AU from the Sun in 25 years" requirement for ballistic trajectories enabled use of Venus and Earth gravity assists for lowering launch energy. Insertion of a 4.2-year segment with one Venus and two Earth gravity assists prior to the required Jupiter flyby would require a 50+% increase in perihelion ΔV from 14.8 to 22.5 km/s to increase solar system escape velocity from 20 to 25 AU/year.

To generate initial conditions for designing a minimum-ΔV ballistic trajectory to a nearby star, a method was formulated for identifying the required location of Jupiter and the corresponding Jupiter-to-perihelion transfer inclination. Applying the law of sines from spherical trigonometry (equation (1.1.3)) leads to the solution for the $i_{J\text{-}S}$, Jupiter-to-4 solar radii perihelion transfer inclination (equation (1.1.4)).

(1.1.3) $\sin(i_{J\text{-}S})/\sin(\delta) = \sin(90°)/\sin(\rho)$

where δ is target star declination - Table 1.1.2, ρ is the perihelion turn angle - eqn. (1.1.1)

(1.1.4) $$i_{J\text{-}S} = \sin^{-1}(\sin(\delta)/\sin(\rho))$$

The corresponding heliocentric longitude of Jupiter, θ_{JUP}, in J2000 ecliptic coordinates is also solved with a spherical trigonometric relationship in equation (1.1.5).

(1.1.5) $$\theta_{JUP} = \alpha \pm \sin^{-1}(\tan(\delta)/\tan(i_{J\text{-}S})), \text{ “+” for } i_{J\text{-}S} \leq 90°, \text{ “-” for } i_{J\text{-}S} > 90°$$

where α is the target star right ascension - see Table 1.1.2.

For a given target star, solar system escape velocity (e.g., V = 20 AU/year), posigrade and retrograde Jupiter-to-Sun transfers ($i_{J\text{-}S} < 90°$ and $i_{J\text{-}S} > 90°$) and the corresponding heliocentric longitude of Jupiter, two Jupiter encounter dates are determined per Jupiter orbit over the range of allowable Jupiter flybys (e.g., 2012-2042). The iterative solution procedure begins with finding the Earth launch, Jupiter flyby, and subsequent perihelion dates that provide minimum launch energy, an unpowered Jupiter flyby, acceptable Jupiter flyby altitude, and minimum movement of the Jupiter flyby date from the initial θ_{JUP}-calculated date. Upon determination of this initial solution, the converged trajectory's heliocentric longitude of Jupiter, θ_{JUP} is attained from equation (1.1.5) by perturbing the propulsive turn angle at perihelion (ρ in eqn. (1.1.1)). If an increase in ρ yielded the updated θ_{JUP}, the solar system escape velocity, V , increases. A decrease in ρ invalidated the solution, because V decreases below the 20 AU/year science constraint. The new value of ρ yields a new value from equation (4) for $\iota_{J\text{-}S}$, which is used to produce a new converged Earth-Jupiter-perihelion trajectory. The revised Jupiter flyby date produces a new θ_{JUP}, and the iteration process continues until established convergence criteria (e.g., Jupiter flyby date changes less than 0.1 days from the previous iteration) are met.

1.1.3 Designing Trajectories from Perihelion to the Stars

The ΔV from equation (1.1.2) is updated using the final V and r_a (approximated by Jupiter's distance to the Sun at Jupiter flyby). This ΔV is applied in the spacecraft velocity direction to the perihelion spacecraft state vector. Propagation of the resulting solar system escape trajectory to 1000 AU delivers the spacecraft to the target star direction. An angle offset of less than the perihelion latitude in declination and still less in right ascension from the target star direction occurs with this method. Most of the offset is explained by the off-normal-to-the-ecliptic components of the perihelion ΔV. Precision targeting of target star was then achieved by lifting the "velocity direction only" constraint at perihelion. The resulting ΔV penalty is then assessed in magnitude and spacecraft orientation offset from thrusting in the velocity direction. The affects of finite burn and maximum permissible thrust acceleration have not been considered at this point.

1.1.4 Designing Reference Trajectories to 36 Ophiuchi and Epsilon Eridani

A 2019 launch to 36 Ophiuchi represents the lowest launch energy (C_3 = 100.6 km^2/s^2) case for a ~20 AU/year trajectory to a star listed in Table 1.1.1. For this launch energy maximum payloads are 425 kg for a 3-stage Delta III (Star 48 upper stage) and 500 kg for an Atlas IIAS/Star 48. This trajectory passes within 799,000 km of Jupiter at 21.519 km/s (12.078 km/s excess velocity) with a 27.0° approach phase angle. The Jupiter flyby lowers perihelion from 0.99 AU to 0.02 AU, lowers aphelion from 8.74 AU to 5.16 AU, and increases orbital inclination in order to provide proper orientation for the perihelion burn. Two years later, a 15.958 km/s perihelion ΔV within 8.8° of the spacecraft velocity direction directs the spacecraft towards 36 Ophiuchi at 20.7 AU/year.

A 2011 launch to Epsilon Eridani represents a low launch energy (C_3 = 117.1 km^2/s^2) case to a star situated near (within 32.1°) the direction from the Sun to the heliosphere "nose." For this launch energy maximum payloads are 256 kg for a 3-stage Delta III and 342 kg for an Atlas IIAS/Star 48. This trajectory, summarized in Table 1.1.3 and depicted in Figure 1.1.1, passes within 653,300 km of Jupiter at 23.527 km/s (12.868 km/s excess velocity) with a 24.7° approach phase angle. The Jupiter flyby lowers perihelion from 1.02 AU to 0.02 AU, lowers aphelion from 11.88 AU to 5.05 AU, and increases orbital inclination in order to provide proper orientation for the perihelion burn. Two years later, a 15.393 km/s perihelion ΔV 12.5° from the spacecraft velocity direction sends the spacecraft toward Epsilon Eridani at 20.2 AU/year.

Event	Date	Sun distance (AU)	Inclination (deg)	Ecliptic orientation	
				Lat	Lon
Towadd 36 Ophiuchi					
Launch	2019 Feb 18.908	0.9884	DLA = -22.0	-0.001	149.686
Jupiter flyby	2020 Jul 31.956	5.1506	0.5	-0.294	293.529
Perihelion	2022 Jul 31.758	0.0186	6.4	0.318	113.315
1000 AU	2070 Nov 9.546	1000.0000	3.5	-3.540	259.960
Toward ε Eridani					
Launch	2011 Jul 12.180	1.0166	DLA = 3.9	0.001	289.350
Jupiter flyby	2012 Oct 26.056	5.0443	1.1	-0.705	67.721
Perihelion	2014 Nov 28.024	0.0186	57.7	0.538	247.827
1000 AU	2064 May 31.509	1000.0000	27.76	-27.760	48.290

Table 1.1.3 Key Events for Sample Missions

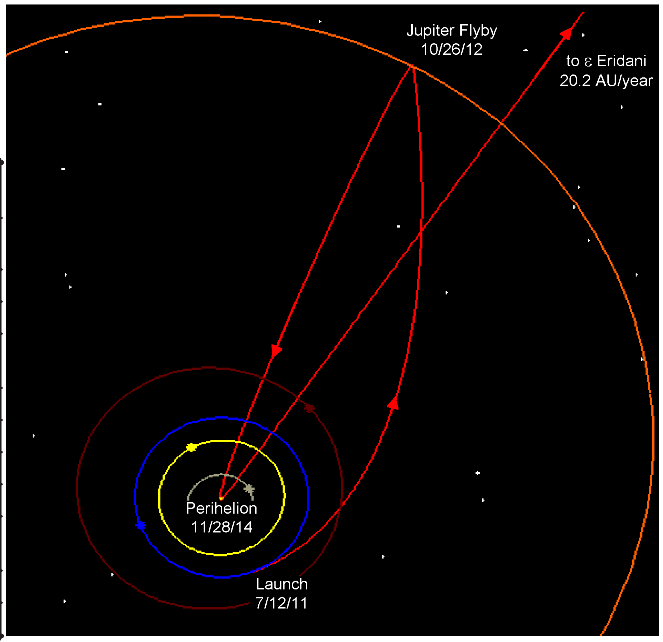


Figure 1.1.1 Ecliptic Plane Projection of Instellar Probe Trajectory toward e Eridani

1.1.4 Indirect Launch Mode

The ε Eridani trajectory is of some interest for reasons already mentioned. However, the relatively large C3 and use of the Delta III severely constrains the available payload mass. To increase the available throw weight, we consider the use of the indirect launch mode [*Farquhar and Dunham*, 1999].

To provide more throw weight and flexibility in the launch window, an additional stage can be incorporated into the spacecraft. Typically, the spent stage is retained to minimize the complexity of the structural design, but with the additional thermal and propulsion constraints that the probe faces during the perihelion maneuver, provision needs to be made to jettison this upper, upper stage subsequent to its firing.

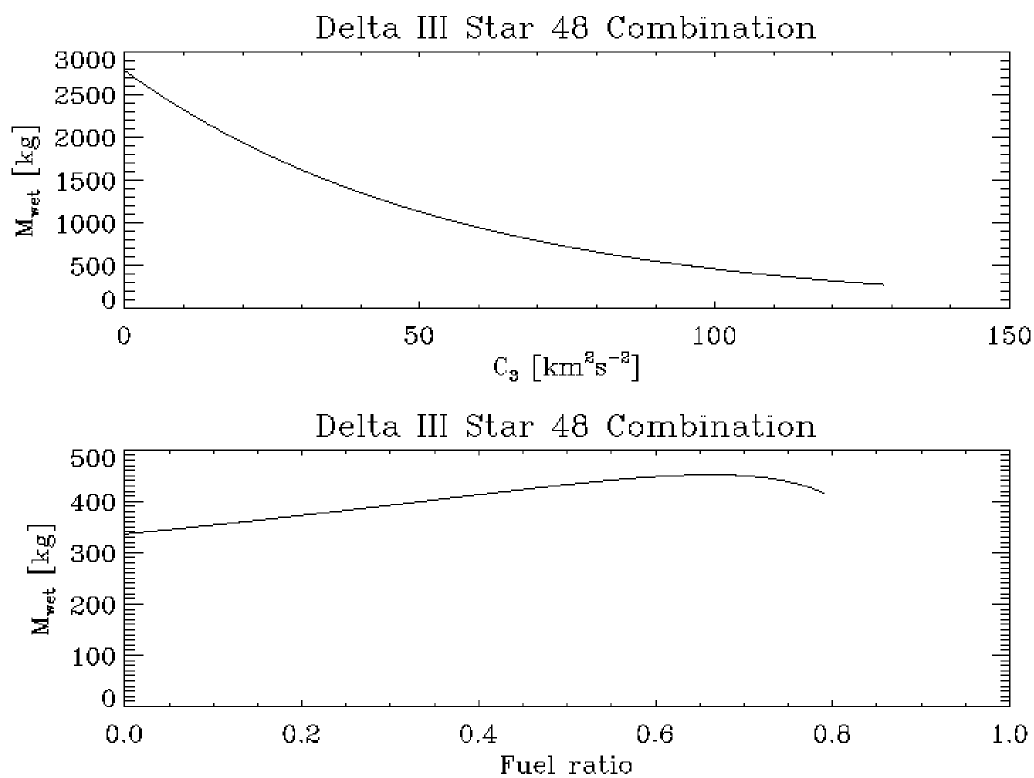


Figure 1.1.2 Approximate Launch Capability of a Delta III/Star 48 Combination. (Top) Capability as a function of injection energy. (Bottom) Capability as a function of the fuel fraction of the additional upper stage. The curve cuts off at high fuel fractions at a $C_3 = 0$ in the absence of the upper stage.

In Figure 1.1.2 we show the available launch mass for a Delta III/Star 48 combination as a function of C_3 (top) and the available launch mass as a function of the additional stage fuel fraction (bottom). These curves are approximate and based upon a fit to the actual Delta III/Star

48 performance curve:

$$m_{wet} = 2776.8 \exp(- C_3/55.463)$$

(linear regression coefficient for ln m_{wet} versus C_3 of -0.99891). The lower curve is constructed by assuming that m_{wet} includes an upper stage as well as the probe such that the overall fuel ratio η is specified. This integral stage is then assumed to fire at an altitude of 300 km (following separation of the Star 48 stage) where the local escape speed from Earth's gravitational field $v_{local\ escap}$ is 10.93 km-s^{-1}. We assume the integral upper stage itself has a fuel fraction of $\varepsilon = 0.93$ and a specific impulse of 290s (performance characteristics typical of the newer and larger Star series of upper stages).

The extra incremental speed that can be imparted by this upper stage is

$$dv = -g I_{sp} \ln(1-h)$$

If we require a given C3 for the mission (117.1 for reaching Jupiter to aim for e Eridani), then the speed of the vehicle at the reference level is

$$v1 = (C_3 + v_{local\ escap}{}^2)^{1/2}$$

The energy that needs to be imparted to the entire probe and integral engine is

$$C_{3,\ new} = (v1-dv)^2 - v_{local\ escap}{}^2$$

Using this energy, we can calculate the net wet mass for the probe (the total mass less the integral upper stage fuel, casing, and associated hardware) as

$$m_{wet,\ indirect} = (1 - h/e)\ 2776.8 \exp(- C_{3,\ new}/55.463)$$

This quantity is plotted in the lower graph of the figure. The analysis shows a significant increase in performance, as could be expected with a local maximum of 452 kg for $\Delta V = 3.07$ km-s^{-1} and $C_{3,\ new} = 32.1$.

To provide for a real system, one must choose an available Star motor of about the correct fuel loading and then rework the problem with the actual Delta III/Star 48 performance curve. Table 1.1.4 we show inject mass limits for various launch vehicle configurations. In this example, a Star 37 FM (I_{sp} = 289.8s, fuel fraction = 0.929, fuel loading = 1065.96 kg) is integrated with the probe assembly. By providing 1.08 km/s in Earth's gravity field at an altitude of ~300 km this stage provides the extra lift capability (see also, *Meissinger et al.*, 1997]. Allowing ~17 kg for hardware associated with ejecting the empty motor casing and 81.5 kg for the empty casing itself, we obtain a net mass of 475 kg injected into a trans-Jupiter orbit. A 30% mass reserve still leaves a design mass of 365 kg, larger than the injection mass provided by an Atlas IIAS/Star 48 with no reserves. This example has not been optimized, but illustrates that a solution exists for sending the probe to Jupiter with the Delta III ELV.

With a target probe final design mass of 50 kg we must add about 40 kg for a Sun shield (this assumes six plys of carbon-carbon configured as a circular shield ~4.1 m in diameter - the best of the thermal configurations studied, as described in more detail below. We add an additional 50 kg for the structure to hold the Sun shield in place, including the means to jettison it and the perihelion propulsion module once the probe is outbound from the Sun to obtain a net mass of 140 kg. Including a 30% mass reserve brings the allowable probe mass to 182 kg. During Phase II, the design for the integrated upper stage/probe/Sun shield/structure will be further refined with the goal of looking for further mass savings while optimizing the upper stage implementation.

We have found that constraints already imposed by the locations of Jupiter, Earth, and the target asymptotic direction preclude the use of other planetary gravity assists without using much more time in the inner solar system "setting up" the Jupiter swingby, e.g., using multiple Venus flybys over a several-year period. An Earth-gravity assist can be used but requires a substantial

Launch Vehicle	Upper stage(s)	Injected mass (kg) at C_3 = 117.1 km^2 s^{-2} (ε Eridani)	Injected mass (kg) at C_3 = 100.6 km^2 s^{-2} (36 Ophiuchi)
Atlas IIAS	Star 48	342	500
Delta III	Star 48	256	425
Delta III	Star 48/ Star 37FM	~475	Not evaluated
Launch date		February 18, 2019	July 12, 2011

Table 1.1.4 Injected Mass Limits

deep space ΔV maneuver (with yet more propulsion and tankage) as well as an Earth return with radioactive material onboard; recent experience with Cassini and its radioactive power source gives another strong reason for not implementing such a mission design. Hence, the only way of further increasing the probe mass and/or margins for a given launch vehicle is by further optimization of the probe, propulsion, and thermal design. The various mission design scenarios considered in this study are shown in Table 1.1.5. Depending upon the various features that emerged, some designs were carried out in more detail than others.

