Ralph L. McNutt, Jr.

The Johns Hopkins University
Applied Physics Laboratory
Laurel, MD 20723-6099

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NASA Institute for Advanced Concepts **3rd Annual Meeting**
NASA Ames Research Center

NASA Institute for Advanced Concepts

Other Contributors

G. Bruce Andrews	System Engineering
Robert S. Bokulic	RF Communications
Bradley G. Boone	Optical Communications
David R. Haley	Guidance and Control
J. V. McAdams	Mission Design
M. E. Fraeman	Ultra-Low Power Electronics
B. D. Williams	Thermal Design
M. P. Boyle	Mechanical Design

D. Lester, R. Lyman, M. Ewing, R. Krishnan - Thiokol Corp
D. Read, L. Naes - Lockheed-Martin ATC
M. McPherson, R. Deters - Ball Aerospace

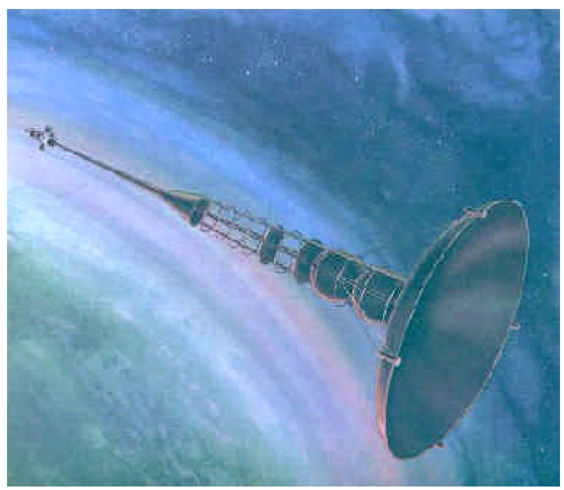
Support From:

Task 7600-039 from the NASA Institute for Advanced Concepts (NIAC) under NASA Contract NAS5-98051

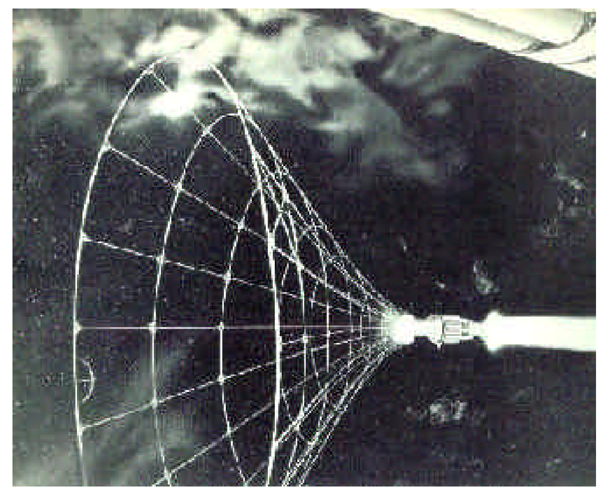
Starship Concepts Have Intrinsically Large Dry Masses (100s of Tons)

- Technology Extrapolations Sound "Too Good To Be True" - and May Be ...
- Driven by propulsion requirements
- Driving costs to scale of current GDP

Daedalus Fusion Rocket (D-^{3}He)

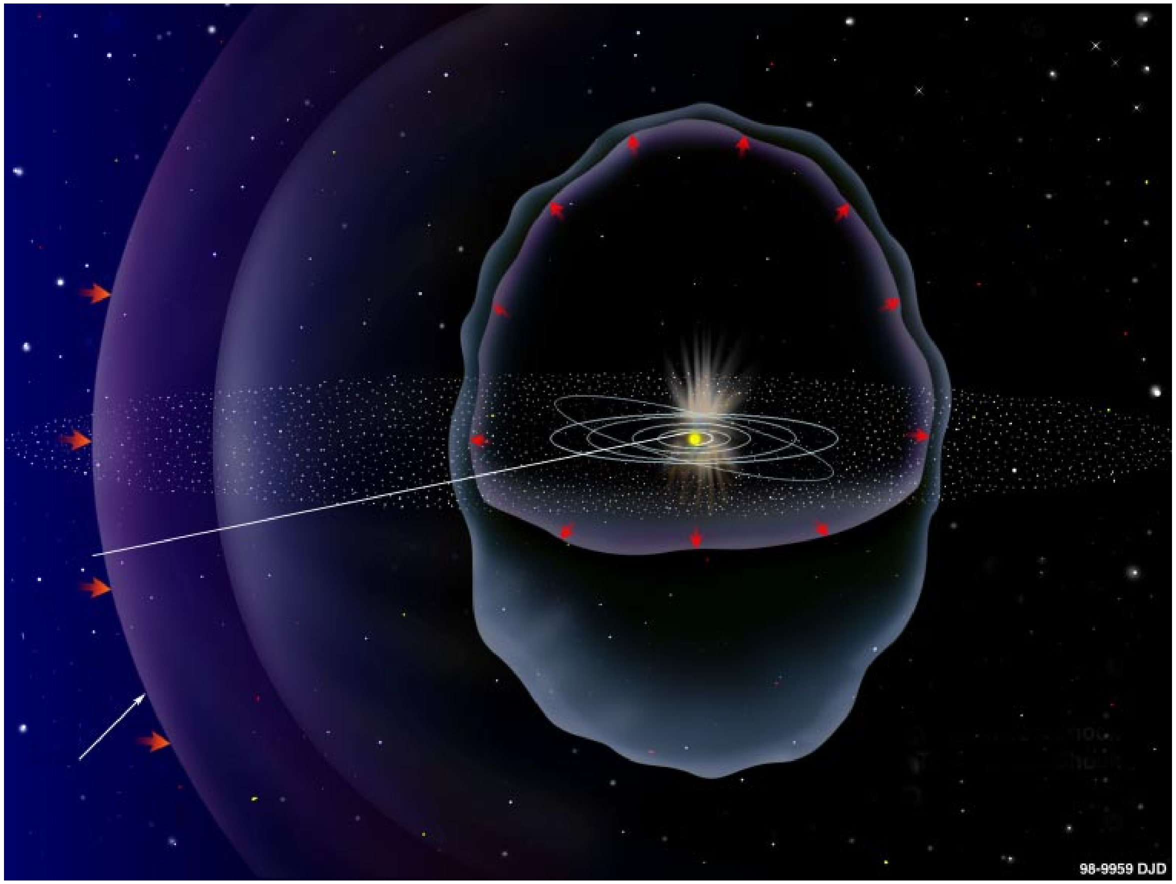


Sänger Photon Rocket



Bussard Ramjet

A Mission to the VLISM is More Modest - But Can be Done in the Near Term



The external shock may be ~300 AU away

- so 1000 AU is "clear" of the influcence of the Sun on its surroundings

Mission Concept

Reach a significant penetration into the Very Local Interstellar Medium - out to ~1000 AU - within the working lifetime of the probe developers (<50 years)

To reach high escape speed use a solar gravity assist (due to Oberth, 1929):

(1) Launch to Jupiter and use a retrograde trajectory to eliminate heliocentric angular momentum

(2) Fall in to 4 solar radii from the center of the Sun at perihelion

(3) Use an advanced-propulsion system ΔV maneuver to increase probe energy when its speed is highest to leverage rapid solar system escape

Enabling Technologies

- **High I_{sp}, high-thrust propulsion (for perihelion maneuver, ~15 minutes)**
- **Carbon-carbon thermal shield**
- **Long-range, low-mass telecommunications**
- **Efficient Radioisotope Thermoelectric Generator (RTG)**
- **Low-temperature (<150K), long-lived (<50yr) electronics**
- **<0.1 arc second pointing for data downlink**
- **Open loop control**
- **Fully autonomous operational capability with onboard fault detection and correction**
- **Possible extension to multi-century flight times while maintaining data taking and downlink operations**

Phase I Study Topics

Architectures that allows launch on a Delta III-class vehicle

Redundancies that extend probe lifetime to >1000 years; software autonomy, safing

Concept that links science, instruments, spacecraft engineering, and reality

1000 AU, 50-year mission with extension to 1,000 years (~20,000 AU)

Optical downlink to support 500 bps at 1000 AU

Propulsion concepts, e.g., Solar Thermal, Nuclear Pulse, Nuclear Thermal - to enable the perihelion burn

Phase II Effort Proposed to Refine and Develop Phase I Concepts

- Refine development of consistent thermal, propulsion, and mechanical design - In process
- Examine use of transuranic isotopes in propulsion and power system - Done
- Conduct STP and NTP system designs and trades including propellant selection and storage - Done
- Breadboard and program self-healing, distributed-processor spacecraft architecture to demonstrate use and resiliency - In process
- Develop optical-communication concept - Maturing
- Examine trades against low-thrust propulsion concepts - Starting

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Approach is to Maximize Use of Talent Base

Goal is the development of a "realistic" concept

Needs dreams as well as solid engineering approach

So ....

Solicit input from lead engineers delivering flight hardware - who also have proven track records

Best solution is find personnel who have ALSO worked TRL 4-6 ATD programs

Use NIAC funds (limited) to maximize technical work by "fitting in" around ongoing flight and ATD programs (ensures best engineering talent - which is also typically oversubscribed)

Mission Design

Solar system escape speed is set by the ΔV at perihelion and the radial distance from the center of the Sun r_p

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

1 AU/yr = 4.74 km/s

A "now-technology" Interstellar Probe could supply a ΔV of 1.56 km s^{-1} at perihelion, about one-tenth of what is desirable. Perihelion distance = 3 R_S => ~7.0 AU yr^{-1}

To reach ~20 AU yr^{-1} the probe needs to be accelerated by ~10 to 15 km s^{-1} during about 15 minutes around perihelion to minimize gravity losses

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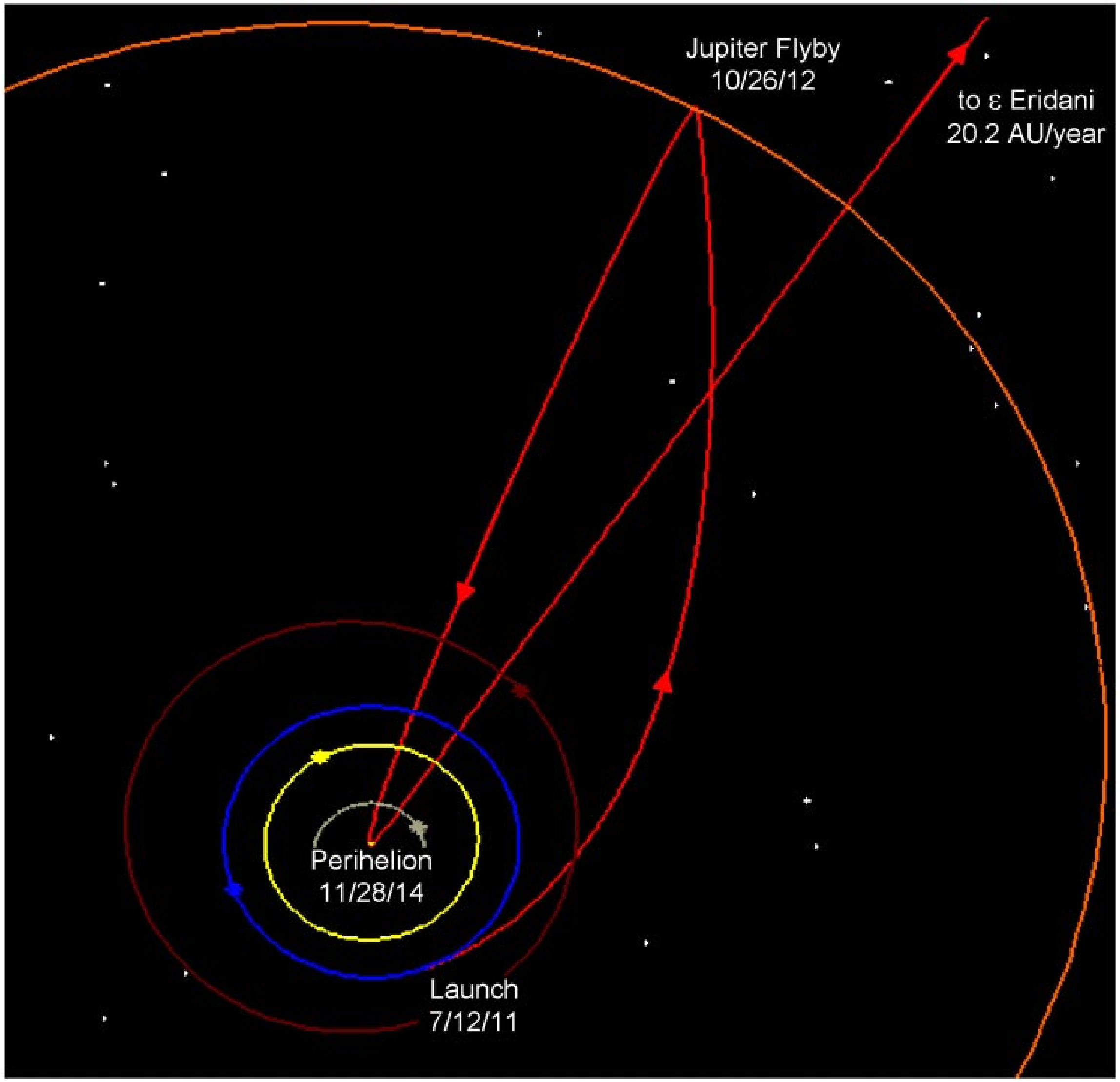
Trajectory toward ε Eridani

Launching toward a star enables comparison of local properties of the interstellar medium with integrated properties determined by detailed measurements of the target-star spectrum, so we target the Sun-similar star ε Eridani, a K2V dwarf main sequence star 10.7 light years from Earth



[Showed Epsilon Eridani Movies]

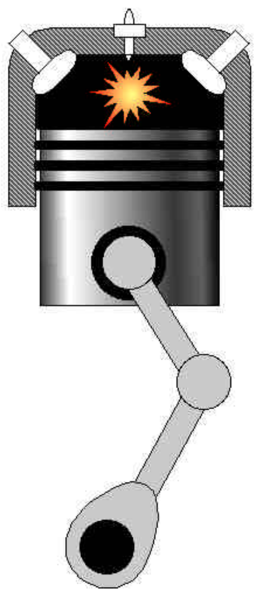
Nuclear Pulse Propulsion

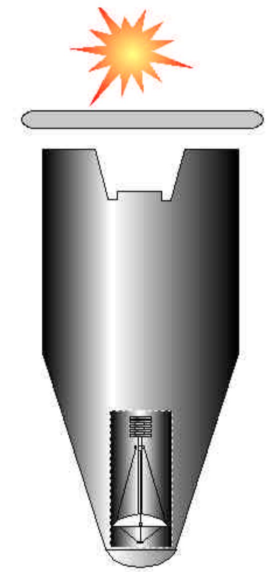
Pulsed fission can, in principle, provide the key element for the perihelion propulsion. For a 260 kg probe (incld. 30% margin) and 215 kg of propellant, we need the fission energy of ~1.3 g of uranium - a total of about 13 tons of TNT equivalent

The problem is the coupling of the momentum into the ship over short time scales ~10^{-8}s and this is exacerbated by yields of ~1 to 10 kT explosions.