Target (Spectral type)	Distance (LY)	Ecliptic orientation		Launch					Jupiter flyby					Perihelion (4 R_S)					1000 AU
		Lat	Lon	Date	Lat	Lon	DLA	C_3 (km^2 s^{-2})	Date	Lat	Lon	Distance (R_J)	Speed (km s^{-1})	Date	Lat	Lon	ΔV (km s^{-1})	V_{asymp} (km s^{-1})	
36 Oph (K1, K1, K5)	18.2	-3.540	259.960	2019 February 18	-0.001	149.686	-22.0	100.6	2020 July 31 @ 5.1506 AU	-0.294	293.529	11.50	12.08	2022 July 31	0.318	113.315			2070 Nov 9
36 Oph (K1, K1, K5)	18.2	-3.540	259.960	2019 February 18	-0.001	149.686	-22.0	100.6	2020 July 31 @ 5.1506 AU	Aim point slightly off 36 Oph		11.50	12.08	2022 August 10			15.800	20.718	
36 Oph K1, K1, K5)	18.2	-3.540	259.960	2016 March 9, ΔV of 0.887 km/s @ 3.203 AU - flyby Earth 2019 January 9 @ 1554 km altitude and 14.03 km/s			-23.0	48.2	2020 July 16 @ 5.1506 AU			11.71	12.07	2022 August 10			14.481	19.781	
α Cen (G2, K5)	4.3	58.232												63.574 MINIMUM REQUIRED				43.679	
191408 (K3)		-15.676	297.055	2020 April 14				134.4	2021 August 11			10.34		2023 July 24					
ο^2 (40) Eri K1, DA, M4)	16.0	-28.387	60.236	2035 July 21				119.3	2036 November 2			6.39		2038 August 25					
ε Eri (K2)	10.8	-27.760	48.290	2011 July 12 @ 1.0166 AU	0.001	289.350	3.9		2012 Oct 26 @ 5.0443 AU	0.705	67.721			2014 Nov 28	0.538	247.827			2064 May 31
τ Ceti (Case 1)	11.8	-24.77	17.66	2022 June 9				115.62	2023 September 18			8.99	13.05	2025 Sept 20			12.699	18.430	
τ Ceti (Case 2)	11.8	-24.77	17.66	2034 June 14				115.13	2035 September 26			8.99	12.91	2037 Sept19			15.810	20.711	
Heliosphere nose	~4.74 x 10^{-3} (~300 AU)	-7.80	74.90	2024 August 21				123.7	2025 December 5			3.56		2027 June 13					

Table 1.1.5 Details of Mission Scenarios Studied

1.2 Propulsion

The basic equation that governs spaceflight is the rocket equation:

$m_{initial}/m_{final} = \exp(\Delta V/I_{sp}g)$ (1.2.1)

ΔV is the change in speed, I_{sp} is the specific impulse ($I_{sp}g$ is the exhaust speed), and $m_{initial}$ and m_{final} are the original mass and the mass following the speed change, respectively. Relativistic corrections were derived over 50 years ago [*Ackeret*, 1946], and these equations were soon applied to the idea of interstellar travel. Acceleration of 1 Earth gravity (1g) for ~1 year suffices

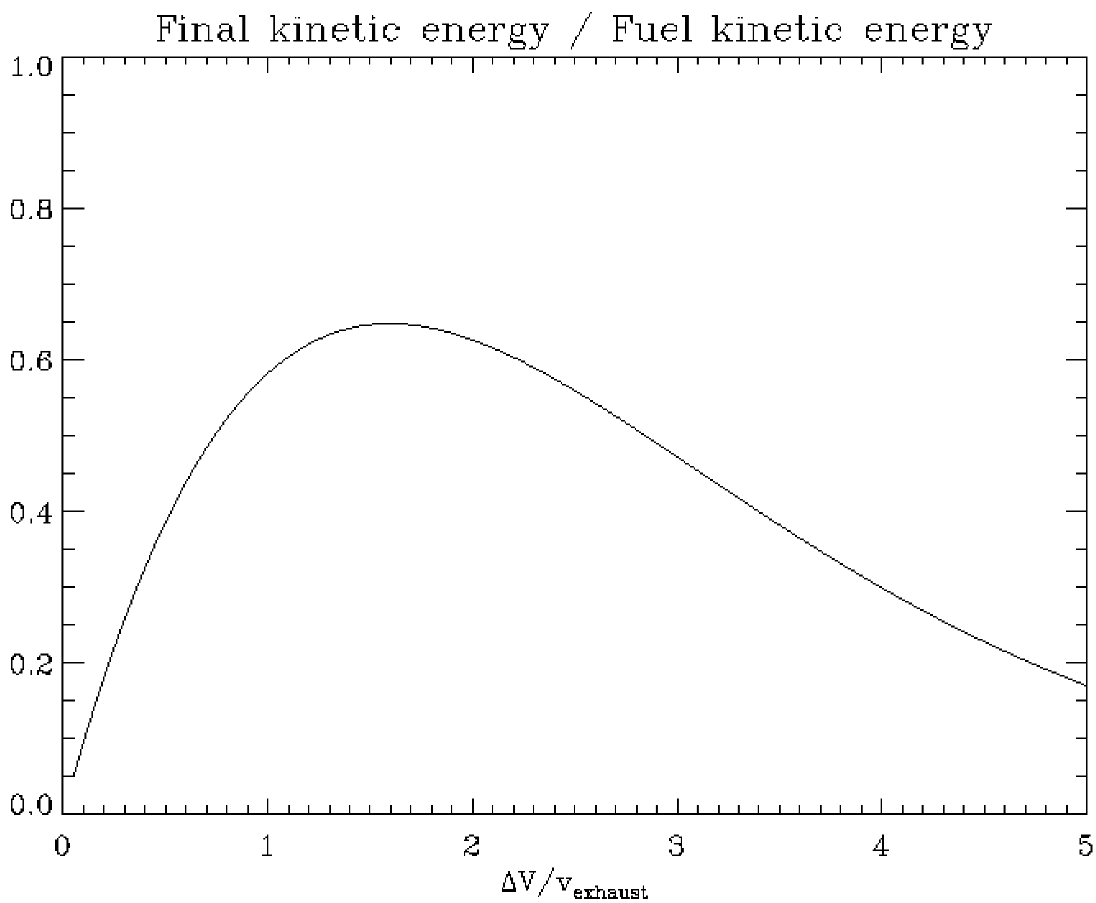


Figure 1.2.1 Kinetic energy Efficiency of a Rocket as a Function of the Ratio of ΔV to the Engine Exhaust Speed

to bring the vehicle to near light speed, producing significant time dilation. To attain high speeds efficiently, exhaust speeds also need to be near light speed; the large speeds required still translate into large mass ratios, requiring in turn, large mass flow rates. These constraints, in turn, translate into the need for fusion or photon rockets [*Shepherd*, 1952; *von Hoerner*, 1962; *Sanger*, 1962; *Sagan*, 1963; *Forward*, 1996].

To date most rockets have relied upon chemical energy for their primary propulsive means; the only exception is Deep Space 1 (DS-1) currently operating with 30-cm Kaufmann electrostatic thrusters using xenon (Xe) as the "fuel." While equation (1.2.1) provides some insight into the constraints associated with space travel, it does not tell the entire story. Especially for more exotic concepts, one also needs to worry about the size of the physical power plant and means required to implement (1.2.1). In particular there is an efficiency [*von Hoerner*, 1962] given by

$$\eta = 1/2\, m_{final}\, \Delta V^2 / 1/2\, m_{fuel} v_{exhaust}^2 = (\Delta V/v_{exhsus)t})^2 / (\exp(\Delta V/v_{exhaust}) - 1) \qquad (1.2.2)$$

The efficiency is shown in Figure 1.2.1. There is a significant maximum in the efficiency ($\eta = 0.64761$) at $\Delta V/v_{exhaust} = 1.59363$ and corresponding to a mass ratio form (1.2.1) of 4.92158. Typically, all of the energy associated with the denominator of (1.2.2) has come from the inherent release of chemical energy while the desire has been to operate away from the exponentially growing part of (1.2.1). The highest packing fractions of fuel are possible with solids that operate at I_{sp} ~200 to 300s. Space-storable bipropellants can operate in a blow-down mode with specific impulses of ~310s to ~350s (projected) for some exotic combinations, but at a cost of more mass associated with the propulsion system hardware. The highest specific impulses > 400s are achieved with cryogens, e.g. liquid oxygen (LOX) and liquid hydrogen (LH_2). These fuels must be used immediately to avoid cryogen-storage thermal requirements (and hence added system mass) and also require pumping, again adding to the system mass "overhead." Hence there is a "hidden cost" as we go to higher ΔV: the required specific impulse increases (in order to stay away from an exponential increase in fuel required by (1.2.1)), but this also drivers the fixed mass of the power plant and overall system mass and size.

Once we move away from chemical rockets, there are new constraints. If energy is now provided by some other means than the inherent chemical energy of the fuel, then this energy must come from some other source. In turn, for such a system to apply this energy in an efficient manner, the I_{sp} and required ΔV. Finally, the required power is then set by the required energy and the propulsive time and is just equal to their ratio for a constant mass flow rate and constant exhaust velocity.

For the reference mission launched toward e Eridani, the required DV is 15.393 km-s^{-1} and the optimal exhaust velocity and corresponding specific impulse are 9.6591 km-s^{-1} and 984.6s, respectively. For a total dry mass discussed above of 475 kg at perihelion (including the probe, thermal shield, propulsion system, and all hardware) and the optimal mass ratio of 4.92, we would require 475(1-1/4.92) = 378.5 kg of fuel (~96 kg of probe mass total). The corresponding total energy required is 0.5 x 378.5 x 9659^2 = 1.77 x 10^{10} joules. Allowing for a 15m = 900s burn time, the perihelion engine must develop a power of 19.6 MW while it is operating. Higher specific impulse allows operation with a lower mass ratio (smaller fuel fraction), but the total power expended to provide the required performance is higher. Lower specific impulses require larger fuel fractions as well as less efficient use. For example, to provide 15.393 km-s^{-1} with a specific impulse of 325s (current technology, non-cryogenic bipropellants) would require a mass ratio of exp(4.83) ~125. For a total mass of 475 kg, the probe (including the tankage!) would require a mass of 3.8 kg; the expended energy would be 4.8 x 10^9 J with a required power of 5.3 MW - but in a <3.8 kg system! For a dry mass of 96 kg and a mass ratio of 125 we would need 11,904 kg of fuel, and total energy and power of 6.05 x 10^{10} joules and 67 MW, respectively.

It is worth mentioning that very high Isp ion thrusters are also very inefficient in their power consumption if the ion exhaust speed is >> the ΔV required.

Chemical production of such high specific impulses is not possible; such large ΔVs are only attainable with large amounts of fuel and staging, e.g. the 19.7-metric ton IUS/PAM-D combination required to boost the 360-kg Ulysses spacecraft by 15.4 km-s^{-1}. The only other means of propulsion that do not rely on exotic - and currently unfounded - physics use either nuclear energy produced on board the spacecraft or energy beamed to the spacecraft, the latter including the use of sunlight for Solar Electric Propulsion (SEP), solar sail propulsion, and lasers or microwaves for beamed energy propulsion. The use of microwaves as a propulsive force has been discussed by *Forward* [***1996***] and the use of lasers is incorporated in the sail technology roadmap as a follow on to solar sail propulsion [*Wallace*, 1998]. These beamed-energy schemes require large power-production infrastructures, albeit on the "ground" and are beyond the scope of our current work, As noted previously, solar-sail propulsion is currently under study for a near-term interstellar precursor mission with a candidate launch date of ~June 2010.

SEP is now being demonstrated with DS-1. The utility of SEP is limited by the mass of the solar arrays, ion thrusters, and power processing units, currently ~65kg/kW. Ion thrusters have also been considered for implementation in Nuclear Electric Propulsion (NEP) schemes that rely upon nuclear (fission) reactors for providing power. However, such systems are heavy, the aborted Nuclear Electric Propulsion Space Test Program (NEPSTP) baselined a variety of thrusters and a then-Soviet TOPAZ II nuclear reactor. The reactor provided ~6 kW of power with a 1000 kg reactor mass (~167 kg/kW). The spacecraft had a wet mass of 3500 kg, including 700 kg of Xe fuel [*Mauk et al*., 1993]. The program was cancelled due to changing national priorities and budgetary pressures. The SP-100 Flight Experiment had, at one time, baselined a 93 kW ammonia arcjet system with a reactor specific mass of 30 kg/kW. The system would run at an I_{sp} ~1000s and a mass flow rate of ~0.25 g/s [*Deininger and Vondra,* 1991]. This system has remained a concept only to date. The baseline design discussed

Pure ion engines are limited by the space-charge within the engine chamber. Allowable current densities are given by the Child-Langmuir relation; minimal spacing between electrodes is set by field emission of electrons if the electric field is increased to too large a value. For example the NASA SEP Technology Readiness Application Readiness (NSTAR) program uses Xe ions and runs at a specific impulse of 1990s at minimum power and 3280s at maximum power. Corresponding mass flow rates vary from ~0.94 milligram/s to ~2.55 milligram/s. About 8000 hours is required to use ~83 kg of propellant; this also corresponds to the engine lifetimes using current technology [*Williams and Coverstone-Carroll*, 1997]. These low thrust rates are set by the inherent physics of the space-charge within the engines, so significant increases in the mass flow rates do not seem likely. The use of radioisotope electric propulsion has also been discussed; current specific masses are ~200 kg/kW but could possibly be decreased by a factor of ~4 with technology development [*Noble*, 1996].

All of these concepts relying on electric thrust and solar- or nuclear-derived electrical power suffer from mass flow rates that are too small by orders of magnitude to meet the perihelion thrust requirement. Although the direct use of nuclear reactions is a well-known (and well-studied!) subject, [*Cassenti*, 1996] typical system sizes are driven to large values by considerations of the means of applying the energy source to propulsion.

1.2.1 High Specific Impulse/High Thrust Concepts

We have considered two concepts in some detail to identify a means of applying the ~15 km-s^{-1} ΔV required at solar perihelion: (1) an Orion-type nuclear drive and (2) solar thermal propulsion. In comparison with sail, SEP, and NEP, the requirement is to provide a high thrust for a brief period of time similar to conventional rockets and upper stages, but, significantly, at a specific impulse about 3.5 times larger, i.e. at I_{sp}~1000s.

1.2.1.1 Orion and Nuclear Pulse Propulsion

The Orion project took place during the late 1950's through the early 1960's to attempt to design a practical rocket relying upon nuclear fission explosions to drive a rocket. At face value nuclear explosions easily contain sufficient energy to power the propulsion system: The energy release in the explosion of TNT is ~4 x 10^{10} erg/g while that in fission is ~7 x 10^{17} erg/g (1 kiloton - kT - is typically taken as 10^{12} calories, and 1 kg of totally fissioned uranium can release ~10 kT, *Serber* [1992]). Hence for the ideal case mentioned previously, with ~2 x 10^{10} joules required, we need the fission energy of ~0.5 g of uranium - a total of about 5 tons of TNT equivalent.

The problem with such systems is the coupling of the momentum into the ship over short time scales ~10^{-8} s. Transferring significant amounts of impulse to a structure over such short times typically causes stress to exceed the yield strengths of all known materials [*Dyson*, 1965; 1968]. The Orion concept [*Boyer and Balcomb*, 1971] and its derivatives [*Solem*, 1993; 1994] require large masses for dealing with the release of ~1 to 10 kT explosions. The alternative is to go to much smaller explosive yields and pulse the system. Ideas adapted from inertial confinement fusion with yields ~0.01 kT have been adapted to a variety of systems studies; however, the spacecraft masses again tend to be large due to the power plant overhead, here the laser system, required to initiate the microexplosions [*Hyde et al.*, 1972].

Small yields are possible (and termed "fizzles" [cf. Chapter 18 of *Winterberg*, 1981; Chapter 17 of *Serber*, 1992]), although still in yields of 10s of tons for a typical weapon configuration. Here the problem is that although very little fission energy is needed per pulse, a minimum mass is still required in order for criticality, in turn needed to provide neutron multiplication for sufficient energy to be generated in a microexplosion (about 20 doublings as compared with ~58 in a weapon [*Serber*, 1992]). The actual yield can be reduced by using tampers, but only at the expense of mass. Whether a workable system based upon fission microexplosions can be implemented at a low system mass is still questionable, but worth pursuing. The real question is whether appropriate geometries can be assembled and initiated, e.g., using a DT neutron source that uses a small (~120 keV) linear accelerator to produce fast neutrons. Here the idea is to drive the system supercritical in a pulsed mode, similar to the idea behind the high-thrust nuclear salt water rocket concept [*Zubrin*, 1991] but at very small system masses.

In particular, we noted in our original Phase I proposal that the "Davy Crockett" or W54-2 is reported to have had an explosive yield of 0.25 kT, with a mass of ~23 kg [*Cochran et al.*, 1984]. Furthermore with the allocation of mass for the perihelion propulsion system of 216 kg, 9 of these devices plus 9 kg for a deployment mechanism one might construct a simplified Orion-type nuclear pulse propulsion system [*Dyson*, 1965; 1968]. However with a velocity transfer per detonation limited to ~30 ms^{-1} *Dyson* [1968] acceleration by 15 kms^{-1} would, therefore, require some 500 detonations, suggesting a system mass of ~500 x 23 = 11,500 kg. What is really required is a yield of ~ 5/500 = 0.01 T each, a fissioning of ~1 milligram of uranium about every 2 s. In a fission detonation *in vacuo*, most of the energy released goes into X-rays [*Hammerling and Remo*, 1995], but this can be captured into debris motion with sufficient high-atomic weight material (in contract with a fusion detonation where the neutrons carry away far more energy.

Even at 4 R_S the solar wind is sufficiently tenuous that little material can be entrained by the blast debris, even with a relatively large tamper. Hence, the time of interaction between the probe and the detonation wave will be increased to at most the shock wave travel time through the tamper material. For a fission temperature ~2 keV [*Hammerling and Remo*, 1995], uranium vapor ions will acquire a thermal speed ~40 km-s^{-1} at thermal equilibrium. In a 23 kg device, e.g., the W54-2, a uranium sphere would have a radius of ~6.6 cm (and slightly smaller for more dense plutonium). This suggests at best a time dispersion in the debris cloud (assuming 100% conversion of all radiation into debris energy) of ~6.6/4 x 10^6 = 1.6 microseconds. Although this is a factor of ~100 slower than the actual fission release [*Serber*, 1992], it is still a very short time compared with the delivery of ~1/9 of the total required impulse.

For a delivery of all of the impulse from one of these devices over ~100s with the debris moving at ~40 km-s^{-1}, the material would have to be distributed in an extended shell ~4000 km across. At this distance, the energy incident on, e.g., the Sun shield would be ~250 T x (4 m/4 x 10^3 m)^2 ~2.5 x 10^{-4} T or ~10^6 J. This is far from an exact calculation, but illustrates the main difficulty of the problem: one requires very small explosive yields in order not to destroy the probe. The alternative of locating the probe far enough away to mitigate short impulse delivery time decreases the energy delivered by too large a factor. Even "microexplosions" tend to drive the propulsion system to large masses (10s of tons), as we have already noted.

The use of fission is preferred to fusion for smaller vehicles as the soft X-rays and fission products from fission have much shorter stopping distances in matter than the neutrons produced in fusion [*Hammerling and Remo*, 1995]. The problem thus becomes one of how small a fissionable mass can be in order to achieve criticality and fission significant generations of uranium (or plutonium atoms). For simple geometries, the problem is well posed but still difficult to solve [cf. Section 6.6 of *Case and Zweifel*, 1967] (so difficult that the wrong solution by Heisenberg may have cost Germany World War II [*Logan*, 1996]). Whether fast criticality can be controlled in a rocket engine remains an open question. For example, it is not clear the nuclear salt water rocket [*Zubrin*, 1991] can actually achieve criticality; a major problem is the hydrodynamic disassembly of the fissile material even prior to a nuclear fizzle [*Serber*, 1992].

Ideally, a pulsed-mode autocatalytic reaction [*Serber*, 1992; *Winterberg*, 1981], similar to the operation of a pulse jet, e.g., the German V-1, is preferred. Compression of the fissle material helps [*Winterberg*, 1981] but can rapidly run up system masses required for compression, a major problem with fusion drive systems [Hyde et al., 1972]. This type of rocket would represent the next step past a gas core nuclear engine, and, if doable, could perhaps bypass the engineering problems of the latter. This is a goal worth pursuing and would clearly be an enabling technology for this interstellar precursor mission as well as higher-speed missions to the stars.