Scalability to low-mass systems is problematic due to critical mass of fission assemblies

Even then most promising known transuranic elements do not solve the problem - Np-236, Pu-241, Am-242m, Cm-245, Cf-249, and Cf-251 evaluated and compared with U-233, U-235, and Pu-239





Conclusion: Cannot be applied to "small" systems

Thermal Propulsion

- Chemical propulsion cannot provide sufficiently high I_{sp} due to the high mean molecular weight of the combustion products
- Suggests using low-molecular weight and a decoupled energy source
- Solar Thermal Propulsion (STP) - tap the Sun's energy via the thermal shield - Being studied here
- Nuclear Thermal Propulsion (NTP) - use a compact ultra-low mass MInature ReacTor EnginE (MITEE) [Powell et al., 1999]
- Advanced architecture (U-233 fuel, BeH_2 moderator, LH_2 reflector) could provide criticality in a ~40 kg package.
- Further decrease could come from Am-242m fuel (supply issues !!)
- Question of possible size of Pu-239 system (plenty of fuel!)
- Needs further study at the systems level

Interstellar Probe Thermal Requirements

- **Survive cruise mode prior to perihelion pass**

  Protect propellant system

- **Survive high heating rates at $4R_S$ (2900 Suns)**
- **Allow perihelion burn to accelerate vehicle**
- **Deploy probe after burn**
- **Use waste heat from RTG (or equivalent) to minimize heater-power requirements**

  Operate probe electronics at ~ 125 K

Trade Studies

- Concentrate on STP system - results also apply to NTP
- Sufficiently large I_{sp} to provide ΔV
  - Examine LH_2, CH_4, NH_3
  - Maximize propellent temperature (up to structural failure)
  - Examine pressure vs flow rate, heating, and recombination
- Size propellant tank/cryostat for propellant requirements
  - Storage for cruise
  - Pressure and expulsion during burn

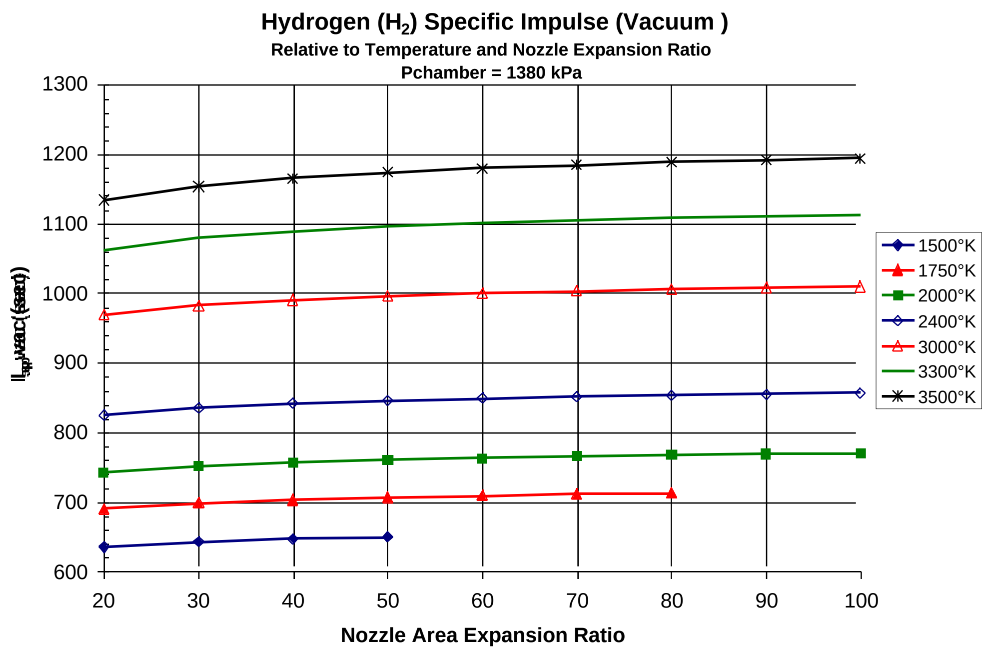
Solar Thermal Propulsion Concept

	Pressure (kPa)	Temperature (K)			
		2400	3000	3300	3500
H_2	517	860	1037	1166	1267
	69	875	1144	1336	1369 †
CH_4	517	480	588	667	705††
	69	485	628	698†	705††
NH_3	517	421	502	559	604
	69	427	547	634	639†† †

† Pressure = 165 kPa

†† Pressure = 910 kPa

††† Pressure = 221 kPa



The baseline propellant hydrogen shows the most promise for obtaining the maximum I_{SP} level

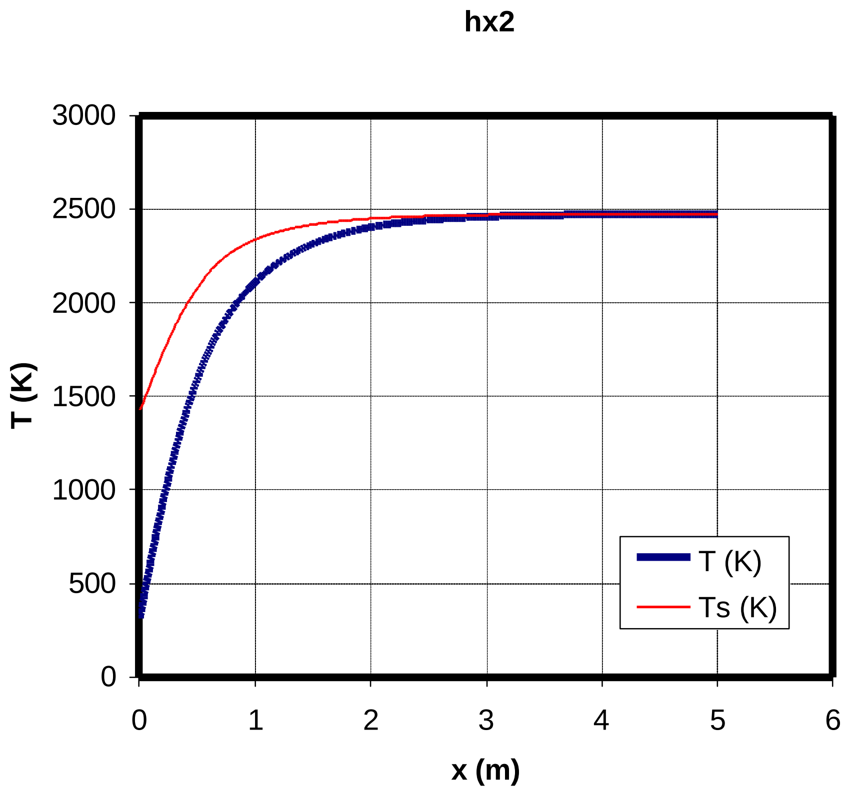
Maximum achievable I_{SP} with NH_3 and CH_4 are 639s and 705s, respectively (at 3500K)

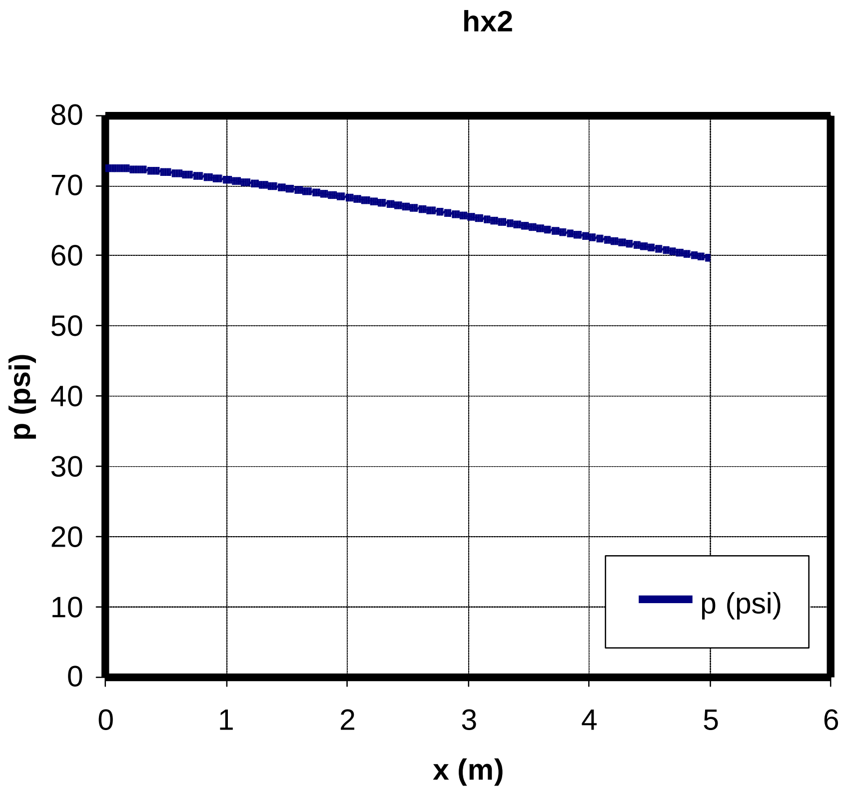
Parametric Analysis

- Incident heat flux:
  - 381 W/cm² at 4 R_s
  - 396 W/cm² at 3 R_s
- # of plies:
  - 1 (0.3 mm)
  - 2 (0.6 mm)
  - 3 (0.9 mm)
- Spacing, s
  - 5 mm
  - 10 mm
  - 15 mm
- Mass flow rate
  - 200 g/s
  - 1100 g/s
  - 2000 g/s

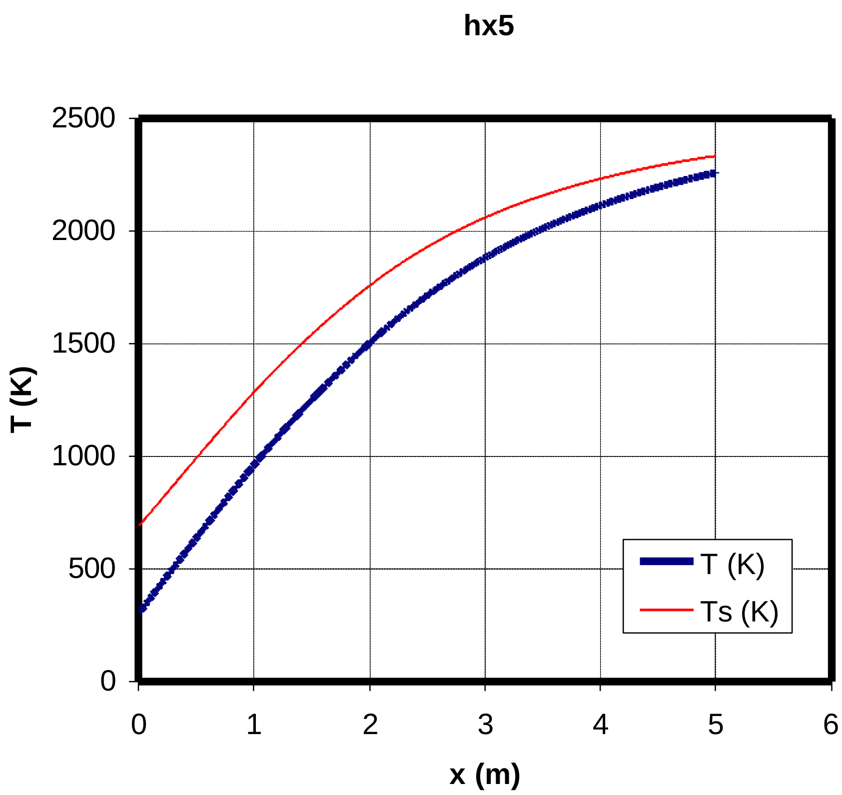
Test Case	q''_{inc} (W/cm²)	# plies	spacing (mm)	m_{dot} (g/s)	p_{in} (Pa)
hx1	381	1	10	200	500000
hx2	381	3	5	200	500000
hx3	381	3	10	200	500000
hx4	381	3	15	200	500000
hx5	381	3	5	1100	1500000
hx6	381	3	10	1100	500000
hx7	381	3	15	1100	500000
hx8	381	3	5	2000	1800000
hx9	381	3	10	2000	1800000
hx10	381	3	15	2000	1800000

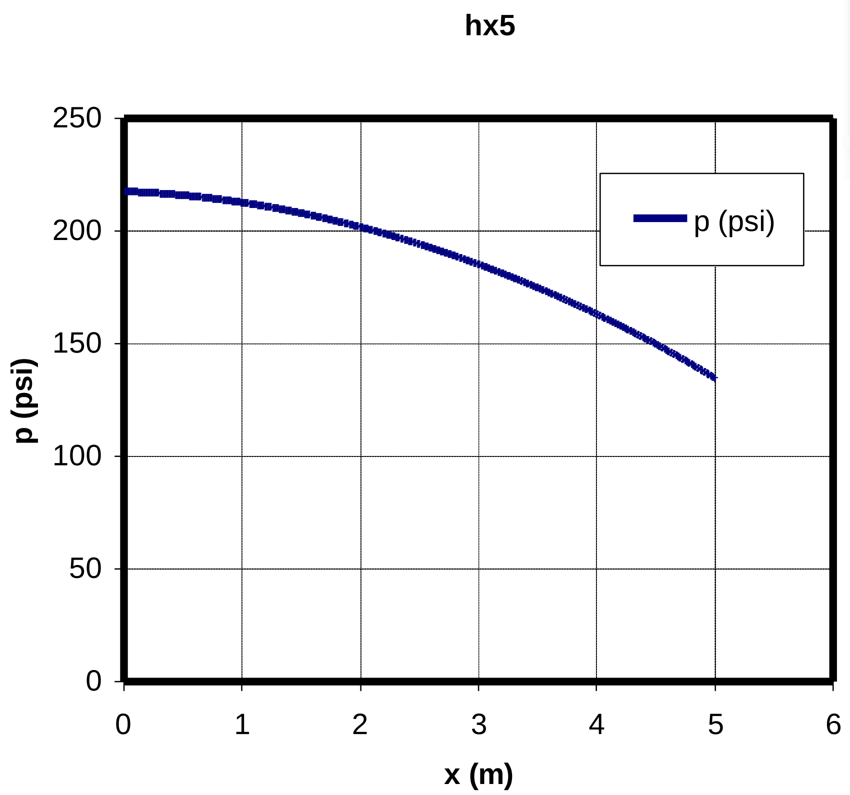
Results – hx2


hx2
3000
2500
2000
1500
1000
500
0
T (K)
0
1
2
3
4
5
6
x (m)
T (K)
Ts (K)


hx2
80
70
60
50
40
30
20
10
0
p (psi)
0
1
2
3
4
5
6
x (m)
p (psi)

Results – hx5


hx5
T (K)
2500
2000
1500
1000
500
0
0
1
2
3
4
5
6
x (m)
T (K)
Ts (K)


hx5
p (psi)
250
200
150
100
50
0
0
1
2
3
4
5
6
x (m)
p (psi)