1.2.1.2 Solar Thermal Propulsion.

Solar thermal propulsion uses thermal energy from the Sun to heat a low-molecular-weight working fluid to a high temperature (~2400K) and use the thermal energy to expel the propellant mass from the system, driving the probe forward. At 4 R_S, the solar luminosity provides ~392 W cm^{-2}. The exhaust speed $v_{exhaust}$ is given by

$$v_{exhaust} = w\,(\gamma/\gamma\text{-}1)^{1/2}$$

where γ is the ratio of specific heats and w is the thermal speed defined by

$$kT = 1/2\ mw^2$$

An "obvious" choice for a working fluid is liquid hydrogen (LH2) due to its low molecular mass. We can estimate that at 2400K, the exhaust speed is ~8.33 km/s corresponding to a specific impulse of 849s. Another choice is ammonia, which is not a cryogen and so can be stored more easily (the flight time from Earth to the solar perihelion point is ~3.6 yr). Fully dissociated ammonia yields a specific impulse of ~400s.

Weikenheiser et al. [1991] estimated that for a large nuclear thermal propulsion (NTP) system, a tank mass/propellant mass fraction of 0.16 may be possible. Allowing 307 kg for the propulsion system mass allows a fuel load of 264 kg of LH2. At I_{sp} = 849s, the probe achieves a ΔV of 6.77 km/s, less than half of what is needed for the 20 AU/yr mission toward ε Eridani. These numbers may be very optimistic for small systems, Cryostats for long-term containment of LH2 roughly follow

$$m_{dry} = 0.15\ \text{kg/liter}\ \ V + 200\ \text{kg}$$

(volume V in liters) for the storage of LH2 at 20K and 1 atmosphere, including cryogen tank, vapor cooled shields, insulation, support structure, vacuum shell and plumbing. With a density of 0.070 kg/liter, in terms of the mass of LH2 this equation becomes

$$m_{dry} = 2.14\ m_{LH2} + 200$$

suggesting a substantial mass penalty for long-term LH2 storage for use as a working fuel at perihelion. As an example, for achieving a ΔV of 15 km/s at I_{sp} = 849s (2400K H_2 gas), the mass ratio is 6.056; a dry mass of 120 kg would require ~556 kg of fuel. Using this equation, the dry cryostat mass would be ~1400 kg for 566 kg of LH2. While enabling a high specific impulse, such a system may bee unworkable due to the mass required for the fuel storage.

The use of ammonia enables standard pressurized titanium tank technology (cf. the ammonia arcjet system described in *Deininger and Vonra* [1991]. Allowing about 0.02 for the tank mass fraction, we could accommodate up to ~301 kg of ammonia. At a 400s I_{sp}, we can reach ΔV ~3.93 km/s. To reach the required performance, the probe mass has to be smaller, the throw weight larger, the specific impulse increased, or all of the above in some measure. At a density of 682 kg/m^3 at its boiling point, the required tank volume is ~0.44 m^3, significantly less tankage to maintain behind the thermal shield than in the case of LH2.

Thiokol Corporation is currently engaged in STP research with the goal of reaching I_{sp} ~1000s for an LH2 system and ~600s for an ammonia system (Lester, private communication). Their current research is focused on orbital transfer vehicles in Earth orbit; operation near the

Sun actually may simplify some of the STP concept design. These higher specific impulses would increase the performance to 7.97 and 5.89 km/s, respectively. More work is required to examine the possibility of conceptually assembling an actual system. The bottleneck,, in addition to mass, is providing sufficiently rapid heat transfer to the fuel in the near-Sun environment in order to provide efficient thrusting.

1.4 Thermal System

The shield design is extremely challenging from a thermal perspective. Passing within an altitude of 3 solar radii of the sun poses a few thermal problems. Three different shield designs have been investigated to determine the most feasible approach to protecting the spacecraft from the intense solar illumination. The solar flux at this distance from the sun is well above 200,000 Watts per square foot, compared to the 1300 W/ft^2 at Earth's location from the sun. Each design will be discussed with the temperature profile for the spacecraft and shields shown.

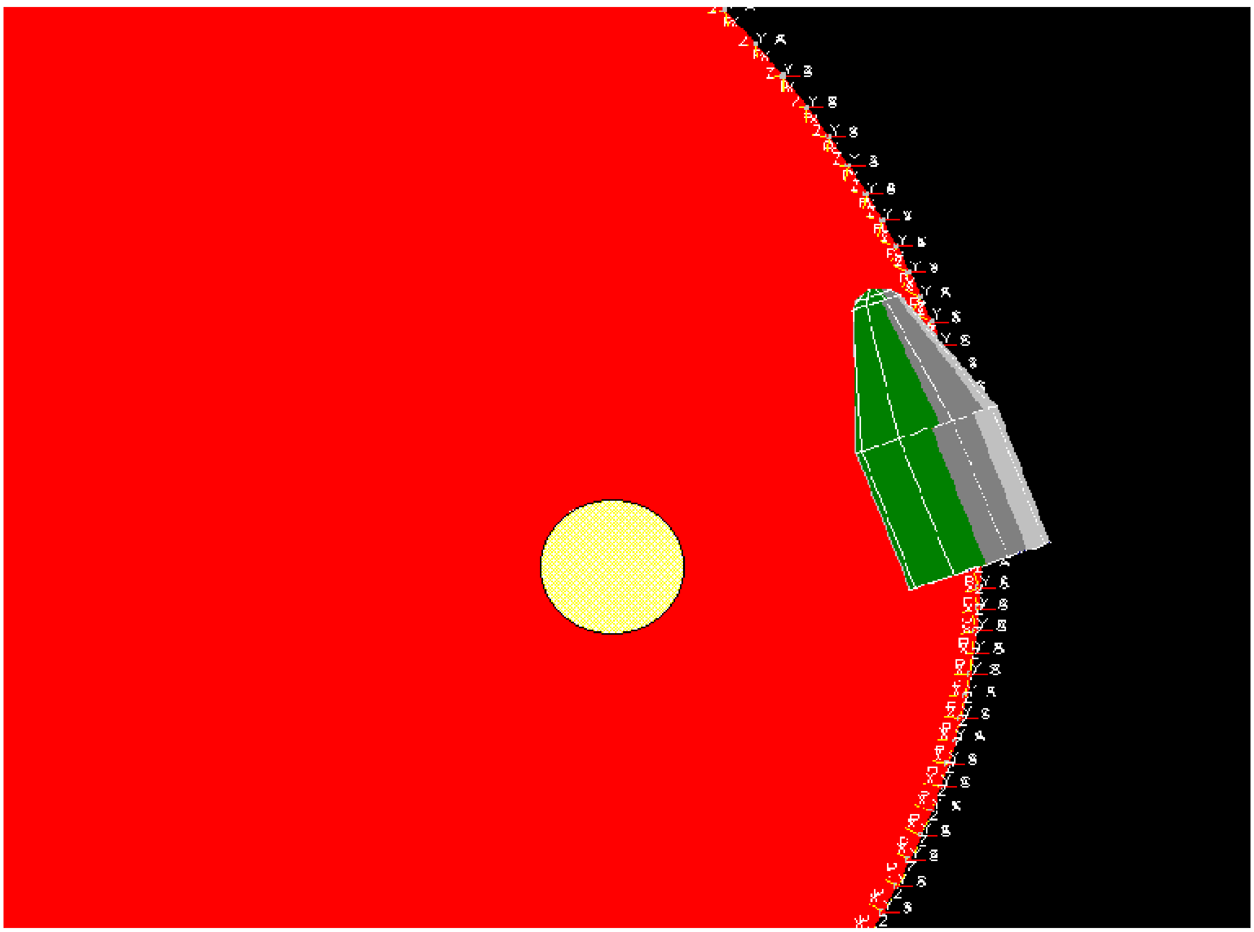


Figure 1.4.1 "Bullet" Thermal Design Concept

The "bullet design", as shown in Figure 1.4.1, was the first considered as a possible means of protecting the stowed probe and the propulsion module apparatus from the sun. The idea was to create a shield that, while spinning, would reject the heat that was absorbed on the sun side as the shield turns to cold space. To prevent the sun from hitting the aft end radiator, the nose of the

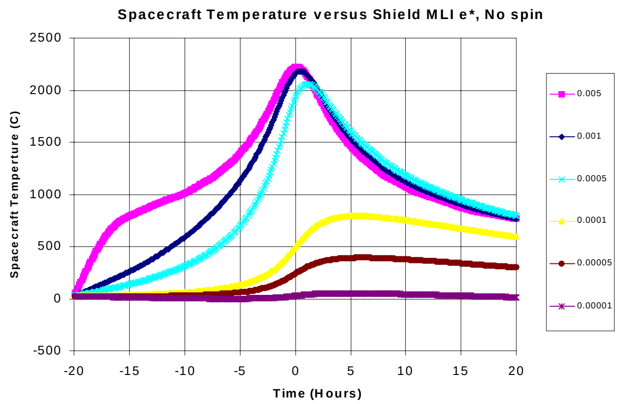


Figure 1.4.2

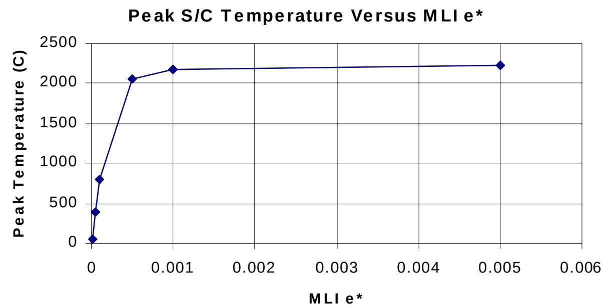


Figure 1.4.3

shield must be turned 18.5 degrees or greater towards the sun. Figure 1.4.1 shows the vehicle as it passes within an perihelion altitude of 3 solar radii of the sun. However, the duration of the perihelion event is over 40 hours, which is too long to prevent the heat from the shield from soaking back into the spacecraft. The aft end of the spacecraft would be used as the main radiator to cool the vehicle during the perihelion pass. Figure 1.4.2 shows the transient response of several surfaces on the spacecraft and shield as a function of time and effective emittance. Figure 1.4.3 shows the peak spacecraft temperatures extracted from Figure 1.4.2 to illustrate the shape of the effective emittance versus temperature curve. The effective emittance (or e*) is a measure of the thermal performance of Multi-Layer Insulation (MLI) blankets. The smaller the e* the more insulating the blanket and smaller the heat flow through the blanket. From the figure, an e* value of 0.00002 or better is required to insulate the spacecraft from the hot shield, maintaining the vehicle below 100 C. Current technology allows from e* values no smaller than 0.001, given great care and expense. Thus at least 2 orders of magnitude improvement in thermal isolation is required to make this design feasible.

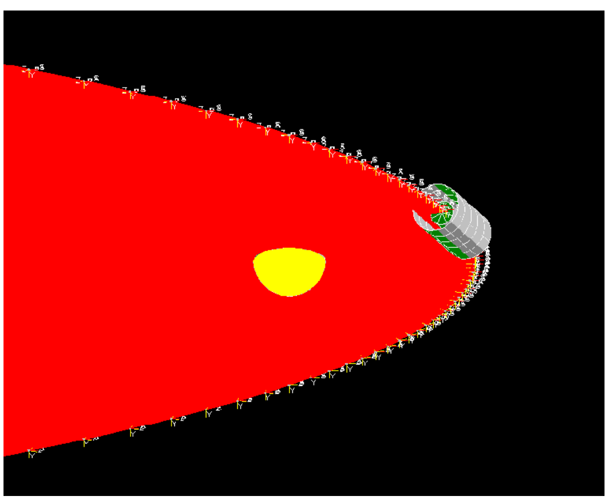


Figure 1.4.4

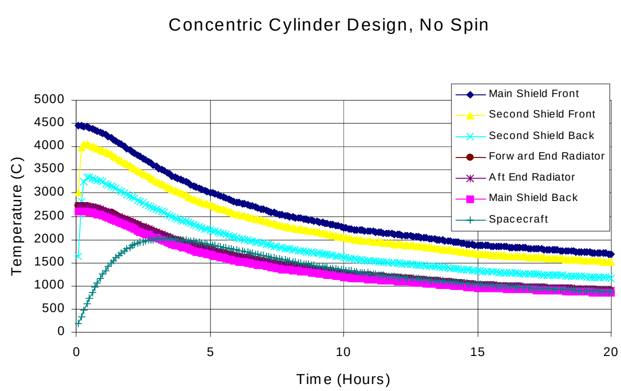


Figure 1.4.5

Figure 1.4.4 shows two concentric cylinders, with the outer cylinder the shield and the inner cannister the spacecraft. The inner cylinder is shorter than the shield to allow for the 18.5 degree solar umbra at 3 solar radii above the sun. Thus, the idea was to allow the sun to hit the inside of the shield on the back, as well as on the outside. However, as shown in Figure 1.4.5, the radiators on the spacecraft, which are the two ends of the canister, have large views to these very hot surfaces. Figure 1.4.5 shows the resulting temperature response of the spacecraft as the shield passes the perihelion point in orbit.

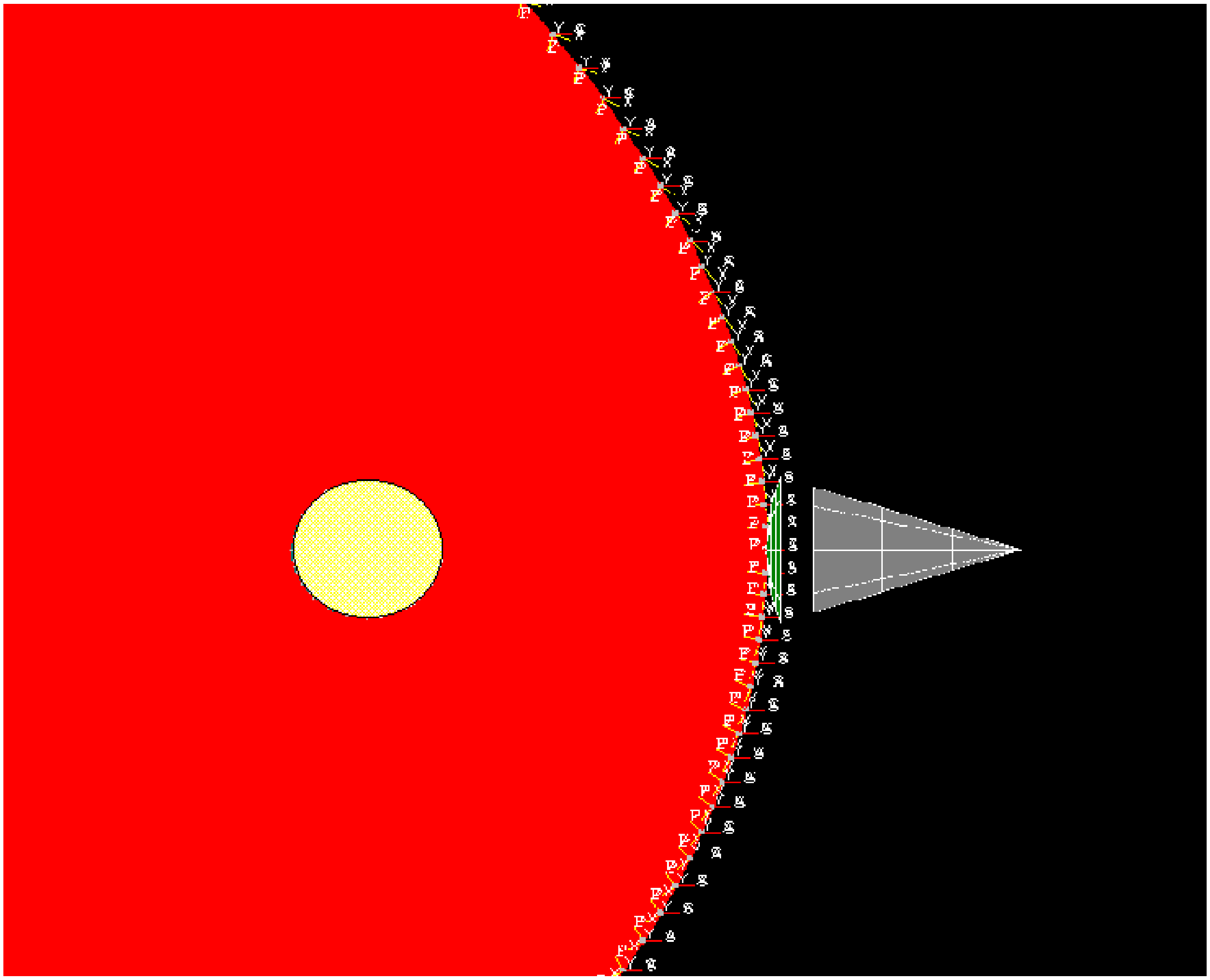
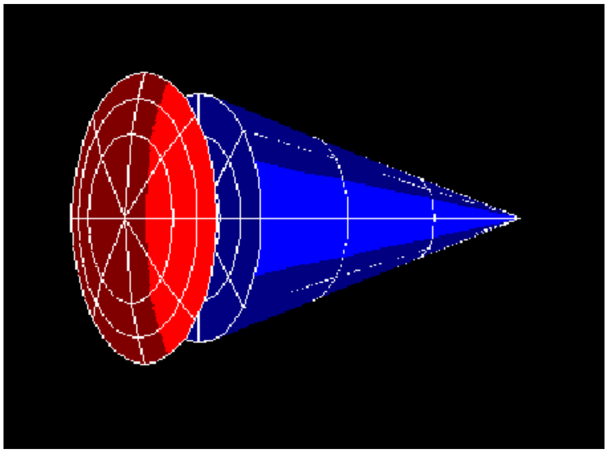


Figure 1.4.6

The last configuration considered is shown in Figures 1.4.6. The carbon-carbon spherical shield protects the vehicle from the sun. The conical shape shown represents the available volume for the probe and the propulsion model. The shield is spaced away from the spacecraft to allow the backside of the shield to radiate its heat to space, as opposed to into the spacecraft. A secondary shield, one foot away from the spacecraft, further reduces the thermal soak back. A standard MLI blanket is used to protect the spacecraft from the backside of the secondary shield. Figure 1.4.7 shows a heat flux plot illustrating that the shield does protect the spacecraft. As Figure 1.4.8 illustrates, although the shield temperature approaches 2700 C, the spacecraft is protected. Furthermore, since the shield is much smaller than the other designs, the available view to space for the spacecraft radiators is drastically increased. Thus, it is not a problem maintaining the spacecraft with normal operating temperatures. Note in Figure 1.4.8 that the spacecraft



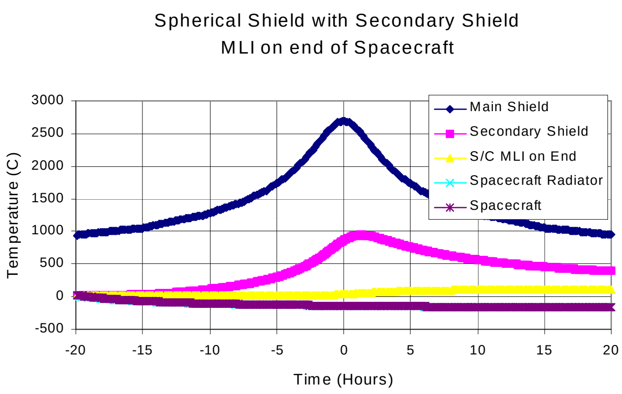


Figure 1.4.7 **Figure 1.4.8**

temperature actually decreases as function of time, which indicates that the available radiator swamps the conducted and radiated heat soaking back through the shields.

This design is different from the other two designs, in that the thrust from the propulsion module must be at a right angle to the main vehicle axis. The bullet and concentric cylinder designs allow production of thrust along the spin axis of the vehicle. This design must be three-axis stabilized, while the other two are spin stabilized.

The thermal shield also must be incorporated into the general mechanical design that must both accommodate the propulsion system and allow for the spacecraft to work under Orion-type or solar-thermal propulsion conditions.

2. Interstellar Probe

The term Interstellar Probe is used to describe the probe which separates from the propulsion system after the perihelion burn and which contains the science payload and all other spacecraft elements necessary to continue the journey into interstellar space. When we refer to the Interstellar Explorer, we mean the entire mission, including the probe as well as the propulsion system which propels the probe.

The Interstellar Probe (IP) has a number of elements which are similar to the conventional spacecraft that currently explore the solar system, such as Galileo and Cassini. However, due to the large distances and long travel times involved with a mission to the interstellar medium, there are a number of aspects of the probe's architecture that must be significantly enhanced or completely redesigned in order to achieve mission success. In the following sections we will discuss each probe subsystem in detail, outlining the requirements which drive each subsystem's design, the various possible implementations and their relative merits, and the characteristics of the selected approach.

2.1 General Probe Mission Aspects

The extremely long duration mission that results from sending the Interstellar Probe to such enormous distances imposes a stringent set of environmental requirements on the Probe design. This means that a number of standard techniques and subsystems normally used on shorter missions within the solar system are not appropriate for a mission of this type. This reality forced a thorough top-down redesign of the architecture of the spacecraft and its subsystems. For example, it was deemed impractical to fly any components that used moving parts, such as momentum wheels, scan platforms or thrusters that incorporated valves. Additionally, any subsystem that required consumables was carefully evaluated. The only consumables permitted on the probe are blocks of solid Teflon used by the thrusters. Thrusters that require liquid or gaseous propellant were excluded.

The Interstellar Probe must have the characteristics of robustness, reliability and adaptability, since, for the most part, it must perform it's mission with minimal external help. The mission scenario which we have developed involves maintaining the IP in a dormant, cocooned mode during the early phase of the mission while the propulsion system carries the probe through the planetary swingbys and finally the near Sun swingby and perihelion burn maneuver. This phase, occurring well inside the solar system, includes the regions of highest radiation, acceleration, and temperature. The probe is protected by the substantial structure and thermal shield of the propulsion system. Since other spacecraft have already studied this region of space, there is no loss of science return by having the probe turned off.

2.1.1 Shakedown Cruise Period

Once the propulsion system has completed the Sun swingby it is left behind and the probe continues on its journey. At this point the initial checkout of the probe commences. During this period two way communication with the spacecraft is required. The duration of the Interstellar Probe shakedown period is dependent on how long two-way communications can be maintained. The various communication system approaches (microwave and optical) for the primary long range system are discussed in detail in a later section. However, these systems are very directional, so extremely good pointing is required to maintain a two way link. If there is a problem with the attitude control system, then these system are not useable. Most conventional spacecraft employ a low rate omni-directional link in order to allow commands to be uplinked, regardless of the spacecraft orientation. Due to the power gain of omni systems, they are an inappropriate choice for a secondary communication system on the Interstellar Probe. Hence, an ultraminiature Ka band system will serve as the two-way communication for the probe during its shakedown period. The size of the antenna is very small to minimize mass and to achieve a reasonable amount of onmi-directionality in the beam. As long as the probe can maintain crude sun pointing, the Ka system will be able to establish a link with the DSN. One bit per second downlink is possible with a 2 watt amplifier and 20 cm dish antenna.

Two way communications allows problems to be diagnosed and remedial action taken (such as a software upload) during the shakedown portion of the mission. Since the probe is moving at 20 AU/ year, it is out of range for two way communication in about six months. However, because the uplink margin is much higher for the Ka system than the downlink, due to the use of more powerful transmitters on Earth, it would still be possible to get commands into the probe for a substantially longer time. If the optical downlink system is also working properly, then two way communications can still be maintained.

2.1.2 Prime Science Period

Once two way communication is lost the Interstellar probe must then enter an autonomous mode where only a communication downlink exists and, due to power limitations, transmissions occur fairly infrequently. The short transmission period is dedicated to downlinking acquired science data, as well as for health information. This is the prime science portion of the mission with a time line that runs from perihelion burn + roughly 6 months to 50 years or greater. During this period the following major tasks are performed:

A. Maintain a slow spin with the spin axis pointing back at the Sun. The spinning spacecraft allows the science instruments to see the entire sky.
B. The instruments collect and process their own data. There is no data recorder on board and the task (software routine) associated with a particular instrument must store the data in the processor module's memory on which the code is running (each processor module has at least 128 MB), or the data can be also be stored in the memory of other processor modules which are underutilized. This processing architecture is discussed in more detail later in this section.
C. At regular intervals the probe must point accurately toward the Hubble class receiving station which is orbiting the Earth and then transmit the science data. Unlike a typical “store and dump” spacecraft system which collects all the data from the instruments onto a data recorder and then dumps the data at a high rate to the ground at regular (daily) intervals, the Interstellar Probe is severely bandwidth limited so only highly reduced data can be transferred for short time periods at more infrequent intervals.
D. Continually monitor the health of the probe and take corrective action. Again, this is discussed in more detail in a later section.

2.2 System Configuration

As was stated above in the Explorer propulsion section, it is extremely important to minimize the mass of the probe in order to alleviate the demands on the propulsion system. Mass is the principal design driver for the interstellar probe, with all other requirements being secondary. The mass allocation of the probe is determined by the choice of the launch vehicle, the size of the propulsion stage which is used to insert the interstellar explorer into a close approach to the sun, and the size of the perihelion burn propulsion system. As stated previously, we are baselining a Delta III launch vehicle with a Star 48 third stage. Our concept for a perihelion burn propulsion system uses 260Kg, leaving 50Kg for the interstellar probe.

Within the constraint of this mass allocation, we must develop a probe which can perform significant science measurements and communicate these measurements to Earth. The probe must be able to produce power and continue to operate for a period exceeding 50 years. The probe must be extremely robust so that minimal failures occur and those failures that do occur must be either repairable or have their effect mitigated so that the mission is not jeopardized.

The major subsystems on a typical spacecraft are the following:

A. Structure
B. Propulsion
C. Power

D. Guidance and Attitude Control
E. Command and Data Handling
F. Communications
G. Thermal
H. Payload

All of these system are significantly different on the Interstellar Probe as compared to a standard spacecraft. We will discuss these differences in the following sections.

2.3 Structure

The probe mechanical design consists of three main mechanical elements – an ARPS assembly, a central support mast, and an optical dish. The ARPS is placed at one end of the mast in order to minimize the radiation dose to other spacecraft components. At the other end of the mast is the large optical dish which faces away from the direction of travel and back toward the solar system. Since this dish is used, for the most part, only as a transmission device, it is not necessary to surround it with a collimator, as would be required with a receiving optical communication system. Various optical system physical configurations are discussed in section 2.8.

Instruments and spacecraft electronics boxes are placed on the back of the optical dish and along the mast. Four booms, used for field measurements, are mounted orthogonal to one another at the perimeter of the optical dish. Since the spacecraft is a spinner the ratio of lateral to longitudinal moments of inertia of the probe is set to be roughly 1.25 to 1 for best stability. The spacecraft secondary battery and power control system are placed inside of the mast roughly at its midsection. The communication system laser beam is also placed inside the mast and points out the end of the mast, through a small hole in the one meter optical dish, toward the hyperboloid reflector. The mass and power allocation for the probe is shown in Table 2.3.1. A conceptual drawing of the probe is shown in Figure 2.3.1.

	Mass (kg)	**Power (W)**
Power system	10	-
Instruments	10	10
Structure	15	-
Communications	10	Intermittent, uses energy storage system
Spacecraft electronics	5	5
Totals	**50**	**15**

Table 2.3.1

The optical system would use composite materials and fabrication processes which are being developed for the Next Generation Space Telescope (NGST). This development is expected to reduce the mass of the primary mirror substantially from what was required on the Hubble Space Telescope. The probe’s mass allocation for a communication system which uses a one meter diameter mirror is based on the goals of the NGST program, which are expected to be realized within the next 10 years.

The mass allocation for the instrument payload is also very tight. The ability to fit a

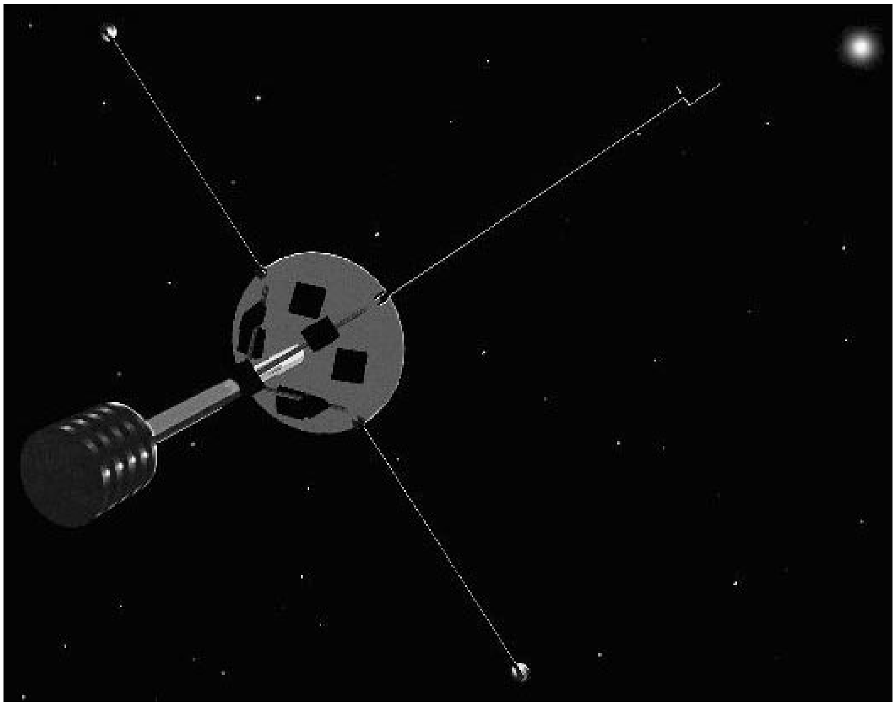


Figure 2.3.1

scientifically useful complement of instruments within this allocation is possible through the insertion of new technology and by the implementation of a novel system architecture. This is discussed in more detail in the payload section (2.10).

The central mast, also made of composite material, is a hollow cylinder with internal bracing for additional stiffness. The stiffness of the central mast is not critical for optical system pointing since the star tracker and sun seeker are directly coupled into the optical system and hence are immune to alignment issues involving the main structure of the probe. The mast is only required to be stiff enough to withstand loads induced by launch or by the propulsion system. Some of the spacecraft electronics are mounted inside of the mast while others are mounted to the outside of the mast, as well as on the back of the primary mirror. The mass allocation assumes a mast roughly 3 meters in length, 0.1 meter in diameter, with a wall thickness of approximately 1 centimeter. The optimum mast geometry has not been fully studied. With further study, a truss structure may ultimately prove to be a better selection for the probe's central support member.

The power system, which has a mass allocation of 10kg, consists of an ARPS, secondary battery cluster, and power regulation and switching electronics. It is discussed in more detail in the Power system section (2.5).

2.4 Propulsion

The 50 kg probe does not contain a propulsion system. Instead, the probe is held in a cocoon inside of the Interstellar Explorer's powerful perihelion burn propulsion system and is mechanically ejected by a stored energy system (spring) after the swingby of the sun is completed. The ejection increases the acceleration of the probe somewhat while slowing down the expended propulsion system. The probe coasts out of the solar system with the propulsion system body drifting behind at gradually greater and greater distances. The propulsion system does not interfere with communication back to the Earth due to a small asymmetric thrust applied to the propulsion system structure which causes it to slowly drift off of the trajectory of the probe.

2.5 Power

The choices for the primary power system for an interstellar probe are severely limited. Solar panels are not feasible because of the significant distance from the sun. For example, assuming a typical solar cell conversion efficiency of 20% and a solar input at 1000 AU of 1.35×10^{-7} Watts/cm^2, we would need a solar panel about 1Km square in order to obtain 1 Watt of spacecraft electrical power. While it may be feasible to design a probe that only consumes 1 Watt of power, the mass needed for the solar panels is prohibitive.

2.5.1 Primary Power – Battery Concept

A battery acting as the primary source of power is theoretically possible but there are significant practical limitations. Lithium batteries are undergoing significant development at this time and their energy densities have been doubling roughly every ten years. Assuming that this progress continues, we could anticipate a battery which could supply about 8000 Watt-hrs/Kg by the year 2030. If we allocate 10 Kg for the battery system on our probe, it would supply 80,000 Watt-hrs of energy. There are probe operational modes which could possibly support using only this much energy for a 50 year mission. One scenario involves having all instruments go into a dormant mode for regular intervals of roughly one week. A low power clock would mark the time and wake them up for a short period of data acquisition and data transmittal. For example, if we assume that they take data for about four hours while consuming a total of 5 watts and then the probe transmits this data to Earth for 10 minutes using 12 watts (see the Communication system section), then the probe would use roughly 22 Watt-hrs during this period of activity. Since there would be ~2500 periods of activity during the mission, the total energy consumption would be 2500 x 22 = 55,000 Watt-hrs, which is within the capability of our proposed battery. However, there are a number of other factors which make this difficult to achieve. First of all, it is unlikely that the doubling of energy density will increase indefinitely. Indeed, battery researchers believe that the rate of increase in energy density may already be tapering off due to the physics of the devices. Secondly, all batteries have a self discharge rate which is roughly 1% per year at room temperature. The discharge rate is much better at the cold temperatures where the probe would be operating (100-150K), but at that temperature the batteries barely produce any output. Hence, in order to use a battery as the primary power source, significant improvement in energy density and cold temperature operation would have to be achieved.

2.5.2 Primary Power – ARPS concept

The most obvious choice for a power system on a mission of this type is an Advanced Radioisotope Power System (ARPS). These have flown on many missions, such as Ulysses, Cassini, and Galileo, with an unblemished safety and reliability record. Existing designs supply about 4 electrical Watts per kilogram. The probe has a continual power consumption of about 15 Watts, so an ARPS weighing approximately 4 Kg is needed. Due to the radioactive output of the ARPS, it is necessary to mount it as far away from the other spacecraft components as possible. Hence, it is mounted at the far end of the probe's central support mast, at a distance of 3 meters. The ARPS also generates a significant amount of waste heat. On most spacecraft this would be a serious concern. However, on the Interstellar Probe the waste heat is needed to keep the spacecraft at a reasonable temperature. The use of this waste heat is discussed in more detail in the Thermal System section.

2.5.3 Secondary Power

The probe has been designed to draw a minimal amount of power during normal operations. The primary power system is capable of providing enough power for all nominal payload and spacecraft functions, including intermittent attitude adjustments and operation of the optical communication system. However, the primary power system does not provide a robust margin for operation of the thrusters and the communication system during the life of the mission. Although only a few watts of laser power are required to provide a positive communication link margin, it is prudent to be able to have sufficient power reserves in order to boost the signal in the event of emergencies or degradation of optical transmission properties. Also, during normal attitude adjustments to point the communication dish at the Earth receiver, very little power is required for the thrusters, but it is possible that significant thruster firing would be required in the event of an attitude anomaly.

Hence, a secondary power system has been baselined for this mission to supply power during transient peak loads for both the communication and attitude control systems. There are essentially two methods for long term storage of energy: a sealed rechargeable chemical battery or a bank of capacitors. Both systems are typically capable of providing significant amounts of power essentially instantaneously for brief periods. The operating scenario for use of the secondary power system is that it would be trickle charged by the primary system during the large fraction of time when the probe is neither communicating nor doing precision pointing. Power would be drawn from the system as needed during the regular (weekly) telemetry communication periods. The selection of an appropriate secondary power system is not straightforward for a mission of this duration. This is because chemical batteries have an energy density that is many thousands of times higher than capacitors, but there is no test data which has established that batteries have a lifetime exceeding fifty years. Rather than be forced to fly a capacitor bank, a scheme which maximizes the life of the battery system has been developed.

First of all, since battery dry storage life is much higher than wet storage life, the secondary system will consist of a number of small batteries stored in the dry state. The lifetime of a battery in the dry state is somewhat dependent on temperature. The main support beam of the probe has a steep temperature differential along its due to the ARPS being at one end and the radiative optical communications dish at the other end. The secondary battery would be placed inside the probe's central support beam at a location along the post so that its thermal environment was optimal for long life. Squibs (pyrotechnic devices) would be used to activate each battery in turn when the previously utilized battery is no longer holding its charge. For example, four small batteries, each used in the wet state for fifteen years, would easily suffice for the fifty year mission.