Heat Shield Structural Evaluation

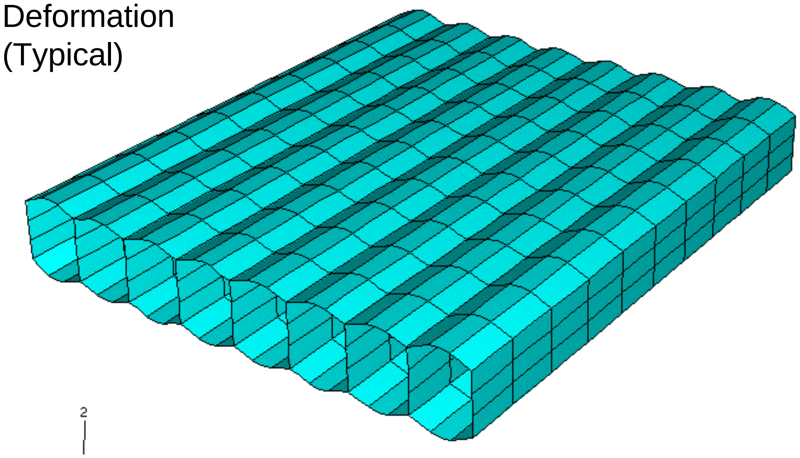
- The 5mm and the 10mm cell size configurations were evaluated for structural integrity
  - Six different wall thicknesses (1ply to 6ply, 0.3mm each)
  - Typical 3D carbon-carbon material properties used @ 3000F
  - 200 psi fluid pressure assumed
  - Stress criteria used to determine acceptable configurations

Maximum Stresses (Allowable stress ~ 14 ksi)

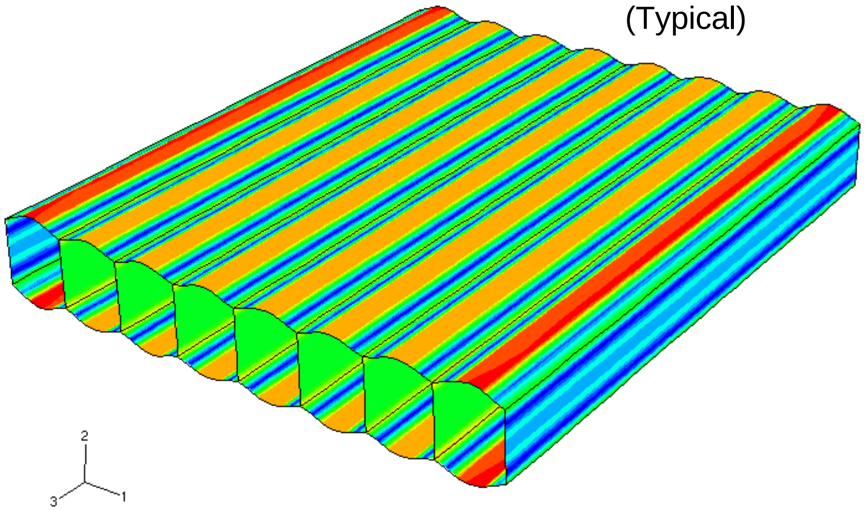
CELL	1ply	2ply	3ply	4ply	5ply	6ply
5mm	23,740	4,278	1,996	1,249	874	658
10mm	91,890	22,540	6,571	4,044	2,779	1,930

Heat Shield Structural Evaluation

Deformation
(Typical)



Stresses
(Typical)



Relative weight of a 1 inch specimen					
CELL (mm)	# cells	total area	2ply wt (lbs.)	3ply wt (lbs.)	5ply wt (lbs.)
5	1730	1021.589	1.56	2.33	3.89
10	868	1025.299	1.56	2.34	3.90
15	581	1029.01	1.57	2.35	3.92

LH_2 Storage Options

SYSTEM	HYDROGEN STATE AT LAUNCH	COMMENTS
Non-vented	Solid	• Parasitic heat leak melts and pressurizes the hydrogen during transit • Small amount of heater power may be required to achieve desired thermodynamic state • Cold pressure relief valve protects against over-pressurization • Stratification issues • Minimum tankage volume and launch mass
	Liquid	• Maintain temperature below 30 K during transit to minimize volume and pressure • Hydrogen is maintained at desired temperature and pressure throughout transit • Active cooling required to maintain < 30 K • Failure of active cooling is catastrophic • Stratification issues • Minimum tankage volume
Vented	Solid	• No liquid management required since the hydrogen is maintained below the triple point until shortly before perihelion • Heater power required to melt solid before perihelion • Minimum tankage volume. Slightly more hydrogen mass at launch (compared to non-vented system)
	Liquid	• Thermodynamic vent needed for phase separation • Larger volume and mass than solid since liquid density is smaller

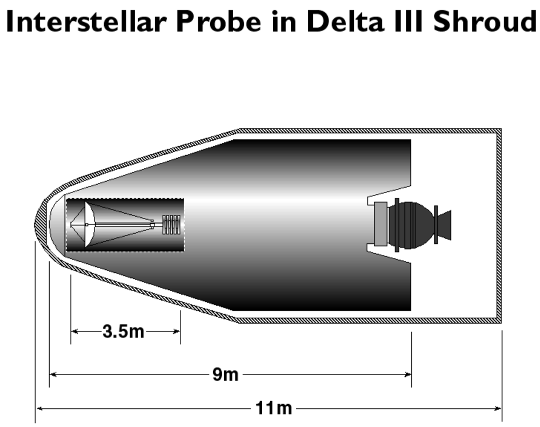
Yellow indicates preferred option: non-vented/solid at launch

	2219 Aluminum		8090 Aluminum Lithium		UL50 Aluminum Lithium		Graphite/Epoxy	
H_2 mass (kg)	system mass at launch (kg)	system mass w/o vac shell (kg)	system mass at launch (kg)	system mass w/o vac shell (kg)	system mass at launch (kg)	system mass w/o vac shell (kg)	system mass at launch (kg)	system mass w/o vac shell (kg)
200	448	386	431	378	415	371	378	353
300	627	536	602	524	579	514	523	488
400	804	685	771	669	741	655	667	621

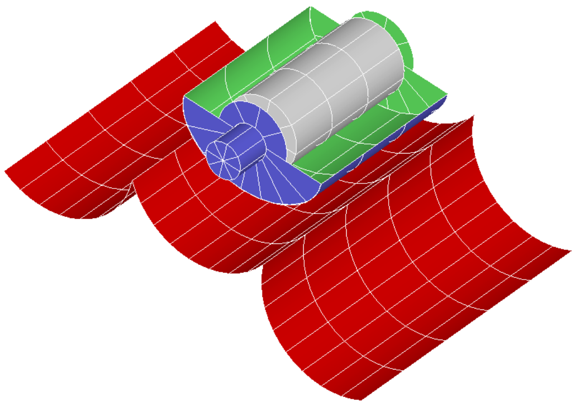
H_2 mass at perihelion (kg)	Parasitic Heat Leak (W)	Additional heating to pressurize H_2 (W)	System Mass (kg)	System Mass after vac shell deployment (kg)	System Envelope (L/D) (m/m)
200	0.0073	0.257	448	386	2.29/2.00
300	0.0076	0.389	627	536	2.60/2.27
400	0.0079	0.521	804	685	2.85/2.48

Best design is for low-pressure system with graphite epoxy; launch with solid LH_2 and gradually melt prior to perihelion

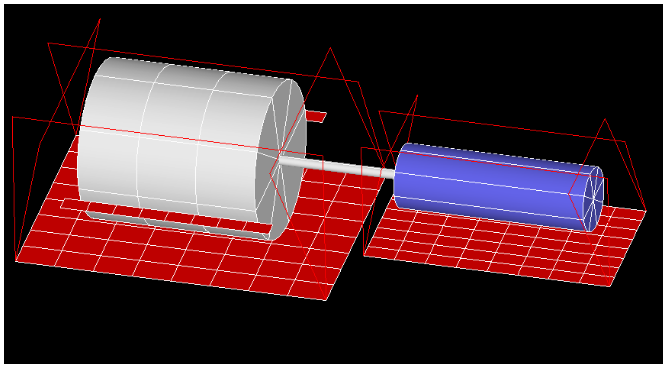
Configuration Evolution Driven by LH_2 Volume