Secondly, the life of a wet battery is strongly determined by the number of discharge cycles and by the depth of discharge. In order to maximize battery life the depth of discharge will be kept to a minimum. The amount of energy that needs to be available from the active battery in order to provide significant margin for the communication and attitude control systems is 60 Watts for 10 minutes. The batteries types which are appropriate candidates for this mission are either Nickel Hydrogen or a variant of a Lithium battery, such as Lithium-Ion. Since a currently available Lithium Ion battery has a specific energy of approximately 200 Watt-hr/Kg, and we are assuming a 20% depth of discharge, a 250 gram battery would be sufficient. Since four batteries would be flown, the total mass allocated to secondary batteries is 1Kg.

2.5.4 Power Regulation and Distribution

The configuration of the complete power system would be fairly conventional, with the primary power bus tied directly to the ARPS and regulated using shunt regulators. The battery would be used for current needs beyond those supplied by the ARPS and boost regulation would be employed to boost the battery's voltage up to the specified bus voltage.

As mentioned previously, minimizing mass is the key factor governing the design of the Interstellar Probe. Since both the spacecraft harness and instrument/subsystem low voltage regulators can be quite massive, it is prudent to select a power distribution system which is as light as possible. One standard tradeoff involves the number of regulated voltages generated by the main regulator, located proximal to the ARPS, versus the number of distributed regulators in each of the subsystems. In many modern spacecraft the main power bus is a single voltage (e.g. 28 volts) and each instrument or subsystem has their own power supply regulator which is used to generate a myriad of voltages (e.g. ± 5, ± 6, ± 12, ± 15) that are unique to the particular instrument. This clearly minimizes the mass of the harness but the combined mass of the regulators is substantial. When the main spacecraft regulator develops all the required low voltages itself, then the harness becomes heavy.

A unique and low mass solution for the Interstellar Probe is facilitated by the use of extremely low voltage inter-communication ASICS (referred to as "SAC"s when discussed in more detail in section 2.7) and processor modules. Redundant inter-communication ASICS are connected to each instrument and spacecraft subsystem and are used to collect relatively raw data from the various sensors (e.g. particle detectors, start trackers, etc). They transmit the data at high rates via wireless technology to a cluster of spacecraft processors. These units all operate at the same voltage which is anticipated to be less than 1.8 volts. Because the inter-communication ASICS and processor modules have been designed to consume very little power, it is feasible to satisfy their power needs by using switched capacitor regulators. Switched capacitor regulators are finding increasing use in the computer industry and the fabrication of these devices has trended toward using Silicon on Sapphire (SOS) radiation hard technology. These regulators provide very robust and very lightweight regulation, but they do not provide isolation. However, on a small, low power spacecraft such as the Interstellar Probe, isolation between primary and secondary sides may not be necessary.

Of course, each instrument and subsystem will still require some higher voltages for various purposes, such as specialized signal conditioning, detector bias voltages or high voltage for microchannel plates. The main bus voltage would be selected so as to be at the voltage for which the current drain is the highest. For example, if a majority of the subsystems use +5 volts, then the bus would be selected to run at +5 volts. The switched capacitor regulators are also quite suitable for generating both plus and minus voltages which are multiples of the bus voltage, such as ±10 and ±15 volts. The +5 volt main bus would also be used as the primary voltage for high voltage power supplies which use a standard voltage multiplier (Cockcroft-Walton) topology to generate voltages exceeding 100 volts. In order to save mass, it is fairly critical that instrument and spacecraft subsystem designers design their electronics and detectors to use as few voltages as possible.

In summary, the pervasive use of distributed low-power processor and inter-communication modules, coupled with compact switched capacitor regulators and an instrument design philosophy that minimizes the number of unique voltages, allows the spacecraft bus to be at a single voltage (e.g. +5 V) with minimal mass used for regulators in any of the instruments or spacecraft subsystems.

2.6 Guidance and Attitude Control

The conceptual design of an attitude determination and control system for the proposed Interstellar Probe vehicle is straightforward, and perhaps even simple. We need only point the antenna boresight at a receiver on Earth once a day to within the beamwidth of the laser signal, estimated to be 0.15 arcsecond. This corresponds to an absolute pointing and control requirement of ±0.075 arcsecond (3-σ). It is at this point that reality and engineering begin to become problems. In order to realistically assess the possibility of meeting such a tight requirement, a detailed and thoughtful analysis of error sources is required. A number of considerations go into this error analysis, including:

A. *Trajectory Knowledge*. It is necessary to have onboard clock and ephemerides to track the receiving station and vehicle so that the vehicle-to-station vector orientation is known sufficiently accurately to effect data transfer. Denote the pointing angle uncertainty by σ_{TRAJ}.
B. *Transmitter Boresight Pointing Knowledge*. There are several factors contributing to the definition of boresight pointing knowledge. First, there is knowledge of a fiducial frame, which we would expect to be defined by a high precision star tracker. Denote the absolute pointing knowledge of the tracker by σ_{STCAM}. Then we must know the alignment of the star tracker relative to the laser antenna. Let the uncertainty in this alignment (projected on the pointing axis) be $\sigma_{FIDUCIAL}$.
C. *Boresight Control***.** In order to minimize complexity, mass and power requirements, the vehicle is proposed to be stabilized as slowly rotating about the axis of the laser transmitter antenna. Preliminary mass properties estimates indicate that a very favorable inertia ratio can be easily achieved, thus minimizing the effect of disturbance torques (solar radiation, for example) on the vehicle and, therefore, complexity of the attitude control system and attitude fuel requirements. Then the function of the attitude control actuators is to move the spacecraft angular momentum vector to follow the receiving station. An immediately perceived difficulty here is in aligning the vehicle principal inertia axis to the antenna boresight and maintaining co-alignment over time. Even small changes in inertia, for example due to propellant expenditures, can move the principal axis from the required position in the vehicle and potentially require unfeasible active control. Denote the total pointing error from these control-related sources by $\sigma_{CONTROL}$.

Then it is immediate that we must have σ_{TOTAL} # (0.075 arcsec)/3 = 0.025 arcsec, where

$$\sigma^2_{TOTAL} = \sigma^2_{TRAJ} + \sigma^2_{STCAM} + \sigma^2_{FIDUCIAL} + \sigma^2_{CONTROL} \quad .$$

Achieving such accuracy for all of these errors appears daunting, but perhaps not impossible in the intermediate term. Addressing them individually:

2.6.1 Trajectory and Clock Errors

Because of the very long light times involved, it is perceived that Interstellar Probe would necessarily operate open loop, with virtually no uploads. In particular, we presume the necessity to integrate all trajectories on-board to sufficient accuracy to know where to point. For example, at 100 AU after 5 years, say, pointing to 0.01 arcsec requires that the relative spacecraft-to-Earth vector be known to approximately 750 km. in the computational coordinate system. This is about 0.05 ppm, and seems reasonable computationally.

2.6.2 Transmitter Boresight Pointing Knowledge

As noted above, the two immediate contributors to knowledge are measurement and alignment of the measuring device. First, consider the star tracker. Commercially available trackers which measure to better than an arcsecond are readily available today, and wide field-of-view trackers with accuracy on the order of 0.1 arcsecond are in final development. Thus it appears that meeting Interstellar Probe's knowledge from a tracking point of view is achievable in the foreseeable future. The show-stopping problem here is that current development efforts do not have the mass and power constraints of IP. This is a clear need for development. There is also a legitimate need for a star catalog with accuracy commensurate with the desired pointing. We note that IP's dynamics are so passive that tracker integration time should pose no problem.

The other error source here is alignment of the transmitter boresight to the fiducial frame defined by the star tracker reference. This error source can be essentially eliminated by using the same optical system for the tracker and the transmitter. While calibration errors cannot be completely eliminated, this procedure should allow that they be measured and compensated on orbit.

2.6.3 Boresight Control

Boresight control may be the most technically challenging part of the IP attitude system because of perturbations, control quantization and fuel requirements, to begin. As noted above, we propose designing the vehicle to rotate slowly about the nominal transmitter boresight. This has the clear advantage of tending to cancel disturbance torques, the major one appearing to be solar pressure acting through a center-of-pressure to center-of-mass offset. Since early on, the vehicle will be required to execute significant attitude maneuvers to track Earth (90°/6 months at 1 AU from Earth), this disturbance cannot easily be ignored. It is fortunate that both angular motion required decreases as 1/r from Earth, and the solar pressure decreases as $1/r^2$ from the sun. There is a clear trade between higher rotation rate for disturbance cancellation and fuel use to move more angular momentum while tracking.

Initial alignment and stability of the physical principal axes of the vehicle has a clear effect on the actual rotation of the IP, and must be carefully controlled. At least one concern is attitude thruster depletion causing inertia changes. In principle, this will be handled by thrusting only in symmetric couples.

Thrusters are envisioned as three matched pairs, which can execute full three axis attitude control via sector firings. One possible implementation is via pulsed plasma Teflon devices, which are currently available in some forms. They have the advantage of relatively high I_{sp} (1000 sec. or more is available in current designs) and low minimum impulse bit (50 μN-s thrusters are currently available). They have the disadvantages of relatively high mass and deposition of Teflon from the plasma dispersion (which should be mitigated by properly shielding the star tracker and other affected instruments from the spherical propagation). Other thruster implementations which may be feasible in the longer term but are currently under study include, for example, Field Emission Electric Propulsion (FEEP), where sub-μN thrust levels, very high specific impulse levels (I_{sp} = 6000-1000 sec.), and extremely small minimum impulse bits (≈1.e-8 Ns or less) are discussed.

2.7 Command and Data Handling

The IP does not have a subsystem on board that can be strictly identified as a "command and data handling system." Indeed, although it has a number of powerful processor modules which do perform command and data handling functions, these modules are not aligned or connected to any particular subsystem or task. In other words, all computing power and the processing that is done on them is distributed and all tasks are only loosely coupled with specific hardware. By "distributed" we do not mean it in the sense that is commonly used today to describe an array of processors which are handling a single difficult task in order to complete the task more quickly. In some ways the IP implementation of a distributed system is the inverse of that. In the IP architecture all the identical processor modules are extremely powerful (each exceeding 200 MIPS performance) so that any processor is capable of handling ALL the tasks on-board single-handedly (e.g. attitude control, instrument processing, etc.) if that were to become necessary. However, in general, a processor module handles the single task it has been assigned, but this situation is very fluid and flexible.

The interstellar probe architecture has the following features:

A. A cluster of processor modules, not necessarily co-located, exist on the probe. Each is extremely powerful yet has minimal power consumption. Each module contains a central CPU, memory, and a wireless communication module. These types of processors are being developed for today's cellular phone market. Current versions have 50 MIPS capability and only consume 18 milliwatts. There is a significant development underway to produce this type of processor for space use. These highly advanced processors, which operate best at extremely low temperatures, are discussed in later this section.
B. All spacecraft subsystems, including instruments, do not have any processors. Instead, they have a high rate wireless communication module which is used to broadcast data at their assigned frequency.
C. All the subsystems are broadcasting their data so that any one or all of the processor modules can listen. Based on how the processor modules have decided among themselves

to configure the tasks, the processors modules will tune their receiver systems to the correct subsystem frequencies and collect data from one or more subsystems. In general, one subsystem (e.g. power management) will be assigned to a single processor module. There are more processors modules than subsystems, so some of the modules are turned off for portions of the mission. If a processor fails, its task will be shifted to a unused processor. If no spare processors are available, then some of the remaining processors will have to handle more than one task. In the worst case scenario, when only one processor is still working, it alone can handle all the tasks of the Interstellar Probe. If failure of the individual processors is due to radiation damage, then sometimes turning them off for a period of time allows them to anneal. The remaining processor, as well as the low level boot module, will try to revive previously failed modules if at all possible. The entire autonomous operating principles are described in more detail below.

A block diagram of the probe, including the major subsystems, is shown in figure 2.7.1.

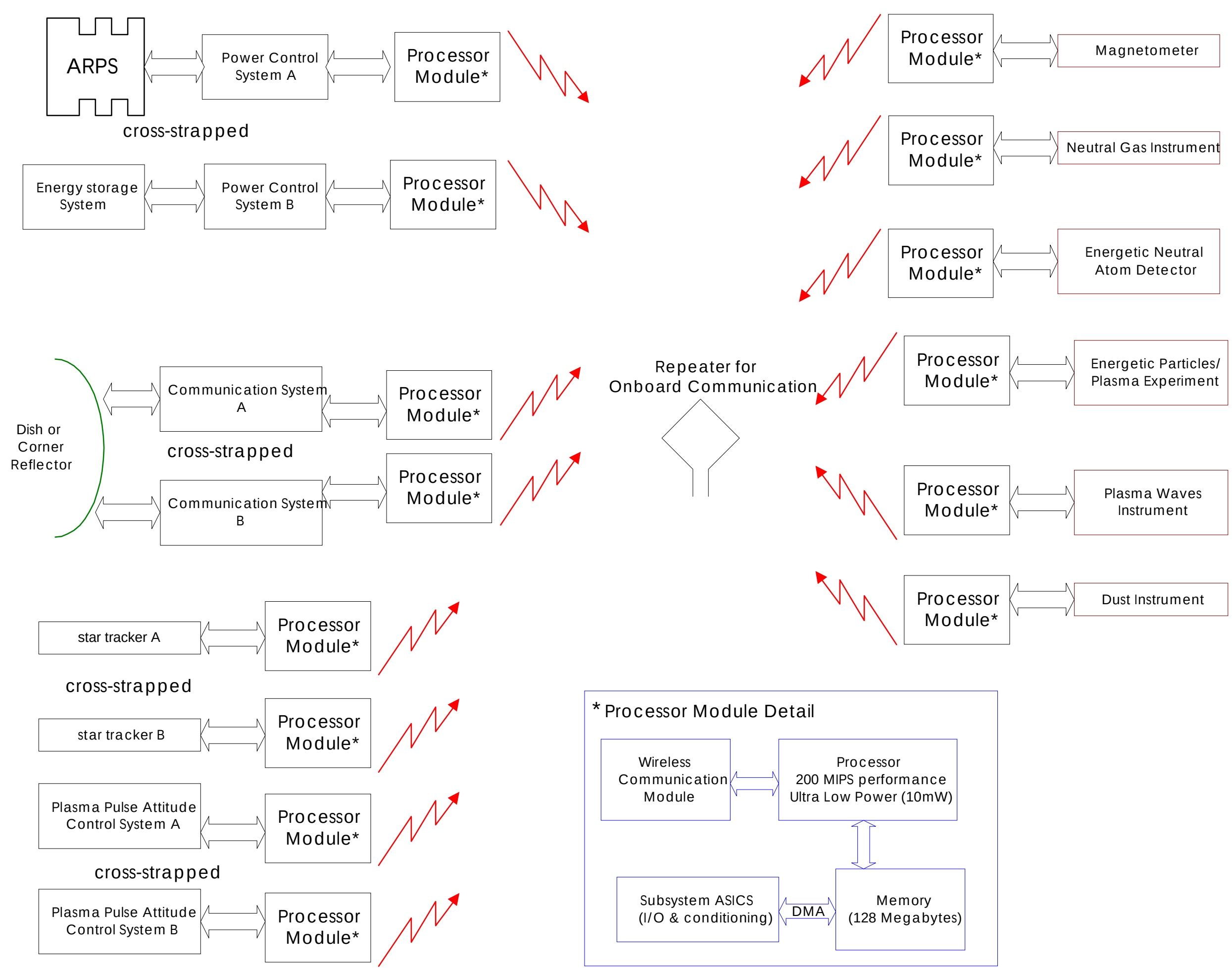


Figure 2.7.1

2.7.1 Low Power Electronics

The large and lucrative portable appliances market will drive the semiconductor industry to make significant improvements to CMOS technology over the next ten years that will also benefit the Interstellar Probe. Fabrication processes developed to minimize operating voltage will be especially beneficial for building application specific integrated circuits (ASICs) for the IP. The new manufacturing technology will likely have excellent total dose and single event latchup radiation tolerance. However, using mass-market catalog parts on the probe will be more difficult. Chips developed for commercial applications will, with rare exceptions, have no provisions for detecting and responding to radiation induced soft errors. Error correcting techniques can overcome single event errors for large memory arrays but solving this problem for logic devices is best accomplished within a chip rather than through external mechanisms. Hence, most Interstellar Probe logic will be contained within ASICs that use internal circuits immune to radiation induced soft errors.

Dramatic reductions in CMOS power consumption appear feasible by lowering voltage. Switching power, P, of CMOS circuits is given by $P = \alpha C V^2 f$ where α is the circuit activity factor, C is the total circuit load capacitance, V is the transition voltage, and f is the operating frequency. However, as voltage is lowered, the gate potential of each off transistor approaches that device's threshold, V_t. Since off current is exponentially related to the difference between gate voltage and V_t, operating logic circuits with a low power supply also leads to increased static leakage current. Burr, et. al. have shown that total energy is minimized when switching energy and leakage energy are equal. Burr has also shown that the energy-time product is optimized when $V = 3V_t$ and that V_t is proportional to the thermal voltage (26mV @ 300°K, 6.7mV @ 77°K). Hence at Interstellar Probe's very low operating temperatures, a power supply as low as a few tens of millivolts is feasible and will result in over a three order of magnitude power savings compared to current technology.

Power can also be reduced, although less dramatically, through careful attention to the remaining factors in the power equation. Architecture and logic design can be optimized to minimize signal transitions, i.e. α, the activity factor. Parallel versus serial architecture trades that consider transition power as well as area and speed are possible. Output glitches can be reduced or eliminated by using hazard free logic design. Signal evaluation order can be rearranged to reduce intermediate transitions. Computer Aided Engineering tools to support power optimized design are now commercially available and will be in widespread use when detailed design of the Interstellar Probe electronics commences. Load capacitance can be reduced by careful attention to device sizing and minimizing wire length although full custom or standard cell design methodologies are needed to achieve this benefit. Insulating substrate technology can also reduce load capacitance although that technology may be incompatible with extreme reductions in power supply voltage.

Extreme voltage scaling as described above is not without difficulty. Controlling device threshold variations across a die is difficult especially to the values required for subvolt power supplies. One demonstrated solution to this problem controls V_t by building devices with near zero or even negative threshold voltage and then adjusting substrate potential to increase the threshold to the desired value. Work now in progress will use a feedback network to generate and control the substrate bias voltage on-chip. The bias circuit will also adjust substrate potential to compensate for temperature and supply variations. Threshold uniformity will still limit the size of the largest chip built using these principles but devices with over 100,000 transistors have already been successfully demonstrated.