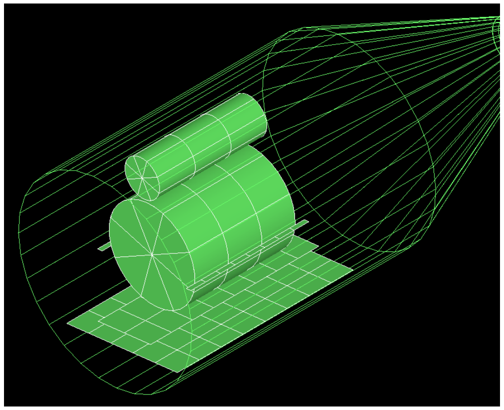
(1) Initial Concept



(2) Maximize volume for Delta III



(3) Size diven by 250 kg (dry) cryostat



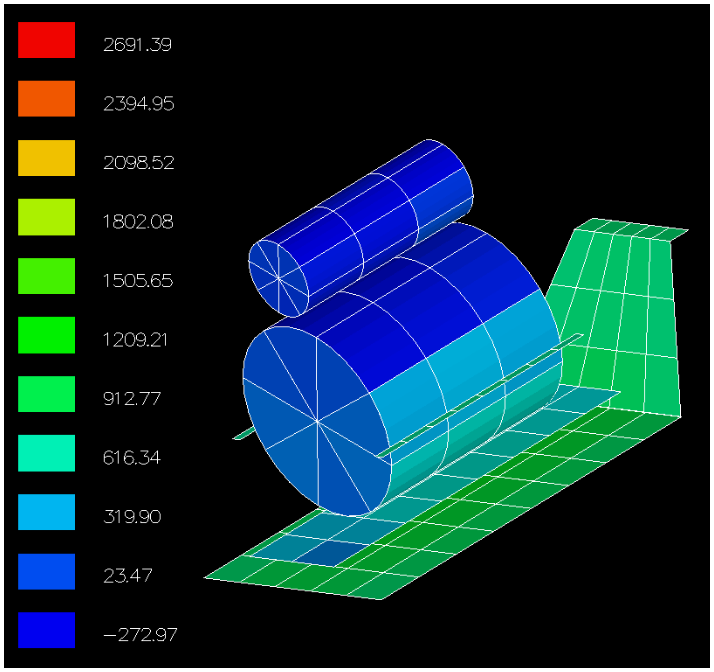
(4) Stack probe and cryostat shield in 5-m shroud

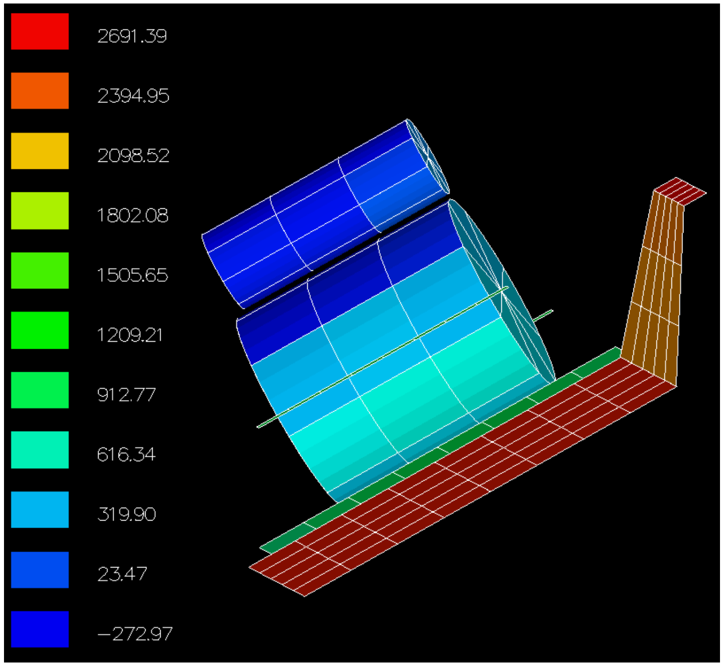
Thermal Constraints Are Met

Analysis with Thermal Synthesis System (TSS) software

CC primary shield with 0.85/0.55 α/ε at temperature (2964K)

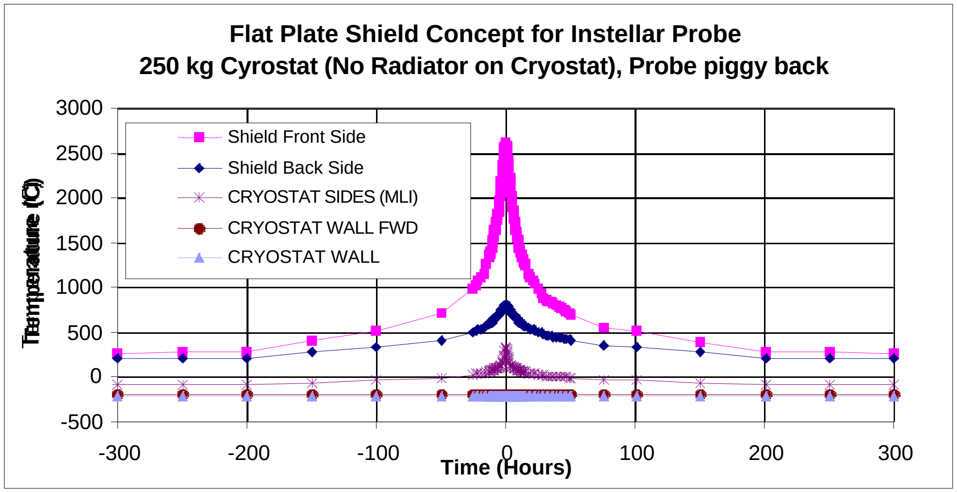
~100 kg of CC aerogel backing on primary shield



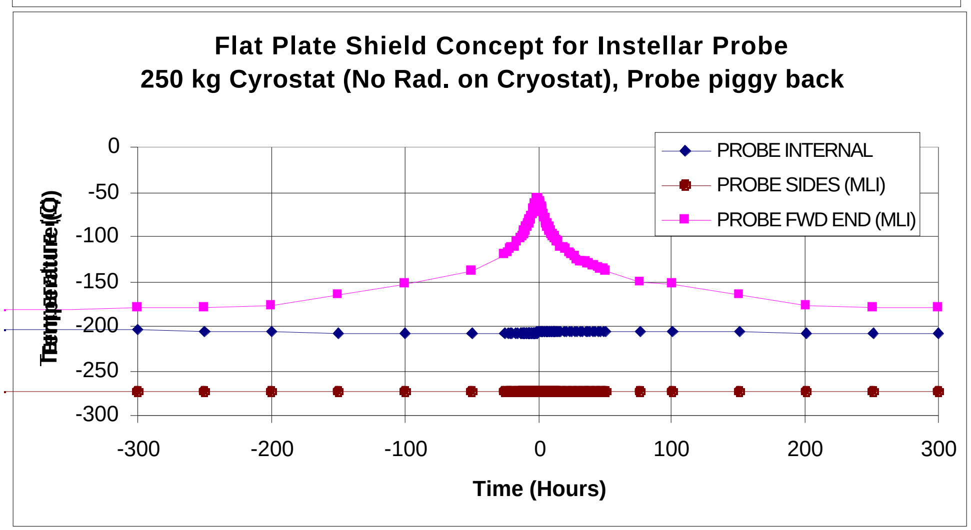


“Fins” on sides of cryostat capture energy to exhaust propellant

LH_2 Thermally Isolated Until Needed



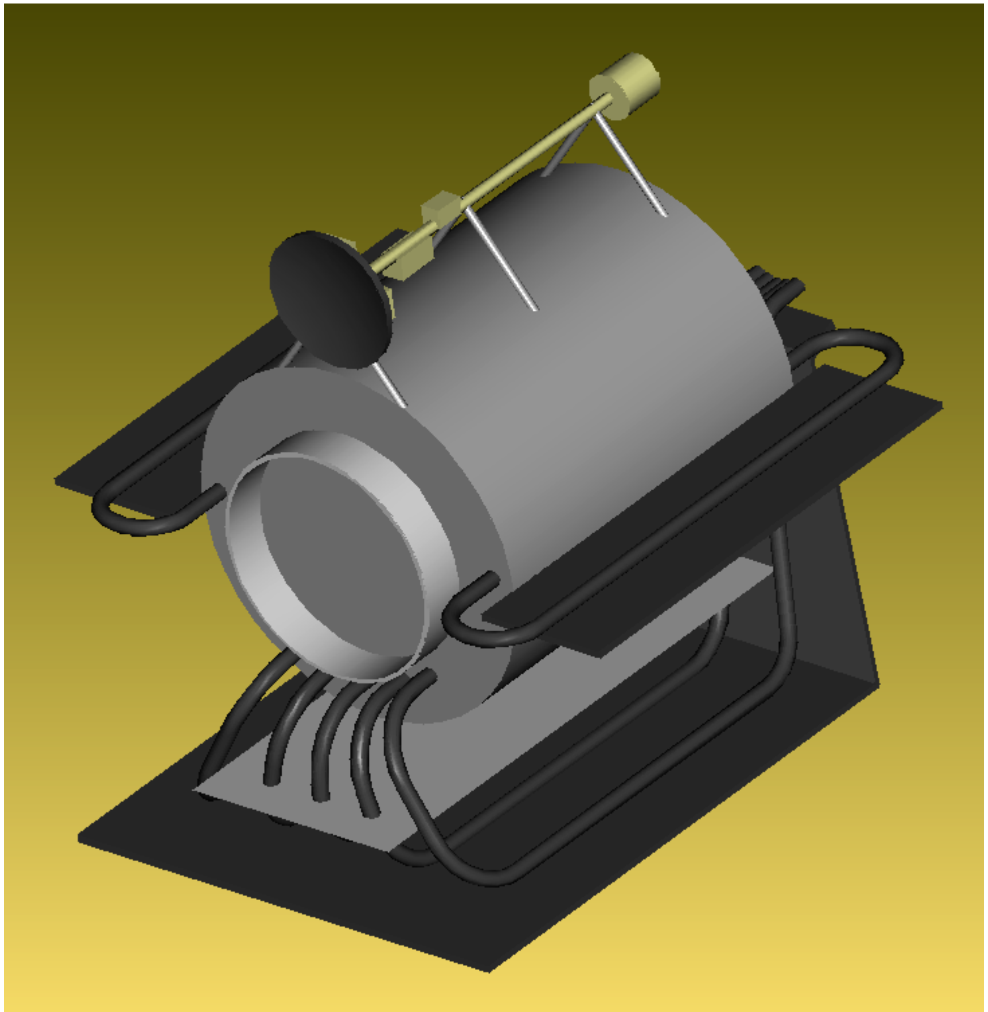
Overall temperatures



Internal temperatures

Current Concept Accomodates 400 kg LH_2

- Protect LH_2 with thermal shield
- Keep CG in line with thrust
- Propellant lines connect tank to shield and to DeLaval nozzle and to perihleion heat exchangers
- Fits in 5-m shroud
- Primary and secondary thermal shields
- Adaptor ring shown
- Probe rides in shadow of propellant tank



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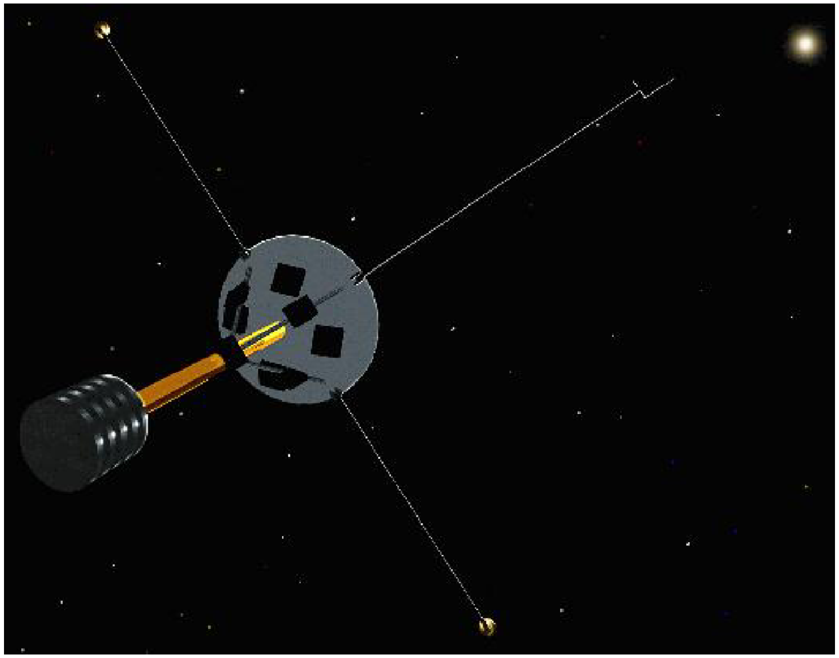
Interstellar Probe Final Flight Configuration