Larger designs can be built using more complex fabrication processes. Using separate P and N wells will allow device characteristics of both FET types to be independently adjusted. If the well regions extend over sufficiently small die regions than threshold voltage will be sufficiently uniform to support very low voltage operation. To ensure adequate noise margin, communications between uniform threshold regions may require special provisions. Potential solutions could include differential signaling or logic constructed with higher threshold devices.

Extreme low voltage CMOS faces additional challenges. Signal swings are very low and may be sensitive to electrical noise from many sources. Careful physical layout and construction of circuit cards can lessen these effects but a practical electronics system functioning with a 50mV power supply at cryogenic temperatures has not yet been demonstrated. Power efficient interfaces between low voltage CMOS and traditional logic must also be developed. These interface circuits should ideally be integrated on low power parts to ease system design and minimize board area. Power conversion circuits compatible with low power requirements must be developed. These circuits must efficiently convert spacecraft power bus voltage to the final application value. Noise immunity and tolerance to input variations must also be acceptable. While these challenges are formidable, they can be solved by the time Interstellar Probe starts detailed design.

2.7.2 Autonomous Operation

The architecture for the Interstellar Probe system is logically similar to familiar spacecraft architectures: a spacecraft system consisting of various subsystems which have individual areas of responsibility - an attitude control subsystem, a navigation subsystem, a power subsystem, a thermal subsystem, various instrument subsystems, etc. Physically the system is implemented differently. Each subsystem consists of two major parts; the first is a set of two or more Sensor/Actuator/Communication (SAC) modules and the second is the Control Logic & Communication (CLC) processes that control the subsystem. The CLC processes are mapped onto a set of physical processors, all of which are identical and any of which can communicate with a SAC. Any processor can do any other processor’s jobs. There are many more processors then are needed to do the job. It is expected that processors will fail, sometimes completely and sometimes only partially. The architecture works with a “quorum” of computers. Others are left powered down, are duty cycled, and are available if needed.

2.7.2.1 Autonomy/Fault Tolerance/Safing

The approach to autonomy, fault tolerance, and safing is hierarchical. In this discussion autonomy refers to true decision making on the part of the system or subsystem; fault tolerance refers to the ability to detect and recover from a fault condition; and safing is the act of shutting down and passing fault conditions on to a higher authority. A subsystem can pass a safe or fault condition to a higher authority. Each level of the architecture is responsible for its own autonomy, fault tolerance, and safing.

Commanding of each subsystem is accomplished using goals. A goal is expressed as “maintaining a defined system state for a defined period”. Goals have priorities and a subsystem is responsible for fulfilling its goals in priority order. If the system cannot achieve the goal or has an uncorrectable fault condition that prevents it from achieving the goal, then the information is fed back to the requesting level.

In the hierarchical approach each system attempts to solve its own problems. The system monitors itself to determine if it is having problems. If a problem is discovered, corrective action is taken. If the corrective action is successful, the upper level system is not notified of the problem. If the corrective action is not successful then the problem is passed to the upper system for possible corrective action. For the most part, corrective action would consist of gross fixes such as power cycling. If the problem cannot be corrected but the system is in a partially useful state then the upper level system uses that information when planning an activity. In the case of a severe fault, the system would safe itself and the upper level system would be made aware.

A level 0 subsystem has responsibility for only a limited goal such as the maintenance of a switch position. Even a system this simple would have responsibility for monitoring of its state and fault recovery. A level 1 system would consist of multiple level 0 systems. This system would have a goal or a small set of goals that could be expressed as goals for the level 0 systems. The level 1 system would be responsible for monitoring and fault recovery for its limited set of goals.

The behavior of all systems is expected to change significantly over time and thus adaptive monitoring methods will be needed. Neural nets will take as inputs various measurements and determine the state of the system. A confirmation or non-confirmation of the state will be used to adjust the weights of the neural net. Fault recovery involves reasoning about the current state and the state history and using that information to pick a predefined fault recovery method.

The processor quorum (or decentralized “brain” of the probe) has the responsibility for trying to do gross fault recovery on the subsystems that have faulted or safed themselves. It also has the responsibility to plan how to perform the mission goals given the resources available, i.e. planning. If the spacecraft system cannot perform the tasks given the on-board resources then it needs to contact Earth for further instructions or begin a shutdown/restart sequence. In the event that subsystems have failed and IP fault recovery has failed and the IP is in immediate danger the spacecraft will go into a safe mode. This mode protects the spacecraft, attempts to contact earth, has the ability to dump diagnostics and perform gross commands. This mode is handled by a separate system developed as a finite-state machine, whose goals are to safe the spacecraft, contact earth, and await further instructions. This system also has the responsibility for beginning the shutdown/restart process. This restart process would begin by finding power and a healthy processor. The processor would be started and complete the restart process. It is expected that the intelligence on-board the spacecraft will have either corrected fault conditions on board or reported problems that indicated the spacecraft would not be able to achieve the mission goals long before the initiation of a safe mode. IP safe mode and the system that supports it is an anomalous condition and it is not expected that IP will enter it.

During the early phase of the mission when commands from the ground are accepted, humans at the ground station are still the final arbiter. Diagnostic information is gathered and (while the spacecraft is safe) analyzed. Once a possible solution is determined, commands are sent to the spacecraft to start to bring it back to a recovery mode and if possible, operational mode. The option to go to the ground for support will only be available while the spacecraft is well within the Solar System. Once outside of uplink range or if no communications is received from earth, the spacecraft safing will go into a periodic shutdown/restart mode whereby the spacecraft attempts to build a quorum.

To illustrate, an instrument subsystem is responsible for collecting and analyzing data. Its goal is to collect and analyze data. In order to do that it needs to have heat, power, data storage, and computing power. These are passed as goals to other subsystems to achieve. Ultimately there will be a failure of some portion of the system. The subsystem whose goal was to provide a certain temperature will attempt to do that by cycling switches or other means. If it can maintain the temperature, the instrument subsystem remains unaware that there was a problem. If it cannot provide the temperature, it will notify the instrument subsystem that it cannot achieve its goal. The fault recovery software for the instrument would begin to try to solve the problem. This may be done by deciding to run cold or by running an adjacent heater at a higher temperature. The instrument subsystem may switch to an alternate SAC to provide the data. Whatever the problem, if the instrument subsystem is capable of meeting its goals then the spacecraft fault recovery processes are unaware of the problem. If the instrument subsystem cannot achieve its goals then the fault condition is passed to the spacecraft. If whatever spacecraft fault recovery that is available is unsuccessful then this information is passed on for autonomous re-planning based on the lack of an instrument subsystem.

2.7.2.2 Sensor/Actuator/Communication (SAC) Module

The mechanical portion of a subsystem is defined as a SAC module. The module has the capability of receiving and acting upon simple, primitive commands. Any complex operation is handled by processes running on the quorum and is implemented as a series of primitive commands passed to the SAC. The module has the capability of sending standard data messages. Any data manipulation, repackaging, or analysis is handled by processes running on the quorum. Communications is done using a broadcast mechanism. A SAC broadcasts a data message, all processors in the quorum can receive the message, and the process that is responsible acts on the message. When a process wants to send a command to a SAC it broadcasts a command message, all SACs receive the message, and the intended SAC acts on the command. Redundant subsystems each have their own SAC.

The combination of a SAC module and the CLC processes form a complete subsystem. A normal spacecraft architecture would have block redundancy of critical subsystems with cross-strapping capability. This architecture is inherently cross-strapped. A part of redundancy is accomplished by using quorum computing. The second part of redundancy will be achieved by using multiple identical SAC modules for a subsystem. The exact number of SAC modules used in a subsystem is determined by reliability and criticality considerations.

2.7.2.3 Processes versus Processors

The architecture uses multiple identical processors all of whom can communicate with each other and with each of the SAC modules. The number of processes is determined by two calculations. The first is the number of processors that will be needed to comfortably handle the computational load. This is called a quorum. The second is a reliability calculation. Processors will fail throughout the mission. The number of processors on the spacecraft will be such that there is a reasonable probability that there will be a quorum at the end of the mission.

Ideally the quorum will consist of more then the minimum number of processors needed to do the job but the architecture will provide for the system to gracefully degrade to that condition. Besides the normal processes that are running there are several additional processes. There is a controlling process and it is running on the controlling processor. This system maintains memory and process synchronization where it is necessary. The controlling processor is also watching all other processes running in the system and will detect problems, restart processes, migrate processes, start new processes or new processors. There are two processes that watch the controlling processor and processes. They watch to make sure that the controlling process is working as desired. They also participate in critical decisions. If there is disagreement then the majority will rule. The odd man is declared suspect and fault recovery is begun. It is assumed that the two watchers and the controlling process are running on different processors. This is ideal but not necessary.

Almost all tasks onboard the probe are not time critical. This fact will be used for process migration. Each process will maintain a checkpoint, which is the point at which it could start processing again. If a process needs to be restarted or migrated, it is simply stopped and restarted at the checkpoint. If a processor begins to fault then the processes running on that processor need to be moved to a another processor.

In the case of a failing process the controlling processor and the watchers will have determined that the process is failing. The controlling process will move to stop the process and remove it from the system. It will attempt to restart the process on same system. If that fails, then the process will be restarted on an alternate system. If more processors are needed another processor is started.

If the process that is determined to be failing is the controlling process then all processing requiring synchronization is halted until a new controlling process is started. In the case of a failing processor, all processes are migrated as above with the exception that no attempt is made to restart them on the same processor.

Each of the processors on the IP are either healthy, sick, or a zombie. A healthy processor can perform any task. A sick processor is a processor that cannot perform some tasks. This could happen if it has lost some I/O channel or memory capability. A sick processor will not be used if it is not necessary. A zombie processor is a processor that has failed. A failing processor will be turned off, making it a zombie. Periodically, as part of system diagnostics, the processor will be turned on to see if can restart. If it does, processor diagnostics are run, and if it passes it is declared healthy. If it fails processor diagnostics it is determine to be sick or a zombie based on the results. If it fails to restart then the time period for the next restart test is length is doubled. For example, the first failure will cause a restart test to be run in 1 week, if that fails then the next test will be 2 weeks, then 4 weeks, etc. If there is not enough processors to build a quorum to sustain critical processes then all of the zombies will be retested.

Each process and processor is responsible for sending health packets. Watchers are looking for the value or the non-existence of health packets to determine the health of the system. Critical IP system processes are also watched using special software monitors. These monitors duplicate the analysis performed by the processes using simpler, more robust, less accurate calculations. If the answers that the critical process gives diverge too much from the monitor, the calculation is called into question and a fault is declared. The monitor is adaptive - if it declared a fault was wrong, the parameters will be widened. If it declared a fault and was right the parameters will be tightened.

A processor that is not needed is left powered down. Periodically it is powered up to check its status. Diagnostics are run and if it passes it is declared healthy and is eligible to join a quorum should it become necessary. Processors are also power cycled. A healthy, running processor is periodically selected for power cycling. A new, healthy processor is started and the processes running on the old processor are migrated to the new processor. The old processor is then powered down. It remains eligible for joining a quorum.

2.8 Communications

A number of communications link methods are being considered for use on the Interstellar Probe Mission which is to be launched within three decades and travel for more than fifty years to at least 1000 Astronomical Units (AU), well outside of our solar system. The spacecraft's characteristics, especially its communication system, have been strongly influenced by the large distances and delta V (~20 AU/year = 95 km/s) involved in the mission. For these reasons conservation of weight and spacecraft autonomy are extremely important. At a range of 100 AU (1 AU=8.3 light minutes) electromagnetic waves (light or microwaves) will take 13.9 hours to travel from the spacecraft to Earth. Interactive control of the spacecraft becomes nearly impossible at these distances necessitating a highly autonomous craft and one-way communications to Earth. The main characteristics of the communications system are, therefore, that it needs to be low mass, low power and capable of one-way transmission at sufficiently high transmission rates.

We have considered many communications schemes which, in general, fall into the categories of microwave and optical. Microwave communication techniques have a long and successful history involving space missions. The shift of these systems to higher frequency carriers, first to X-band (8.4 GHz) and now to Ka-band (32 GHz), has improved microwave communications link directivity and bandwidth. There is also a large infrastructure to support the use of these frequencies such as existing ground-based arrays.

Optical communications links (unguided) are a relatively new approach in the space industry. Optical terminals tend to be much smaller in size and have much higher gain than their microwave counterparts. Satellite to satellite, satellite to ground, and satellite to under-sea links have been tested on occasion (particularly by the military) since the mid 1970's. The interest in optical systems was typically driven by the extremely high directivity, minimizing hostile interception of the signal, and high bandwidth, allowing high data rates. However, these early laser systems tended to be power hungry and heavily affected by atmospheric conditions. Even the most optimistic studies using multiple, geographically spread optical up-link sites concede the possibility that bad weather could render communications with the spacecraft impossible.

Based on *today's* space qualified technology an optical link to a space probe in not realistic. However, using new technologies, all of which are presently in the early stages of development and many of which are driven by commercial applications, several optical link schemes can be designed and built in a timely fashion for the Interstellar Probe. Developments such as commercially available, high power, high (electrical) efficiency laser diodes are of high importance. In the near future Quantum Cascade (QC) lasers will allow high output powers, less temperature sensitivity, and narrower linewidths. These devices offer the potential of using coherent, homodyne, detection of optical signals from spacecraft, dramatically improving the signal to noise margin as compared to video (direct) detection.

2.8.1 Microwave System

A microwave communication system has many advantages relative to an optical system. Assets currently exist at the DSN for supporting telemetry links as high as Ka-band (32 GHz). In addition, the required spacecraft technologies are relatively mature, lightweight, and power efficient. For example, transponders that operate at X and Ka-band exist and are continually being improved to reduce their mass and power. Solid state and travelling wave tube (TWT) technologies are also readily available at both X and Ka-band.

The primary disadvantage of the microwave approach is its limited bit rate capability when compared to optical systems of modest aperture size. For our Interstellar Probe overall system design, the mass allocation for the entire communication system, including the antenna, is only 10 kilograms. As will be seen, an appropriately sized microwave antenna is at least 60 kg, so a microwave system acting as the primary communication system does not fit in well with the overall probe concept. However, if propulsion advances were to be achieved that allowed the probe to be larger, then perhaps a large microwave antenna system could be accommodated.

2.8.1.1 Microwave System Design

For the purpose of a future Interstellar Probe, the use of Ka-band for a microwave communication system is preferred because of the increased performance it offers over lower frequency bands. Technologies in this frequency band are currently a focus of future product developments, both within NASA and in the commercial world.

For this study, we chose a microwave system that incorporates a combination of existing, near-term, and far-term technology developments. For example, the power amplifier is assumed to be a 10-watt unit similar to that flown on Cassini. The transponder is assumed to be the Space Transponding Modem (STM), currently in development and planned for operational use in the 2003 timeframe. The antenna is a longer-term development item. We have assumed a 14-meter inflatable dish similar to the one manufactured by L'Garde Inc. and flown as an experiment on STS-77.

We have assumed a redundant communication system consisting of two STMs, two 10-watt TWT amplifiers, a 14-meter inflatable antenna, and associated switching and filtering equipment. The total system weight of 71 kg is dominated by the antenna, which weighs 60 kg.

2.8.1.2 Microwave Link Analysis

A link analysis of the Ka-band communication system was performed to determine its downlink bit rate capability. A 34-meter beam waveguide ground antenna was assumed, along with the use of Turbo coding. An 80% weather model was used, meaning that the link will have the predicted capability at least 80% of the time.

The results of the analysis show that a 20 bps downlink capability exists at an Earth distance of 1000 AU. For distances much further than this, the capability drops rapidly because of carrier and bit synchronization issues with the ground equipment. At distances >>1000 AU, an optical communication system will be required to achieve a reasonable downlink capability.

2.8.2 Optical System

The development of an optical communications system for the Interstellar Probe to be launched within three decades is very realistic. Using a variety of technologies it is possible to construct a system that can communicate to Earth from distance exceeding 1000 AU. While some of these technologies already exist at commercial and "space ready" levels others are still in early research and development. However, there are no gaps in technology and no missing links that would block the overall development.

2.8.2.1 Basic Optical Communications System Design

The large round-trip light travel time (~28 hours at 100 AU and ~12 days at 1000 AU) mandates that the spacecraft have a high degree of autonomy. While the spacecraft will have the ability to transmit and receive optical communications the reception of signals from Earth is intended for use at close range, for a relatively brief time after the perihelion burn. This ability will allow controllers to conduct “wellness” test on the probe after launch. Once the spacecraft has left the inner solar system communications will predominately concentrate on the one-way, Earth bound, link.

The location of the Earth terminal is also of great interest. In the mission’s initial phase, while the probe is in the inner solar system and communications are bi-directional, the Earth terminal can be located on the ground. However, as the probe’s distance increases, the uncertainty and attenuation of the Earth’s atmosphere will become the limiting factor in the performance of the optical link. This stumbling block can be remove by simply placing the receiving terminal in Earth orbit. This orbiting relay station may either be a new platform or a decommissioned and retrofitted Hubble Space Telescope (first or second generation). The signal can then be cued in memory and down-linked to Earth via a microwave link.

The mission's long duration also translates into power limitations. As a result, the communications system will be expected to function with high power efficiency. It is this combination of low mass and high power efficiency that has in the past limited the applications of laser communication systems in space. Modern laser diode technology, driven predominantly by commercial applications, has removed these limitations.

The basic optical terminal to be carried by the Interstellar Probe is shown in Figure 2.8.1. The front end of the system is a low mass Cassegrain Telescope (without chromatic correction). The system is kept as simple as possible with a minimum of optical components. This particular configuration shows a baffled telescope which minimizes stray light striking the primary mirror. This feature is necessary to allow the optics to receive signals from the Earth-terminal. On a downlink-only system, this cylindrical element could be eliminated.

The system functions at two wavelengths. The transmit wavelength $\lambda 1$, characteristic of the laser diode on the probe, and the receive wavelength $\lambda 2$, characteristic of the laser system on the Earth-end terminal. A dichroic beamsplitter just behind the telescope separates the transmit and receive signals.

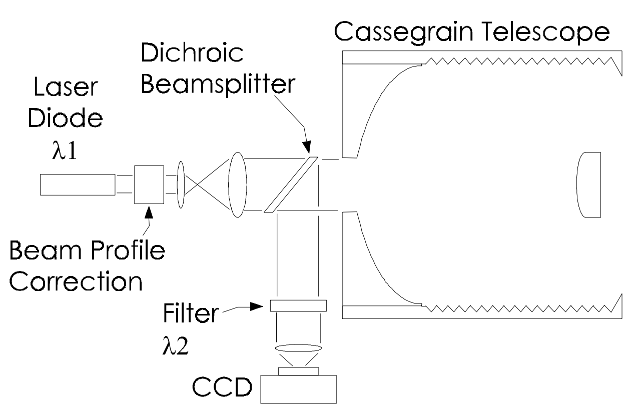


Figure 2.8.1. Interstellar Probe optics

The transmit signal beam is initiated by a modulated, high power laser diode. Beam correction optics are used to convert the laser diode's oval beam cross section to a circular one. Lenses are used to expand the beam and project it through the dichroic beamsplitter where it is coupled to the rear of the telescope for transmission to Earth.

The received signal passes though the telescope and is reflected off of the dichroic beamsplitter. A narrow bandpass filter, at wavelength λ2, is used for spectral isolation of the Earth based signal. Finally, a lens forms the received signal image on a charge-coupled device (CCD). This CCD is used as a signal detector while the spacecraft is in the inner solar system and receiving signals from Earth. More importantly, the CCD serves as a Sun tracker which is key to locating Earth and pointing the telescope.

The optics that make up the Earth-terminal are not limited by either mass or power constraints. By comparison to the probe's nuclear thermoelectric generators (ARPS) the Earth-terminal's solar panels will supply nearly limitless power. While the Earth-terminal is basically a relay satellite in low Earth orbit (LEO) it is important to remember that its proximity to Earth means that it can be upgraded at any time.

The optics (Figure 2.8.2) in the Earth-terminal will basically resemble those in the probe. However, the laser diode will likely be replaced with a more conventional laser system and external modulator with high output powers, though worse power efficiency.

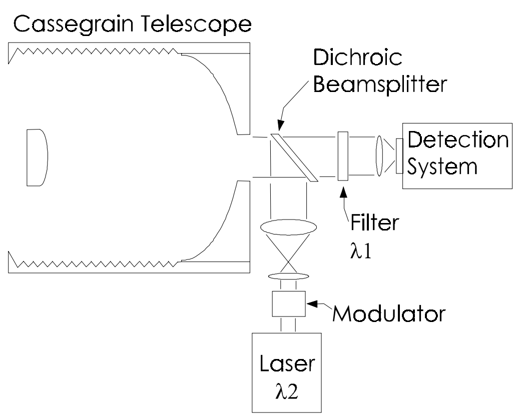


Figure 2.8.2. The basic optics in the Earth-terminal.

Another variation between the two terminals is the "detection system". The contents of the "detection system" will depend on whether incoherent or coherent optical detection is used. An incoherent detection system uses a detector, typically a photo-multiplier tube (pmt), to directly detect the optical signal power. A coherent detection system is much more complicated using a "local oscillator" and signal mixing at a photo-diode (pin) which acts as a gain mechanism. Coherent detection has much better signal to noise ratio, typically 10 to 100 times depending on background noise, and can detect much weaker signals.

2.8.2.2 Optical System Requirements

Table 2.8.1 shows the communications system requirements for the Interstellar Probe.

Table 2.8.1. Communications System Requirements.

Requirement	Value
Data Rate	500 bps
Bit Error Rate	10^{-9} (encoding dependent)
Range (Probe)	100 to 1000 AU
Range (Earth Terminal)	High Earth Orbit
Probe Tracking	Sun Tracker & Star Camera
Pointing Accuracy	0.15 arcsec (0.72 μ radians)
Electrical Power Load	10 watts (continuous)
Mass	10 Kg
Lifetime	>50 years
Reliability	95%
Transmission Redundancy	Probe Highly Autonomous. Transmissions Labeled w/ Header & repeated X times at predetermined intervals.

There are many possible variations in design that would meets the communications requirements expressed Table 2.8.1. One possibility which meets the requirements specified above and uses probe and Earth terminals of the form shown in Figures 2.8.1 and 2.8.2 is characterized in Table 2.8.2.

Table 2.8.2 Example System Properties.

Property	Description
Probe Light Source	Quantum Cascade (QC) Laser
Wavelength λ1	~890 nm –Near IR.
Probe Optical Aperture	1 m (Gaussian Beam Angle is 1.13 μ rad.)
Earth Terminal Aperture	4 m
Modulation (Probe to Earth)	External Binary Phase Shift Modulation –BPSK
Modulation (Earth to Probe)	Amplitude Modulation - binary
Earth Terminal Detection	Coherent- Homodyne
Probe Terminal Detection	Incoherent (direct detection)

The system laid out in Table 2.8.2 is realizable in the near future with technology that is now being developed in the commercial sector. The QC laser is perhaps the most immature technology specified but its inclusion makes the coherent detection of the signal at the Earth-Terminal significantly easier to implement. At a distance of 100 AU the probe's 1 m aperture (1.13 μ Radians divergence) will have a spot size (e^{-1} from peak) at Earth of 17.1 x 10^6 m (1.35 Earth diameters). The 0.72 μ Radian pointing capability of the spacecraft means that acquisition of signal from the probe will be achievable.

The optical link from probe to Earth is a coherent link while the link from Earth to probe is incoherent. This asymmetry in intended to reduce the assets needed on-board the probe, i.e. optical local oscillator, and instead place the burden on the Earth-terminal where space and mass are less of a problem. The Earth-terminal carries an optical local oscillator to coherently detect the phase modulated signal from the probe. It also carries a higher power, amplitude modulated laser system so that the probe needs only to be equipped with incoherent, direct detection capabilities.

The advantage of having a coherent link from probe to Earth is that is virtually guarantees the detection is shot noise limited. The power of a coherent local oscillator (LO) acts as a gain mechanism during mixing and magnifies the signal (and shot noise). At a point this effect elevates the signal and shot noise above the other combined noises creating conditions optimum for signal detection. The mixing of the LO and the signal occurs in the photodetector which is a power (square law) detector. The power levels that occur in this technique are ideal for use of highly efficient PIN diodes (η=~70%).

2.8.2.3 Calculation of Probe Optical Power

In the simplest case, for perfect alignment of the two systems, the power received at the Earth-terminal, P_e, is given by

$$P_e = P_p \frac{\Omega_e}{\Omega_p} \qquad \text{Eq. 2.8.1.}$$

Where P_p is optical power of the signal leaving the probe. Ω_e and Ω_p are the solid angles received by the Earth-terminal and the solid angle subtended by the laser beam transmitted from the probe, respectively. With the proper substitutions Equation 2.8.1 can be rewritten as

$$P_p = P_e \left[\frac{4\lambda R}{\pi D_e D_p} \right]^2 . \qquad \text{Eq. 2.8.2}$$

Here D_e and D_p are the aperture diameters of the Earth-terminal and the probe, respectively. R and λ are the distance of the probe from Earth and the wavelength of the light source in the probe. Equation 2.8.2 calculates the power needed at the probe to attain a particular power at the Earth-terminal's detector. The power needed at the Earth-terminal to provide a particular signal to noise ratio (SN) in a shot noise limited system is

$$P_e = \frac{SN\ BW\ hc}{\lambda\ \eta} . \qquad \text{Eq. 2.8.3}$$

BW is the analog bandwidth of the detection system. The detector's quantum efficiency is η. Plank's constant (6.63 x 10^{-34} kg m^2/s) and the speed of light (3 x 10^8 m/sec) are h and c, respectively.

The signal to noise ratio (S/N) that is needed is based on the bit error rate (BER) and, for the case of Binary Phase Shift Modulation, is given by

$$BER = \frac{1}{2} erfc \sqrt{\frac{SN}{2}} . \qquad \text{Eq. 2.8.4}$$

If the desired BER is 10^{-9} then S/N must be 35 or greater (unencoded transmissions). Assuming a signal to noise ratio of 35 and an analog bandwidth of 1kHz (twice the digital bit rate) the power needed at the detector of the Earth-terminal given by Equation 2.8.3 is 1.1×10^{-14} watts. Using Equation 2.8.2 the probe laser power must be 0.2 watts at 100 AU and 20 watts at 1000 AU (holding S/N and BW constant).

These numbers are based on the coherent detection of a BPSK with 1kHz bandwidth and can be attained by the technologies that are going to be used. However, link losses have not been added and signal encoding has not been taken into account. There are many encoding techniques that could be used to significantly increase the signal to noise ratio of the link. In general they trade-off a slight reduction of bit rate for a large relaxation of the bit error rate (~3 dB). A typical encoding scheme can provide an effective BER of 10^{-9} with an actual BER of 10^{-6}. This means, using Equation 2.3.4, that the required S/N is about 20. This decrease in needed signal to noise ratio can be viewed as a 2.4dB bonus to the link margin.

The types of link margin loss that are expected in this communications link (high earth orbit to distant probe) are power losses at optical components at the transmit and receive systems, and pointing error losses. In this case of the Interstellar Probe the Earth's atmosphere is not a consideration. Also, since the probe will leave the plane of the ecliptic, interplanetary dust will not significantly scatter the communication signal. It is estimated that link losses as low as 3 dB can be achieved with proper optical design. In general, the link losses associated with the mission are on the order of the link gains offered by encoding.

If a system is considered that meets the specifications shown in Tables 2.8.1 and 2.8.2, including encoding and link losses, and it is assumed that there is a 1 watt optical source then Figure 2.8.3 shows the trade-off between bit rate and distance in during operation.

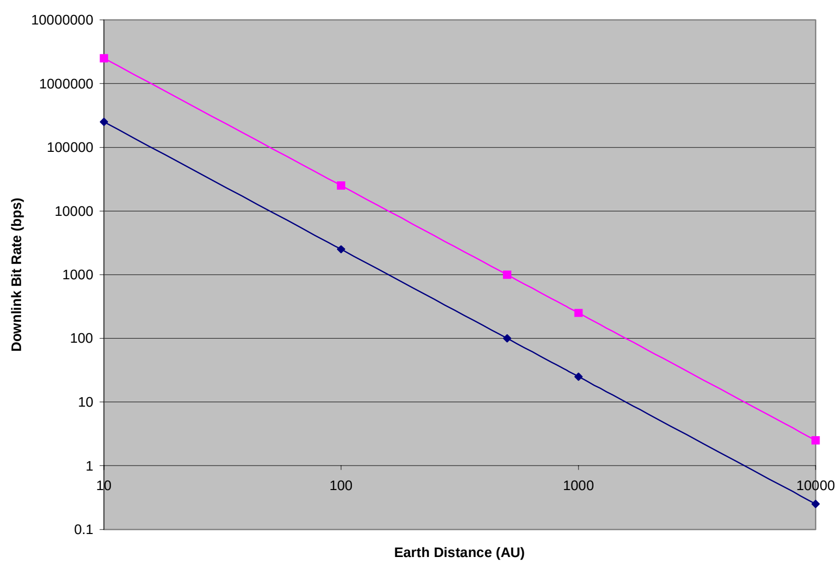


Figure 2.8.3. The Interstellar Probe downlink bit rate versus distance for optical powers for 1 watt (lower line) and 10 watts (upper line). (Bit rate is half the analog bandwidth)

2.9 Thermal System

Once the probe separates from the cocoon formed by the surrounding perihelion propulsion system, it quickly moves away from the Sun at 20 AU/yr. It spends a vast majority of its mission in the cold of deep space.

The interstellar probe contains a one meter, highly polished, mirror with a graphite epoxy substrate providing thermal and structural stability to the mirror. Electronics boxes and instruments are mounted to the back of the mirror support structure. An ARPS is used to provide the 15 Watts of electrical power to the instrument suite. The ARPS is mounted on the end of a 3 meter long mast, isolating the instruments from the harmful radiation emitted from the ARPS. Figure 2.9.1 shows the probe thermal model used to predict in-flight temperatures.

The thermal analysis performed on the probe assumed that the probe was far enough away from the sun to ignore any solar input. At a speed of ~100 km/s, this assumption becomes true twelve days after the perihelion burn. The goal of the thermal design for the mirror and its attached hardware is to operate between 75 and 125 K. Figure 2.9.2 shows box radiator area required to reject the heat dissipated inside the box at different box temperatures and at two different sink temperatures. The sink temperature is the effective temperature of everything surrounding a given radiator surface, including the ARPS, the adjacent boxes, the mirror support structure, the mirror and, of course, space. Note that at radiator temperatures above 100 K, the sink temperature variation is not a factor. Thus, a box top that has a surface area of 11” X 11” (0.078 m^2), dissipating one watt, will run at 126 K.

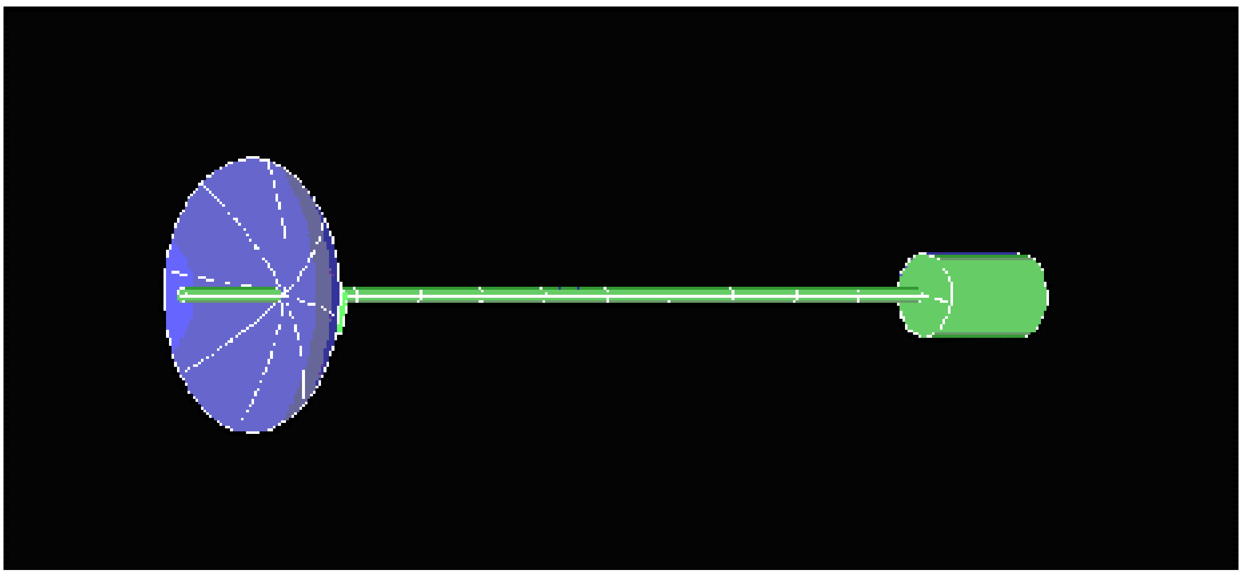


Figure 2.9.1

The thermal analysis performed on the probe assumed that the ARPS generates 100 Watts of thermal energy, with 15 watts going out as electrical power to the instruments. The 85 W of heat is rejected at the ARPS locally. The ARPS were assumed to have a surface temperature of 200 C. Table 2.9.1 shows the temperature predictions for the different components of the probe, for two different mounting conditions, isolated from the mirror structure and conductively mounted to the structure. Note that by conductively mounting the boxes, and leaving the outside of the boxes exposed, the box temperatures are lower than thermally isolated case. This shows that the

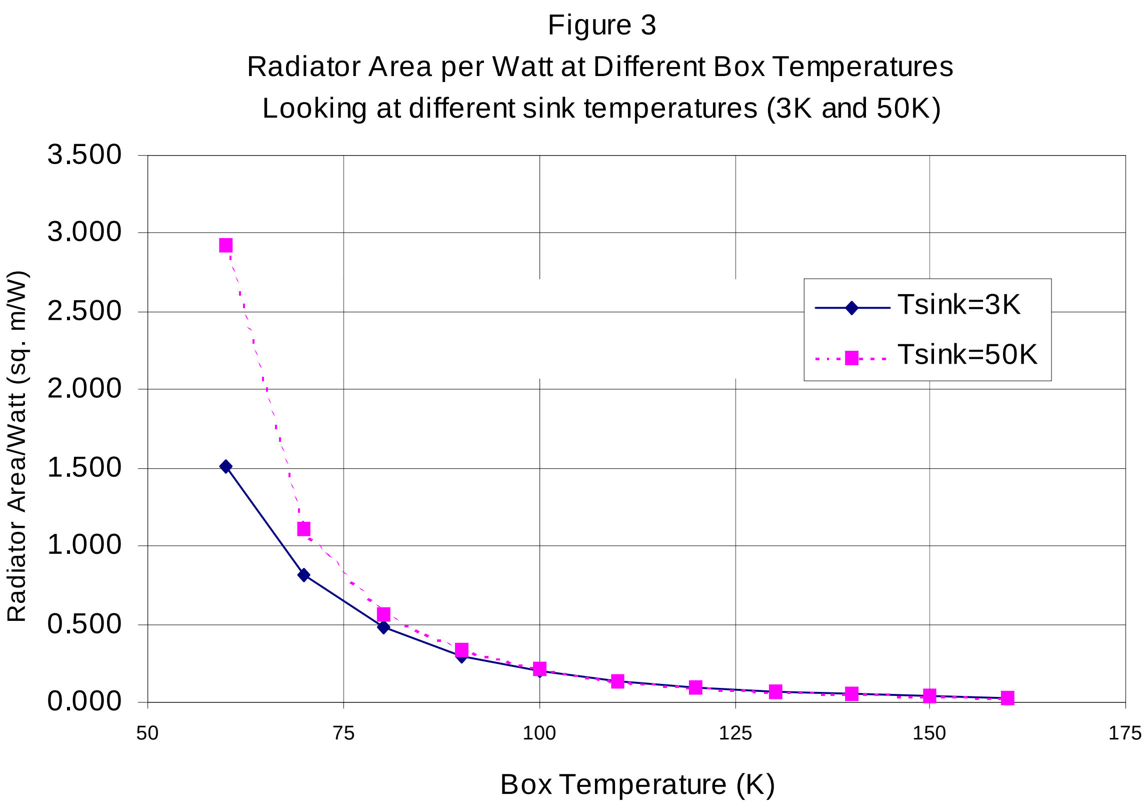


Figure 2.9.2

mirror can actually help spread the heat from the boxes, lowing the boxes temperatures. However, the mirror temperature increases slightly from the heat absorbed from the boxes and there is the possibility of local thermal distortions of the mirror, although at these power levels, that is a slight risk.

Table 2.9.1 Steady State Probe Temperatures with No Environmental Loading

Cases	Mirror Temperature	Box Temperature	ARPS Temperature
2 W/Box Conductive	-165 °C (108 K)	-155 °C (118 K)	200 °C (473 K)
1 W/Box Conductive	-175 °C (98 K)	-170 °C (103 K)	200 °C (473 K)
2 W/Box Isolated	-177 °C (96 K)	-115 °C (158 K)	200 °C (473 K)
1 W/Box Isolated	-182 °C (91 K)	-142 °C (131 K)	200 °C (473 K)

Several thermal concerns arise with operating electronics boxes and instrument at very low temperatures. First, since the probe will be integrated at room temperature, differential thermal expansion of the system must be considered to avoid distortions of the mirror and damage to the various subsystems. Second, special electronics components must be used to survive down to 100 K and below, considering standard Class-S parts are designed to operate between –55 C and +125 C (218 K to 398 K). Both of these concerns are not severe design drivers, but need to be considered in early design phases.

2.10 Payload

The scientific opportunities abound for a mission into the unexplored realm of the Local Interstellar Medium (LISM). For this reason an Interstellar Probe would be expected to carry a large complement of diverse instruments to measure a wide variety of physical phenomena. A number of studies (Jaffe & Ivie, 1979; Mewaldt, et al, 1995) have developed a framework for scientific exploration in this region, as well as a list of candidate instruments appropriate for the mission. In this section we will revisit some of the scientific issues and take a new look at the requirements governing the selection of an instrument payload for the Interstellar Probe. A candidate payload, with mass allocations which meet the overall 10 kg goal for the payload, is shown in Table 2.10.1.

Instrument	Identifier	Mass (kg)	Power (W)
Plasma waves/dust detection	PWD	1.5	2.5
Plasma/particles/cosmic rays composition and spectra	PPC	1.0	1.5
Magnetometer (w/boom)	MAG	3.0	0.5
Lyman-α imager	LYA	1.0	2.0
Infrared imager	IRI	1.5	1.5
Neutral atoms composition, density, speed, temperature	NAC	2.0	2.0
Totals	—	10.0	10.0

Table 2.10.1

It should be pointed out that the proposed science objectives are strongly dependent on the particular scientific community that is involved in the development of the objectives. Astronomers and astrophysicists seem to prefer optical telescopes and spectrometers which would be used to study faint distant objects, while the heliospheric and magnetospheric scientists naturally emphasize particles and fields instruments. As will be explained later in this section, the selection of instruments for this mission depends just as strongly on engineering constraints as on any particular science objectives.

The science objectives can be broken down into two regimes: a “near science” region from about 100 to 200 AU and a “far science” region extending from 200 AU to 1000 AU and beyond. The three “structures” of interest in the near region are the termination shock, heliosheath, and heliospheric bow shock. The ability to identify the termination shock requires measurement and processing of upstream and downstream parameters such as plasma density, temperature, magnetic field strength and direction, as well as ion and electron energetic particle fluxes and isotropies. Measurement of heliosheath characteristics is dependent on the direction of travel with respect to the solar apex and this plays into the choice of trajectory out of the solar system. Observation of the Hydrogen “wall” upstream and determination of the source of the heliospheric 2-5 kHz upstream radiation is desirable. Detection of the heliospheric bow shock, estimated to be at 200 AU, is another important goal of this mission.

In the far science region the important question involves ascertaining the spatial and temporal scales of dynamics with the LISM. Also, since the mission would penetrate partially into the Oort Cloud, sensing of the expected enhanced cloud density is important. From an astronomical perspective, measurement of stellar and intergalactic distance scales, optical observation of galactic and extragalactic objects, and UV spectrometry of the galactic hydrogen distribution are valuable contributors to our understanding of the universe.

The major driving force behind instrument payload selection is mass. The Interstellar Probe was intentionally designed to be very lightweight in order to minimize the demands on the propulsion system. The total mass of the probe is 50 kg, with 10 kg allocated for the scientific payload. However, the scientific payload cannot be made to fit within an arbitrarily tight mass allocation simply to meet the design constraints of the propulsion system. The mass allocation for the payload must be reasonable. It is believed that the mass allocation of 10 kg is reasonable because a number of technological advances, as well as innovative system level architectural choices, makes it possible for the suggested complement of roughly ten scientific instruments to fit within the allotted mass.

The architecture of the instrument payload is novel in several respects. First of all, none of the instruments have their own processor, but instead rely on a common bank of processors which support all operations performed on the probe. This distributed, self healing system is discussed in section 2.7.2 of this report. Secondly, the migration of functions from the hardware to the software realm occurs much earlier in the signal chain than in today's instruments. Of course, this trend toward implementing functions in software is not new, but development efforts being expended at this time will enable the software to be inserted at a point just after the conversion of the raw sensor signal to the digital world. As an example of this type of methodology which is currently being implemented at the Applied Physics Lab, consider an energetic particle instrument which measures energy using solid state detectors and particle velocity using a time-of-flight sensor head. Typically, a substantial amount of electronics is required (often a number of boards weighing a kilogram or more) to amplify and condition the energy signal(s), amplify and detect the timing signals, measure the time-of-flight, perform coincidence timing as well as valid event determination.

Recent advances in microelectronics have put a charge amplifier, shaping circuit, discriminator and A/D converter on a single small integrated circuit. Similarly, a TOF chip has been developed which contains dual constant fraction discriminators and a time-to-digital converter. It directly accepts the timing signals from the anodes in the TOF head. Hence, with minimal mass and power the detector signals are converted into the digital domain. However, the coincidence, binning, and valid event logic still consume a fair amount of electronics board space. This electronics is eliminated by simply applying digital timetags to all the various values (e.g energy and TOF) acquired by the instrument. The values, with their separate timetags, are fed directly into a high speed processor, where realtime coincidence determination and valid event selection are performed completely in software. Although software development is complex in its own right, it does mean that the mass of the sensor is reduced, that electronics parts which are subject to radiation are minimized, and that maximum flexibility is obtained. Although it may be argued that the electronic effort has simply been moved from dedicated hardware into other electronic parts (the bank of central processors), the processors are much more robust due to their high redundancy (many interlinked processors where each individual processor has the computing power to singly handle all the Interstellar Probe tasks, if required) and self healing capability, which involves sophisticated health monitoring and reconfigurability.

Each instrument communicates with the bank of central processors via a wireless high speed data link. In a sense, the entire spacecraft is inter-communicating via a type of cellular telephone system, where each instrument broadcasts on its own frequency. Each central processor knows which instrument is it's responsibility, so it tunes into the appropriate frequency and collects and processes the data from the correct sensor. This wireless link eliminates most of the heavy harness which is required on conventional spacecraft for connection of the instruments to the command and data handling subsystem (C&DH).

Another key advantage to this architecture is in the increased ability to make the remaining instrument systems robust and redundant. Instruments have certainly failed on past space missions and often this failure is due to the failure of an individual detector, a power supply, or due to a latch-up condition in the electronics from radiation damage. In the past it has been difficult to make instruments redundant since the number of parts to duplicate has been substantial in both mass and cost. With the instrument architecture described here, it becomes more reasonable to make portions of the instrument redundant since the amount and mass of electronics in the sensor head has been minimized. For example, it may be possible for a solid state detector and its associated "energy chip" to be made redundant with very little penalty in mass or design complexity.

In the following sections we examine a small subset of the possible instruments that would be appropriate for the Interstellar Probe. However, it must be stated that an enormous effort is currently being expended by scientists and engineers at various institutions to improve the sensors for *all* the candidate instruments. To a certain extent, those investigators whose instruments are chosen to fly will likely have been the individuals who were most successful in miniaturizing the sensor heads while still maintaining significant science gathering capability.

2.10.1 Magnetometer

The Interstellar Probe has been designed with a pair of booms for magnetometer sensors and a pair of booms for electric field measurements. The booms are mounted to the edge of the optical communications dish and are stowed alongside the probe's central mast when the probe is enclosed in the propulsion system. When the probe is pushed out of its "cocoon" after the perihelion burn maneuver all four booms swing out and lock into position.

Significant progress has been made in recent years with both magnetometer sensors and front-end electronics. Currently available fluxgate sensors have sensitivities adequate for the weak fields expected in the LISM. Their mass is around 250 gms and recent developments indicate that the sensor mass can be reduced to nearly 100 gms with no loss in sensitivity. Also, an alternative sensor is being developed using MEMS technology which uses an ultra-miniature oscillating cantilever. At present, their sensitivity does not match that of the fluxgate sensors and this limitation is a primary area of current research. The front-end electronics are being minimized in size by the use of chip-on-board (COB) fabrication techniques, as well as by moving some of the filtering operations from onboard passive components to digital filtering in software. Delta sigma converters are also undergoing significant improvement. For example, Analog Devices now has an A/D converter with three differential input channels (ideal for 3 axis magnetometers), programmable gain amplifier (useful for range changing) and 16 bit accuracy that consumes only 1 milliwatt.

One of the key aspects to successful operation of the scientific payload on the Interstellar Probe mission is data reduction. Because of the large travel distances involved and the need to keep the communication system small, the communication downlink rate is quite low. Transmissions only occur once a week and the total amount of bits allocated to a particular instrument is minimal. Lossless and lossy compression systems help somewhat, but it is necessary to perform a certain amount of high level data analysis onboard in order to distill the large amount of data down to a quantity that can fit into the telemetry allocation. The magnetometer instrument, which typically provides continuous time series data of the magnetic field strength, must convert the data to vector quantities and then heavily average the data (perhaps over periods of several hours). This averaged data is useful for gross tracking of magnetic field variations. However, the processing software would also be capable of identifying regions of greater interest which had occurred during the data acquisition period and save that information at a higher time resolution. The magnetometer's telemetry allocation would be able to accommodate transmission of a few of these "zoomed in" regions during the regular communication period.

2.10.2 Plasma/Radio Wave Experiments

A pair of opposed booms are available for placement of electric field sensors. Compared to most particles and fields instruments, the amount of front-end analog electronics that is needed is actually quite minimal. Typically only small, low-noise differential amplifiers are needed at the receiver to amplify the signals to a level where digitization is possible. After that point, plasma wave sensors have historically used a significant amount of specialized dedicated hardware in order to perform high speed narrow band swept frequency analysis as well as Fourier (FFT) analysis. Since the interstellar probe will have an extremely powerful bank of processors, it will be possible to perform virtually all this analysis in software. Hence, the significant amount of mass and power normally required by these instrument will be significantly reduced.

As in the case of the magnetometer instrument, the plasma wave instrument produces continuous streams of time series data which are usually converted into the frequency domain. The magnitude and phase results are then usually log compressed before being telemetered. This data would also have to be severely time averaged. However, higher level processing would be able to detect interesting structure in the data and save that portion of the data at a higher resolution for subsequent transmission to the ground.

2.10.3 Neutrals, Plasma, and Suprathermal Dynamics and Composition

The number of sensor heads required to cover the wide range of particles energies, charge states, ion species, and electrons populations expected during the mission is substantial. Science working groups have studied the instrumentation required and one team in particular has developed an instrument suite which measures Neutrals from 10eV to 6KeV, Plasma Ions from 0 to 10 KeV/q , Plasma Electrons from 10ev to 800eV, and performs composition and charge state measurements from 0 to 1 MeV/q. The number of sensor heads in this instrument package is seven, which clearly would exceed the mass allocation for the entire science payload. Many of these sensor heads are most needed for the "near" science region. If an Interstellar Probe Precursor mission (to 200 AU) or Interstellar Probe Pathfinder (remote sensing from within the solar system) are undertaken prior to the Interstellar Explorer mission, then some of the "near" region questions would already be answered and the number of sensor heads could be paired down for the Interstellar Probe.

A minimum complement that is needed for the LISM would consist of a low energy Neutral Gas sensor head covering the 10eV to 1000eV range, with composition capability being desirable, and high resolution ion and electron plasma sensors covering 0-10KeV/q and 10eV-

800eV, respectively. Neutral Gas analyzers have taken several forms; the simplest and lightest units cannot do composition, while complicated and heavy charge conversion systems can do composition. Unlike Plasma Wave instruments, where the sensor is relatively straightforward but the electronics are complicated, the neutral gas sensors have sensor heads which are difficult to reduce in mass, while the electronics are quite capable of being miniaturized.

Extremely elegant designs for ion and electron plasma heads have been developed by investigators in Germany and the United States during the past few years. The FIPS ion plasma head weighs only 700 gms while the MAREMF electron head weighs 450 gms. These heads are especially suited for placement on spinning spacecraft and provide nearly 4 pi coverage. The processing for these heads is straightforward and has been determined to require about 5 MIPs of computing power for full computation of plasma parameters. Hence, a portion of the computing power in a single Interstellar Probe processor module will suffice to perform all processing for these heads.

Again, due to the limited downlink bandwidth available, the data from these instruments, such as sectored M/q distributions and velocity measurements, must be heavily time averaged. Again, onboard algorithms that identify areas of scientific interest can be zoomed in on and a limited amount of high resolution data can also be downlinked. More advanced processing could conceivably involve generating composition and/or charge state plots which are overlaid over stored templates which represent a scientists prediction of various predicted physical phenomena (quiet times or shock profiles). If the plot representing the acquired data matches the template, then on-board decisions could be made to simply send down simple “condition indicators” (e.g. all quiet), rather then simply time series data. The condition indicator would remain valid until the data matched a new condition, at which point the condition indicator would be changed and , at that point, some representative data could be included in the telemetry stream.

3. Extension of the Mission to Longer Durations

The Interstellar Explorer mission provides a significant science return, performing initial investigations of the totally unexplored realm of Interstellar Space, as well as the interesting regions at the boundary of our solar system. Additionally, a number of engineering milestones will be achieved. Perhaps the most important milestone is enhanced mission duration, since it is clear that some future missions, such as those that journey to the stars, will be more than an order of magnitude longer in duration. Indeed, this mission is the first small step toward realistically achieving a true interstellar capability.

There are two keys aspects associated with performing extremely long duration missions. The first involves the reality of life on Earth. Wars and other extremely disruptive events often occur, making it difficult to maintain any infrastructure for periods longer than several hundred years. However, there are a few buildings that have been continually maintained for over a thousand years, so maintaining contact with a probe travelling to the nearest stars may be possible for that length of time. In order to maximize the probability of continued contact, it is important to distribute the data gathering knowledge among as many diverse peoples as possible. If the entire data gathering equipment and personnel are centralized, then a single catastrophe can wipe out all knowledge of the mission.

The second key aspect to a long duration mission is the selection of Interstellar Probe materials. Even though clever architectures can make systems robust and fault tolerant, long term aging processes can reduce virtually any system to dust over time. A number of previous missions, such as the Long Duration Exposure Facility (LDEF), have shown that even the low densities of particles in space can slowly degrade space materials. Hence, the choice of materials is extremely important for missions with durations longer than 50 years. One example is in the area of microcircuits, which are manufactured using silicon and other materials. These device are subject to electromigration, which eventually may cause the devices to fail. The feature size of the device, the current densities, as well as the interconnection material, are factors in the rate of electromigration. It may be necessary to use integrated circuits with large feature sizes, which are in some ways considered old technology, in order to achieve long life. However, a number of recent developments, such as ultra low power circuits (which have very low current densities), and the transition to copper and tungsten interconnects on sapphire substrates (instead of aluminum on silicon), may significantly improve the ultimate lifetimes of modern electronics.

4. Conclusions

We have provided a first-order cut at many of the engineering realities associated with sending a small interstellar precursor mission out of the solar system at a high speed. The primary engineering concern remains primarily propulsion, but we have also identified other constraints and interactions that need to be approached in a systems manner:

To maximize the asymptotic escape speed from the Sun, a target direction near the plane of the ecliptic must be chosen; although the constraints may not be as great, this is probably the case for low-thrust schemes as well, e.g., NEP and solar-sail propulsion.

If a Jupiter flyby is required to reach the Sun to do a perihelion maneuver to boost the escape speed, then additional planetary flybys will not help to ease the mass constraints of a given launch vehicle; they over-constrain the trajectory design problem.

Some form of the indirect launch mode - allowing an additional stage for the launch vehicle - is doable and probably required to launch a reasonably sized probe first to Jupiter and then to the Sun for the perihelion maneuver.

The Orion concept, *per se*, simply will not work with the limited mass available on a probe such as this. Fission may well provide the key element for the perihelion propulsion, but only in a pulsed mode with extremely-low fission yields per pulse.

Solar thermal propulsion and a near-Sun maneuver seem to be "made for each other"; problems that require more study are heat transfer to the working fuel and how high a specific impulse can be obtained. Liquid hydrogen (and its dissociation at very high temperatures) appears to offer the best solution, but suffers from the mass associated with storing a cryogen for over 3 years in deep space. Ammonia offers an excellent storage solution, but appears to limit the specific impulse. More study of the systems aspect is required with emphasis on the mass of the overall propulsion system hardware plus fuel.

Thermal shielding of the probe near the Sun is not an issue as long as the probe is not spinning and can actively point a Sun-shield toward the Sun during the perihelion passage. The umbra (shadow) of the shield does drive the mechanical configuration of the probe during the perihelion passage segment of the mission.

Data downlink, attitude control and knowledge, and communications means and power are all intimately linked; the data downlink requirements tend to drive the entire probe architecture for the interstellar flight configuration. Implementation is simplified by using "fire and forget" operations - the probe requires an autonomous and self-healing character so that uplinks are no longer necessary following final departure outbound from the inner solar system.

Low-power operations will help to ensure longevity of the probe while minimizing the required mass for a radioisotope power system. Isotopes longer lived than plutonium that have lower power densities offer engineering challenges in providing efficient production of electricity; the subject requires further study.

Continued miniaturization of scientific instrument electronics and detectors is required to implement a mission with a reasonable science return - the reason for the mission in the first place.

The concept we are pursuing for a realistic Interstellar Explorer has applicability to any robotic interstellar precursor mission. In addition, this concept offers potential advantages - as well as an alternative - to low thrust probes. Such systems require a great deal of control for their primary propulsion system. The approach here has only a few, short-duration critical operational periods. From a systems point of view this approach offers distinct advantages.

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