50 kg, 15 W probe

Operate at ~125K

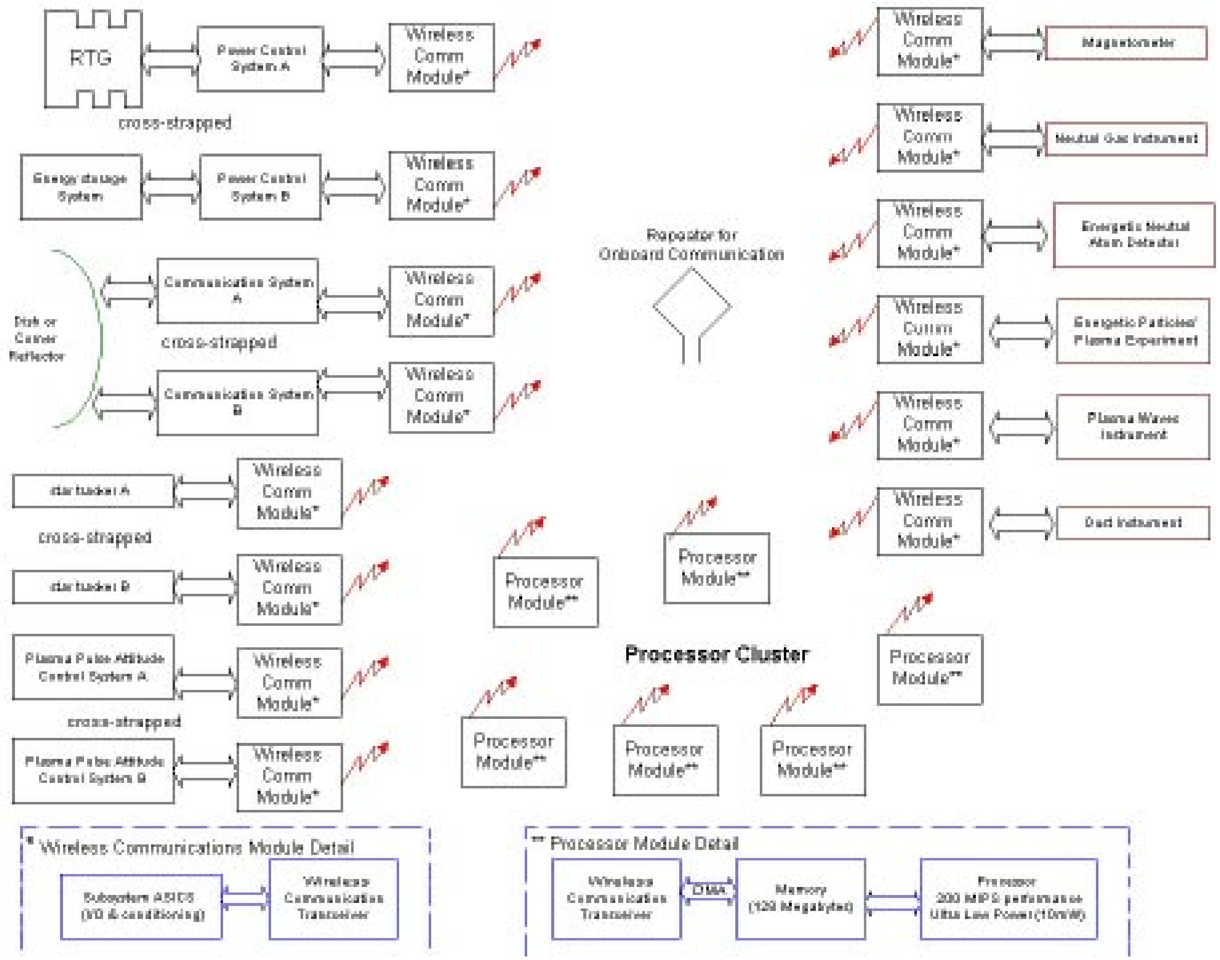
Includes:
10 kg, 10W
Science Instruments

Following the perihelion burn, the probe consists of three main mechanical elements – an RPS, a central support mast containing the comm laser and battery, and an optical dish pointing toward the solar system.

Instruments and processors mount to the back of the dish.

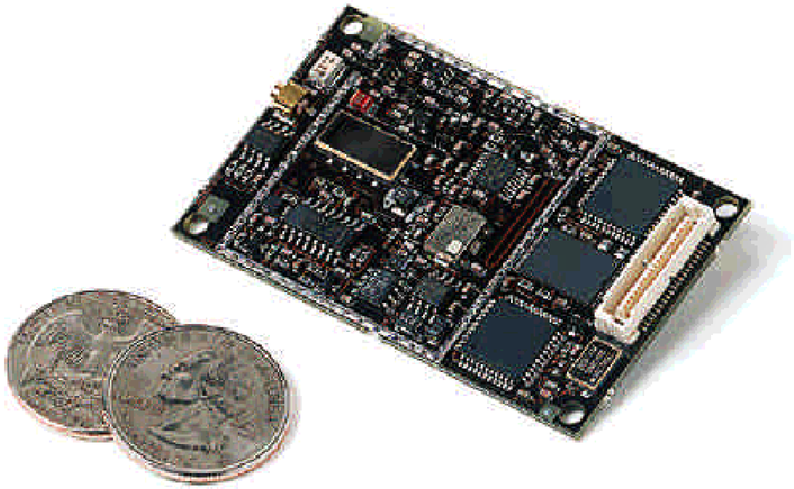
Probe Block Diagram

Perihelion propulsion module is not shown

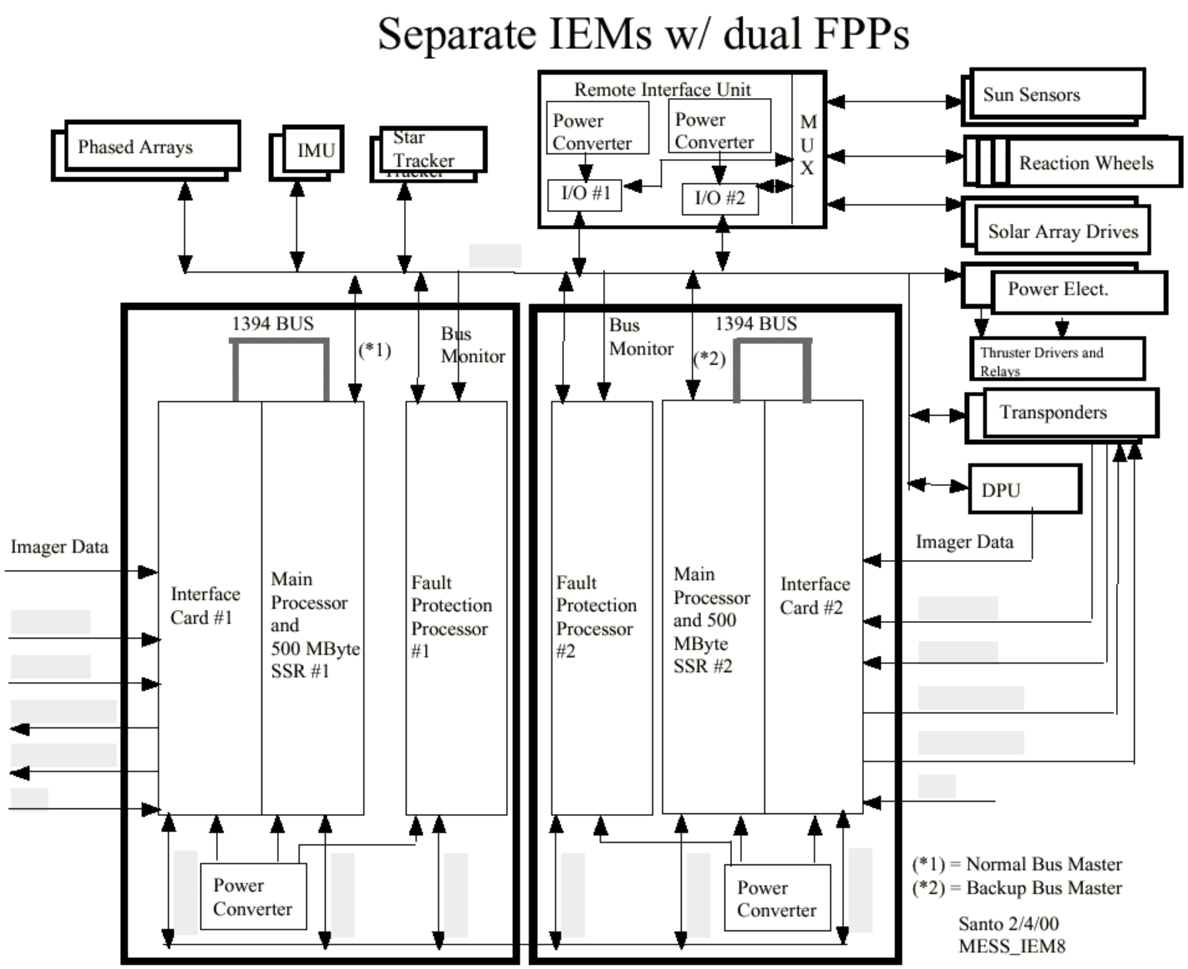


Wireless Communication Module

- 2.4 GigaHz operation
- 72 channels
- Two will be interfaced to each ultra low power processor using the serial RS-232 ports (which support 56 K-baud communication)
- One is used for inter-processor communication, the other for communication with subsystems



Typical S/C Architecture (MESSENGER)



Why are Simple Dual Redundant Systems the Current "Standard"?

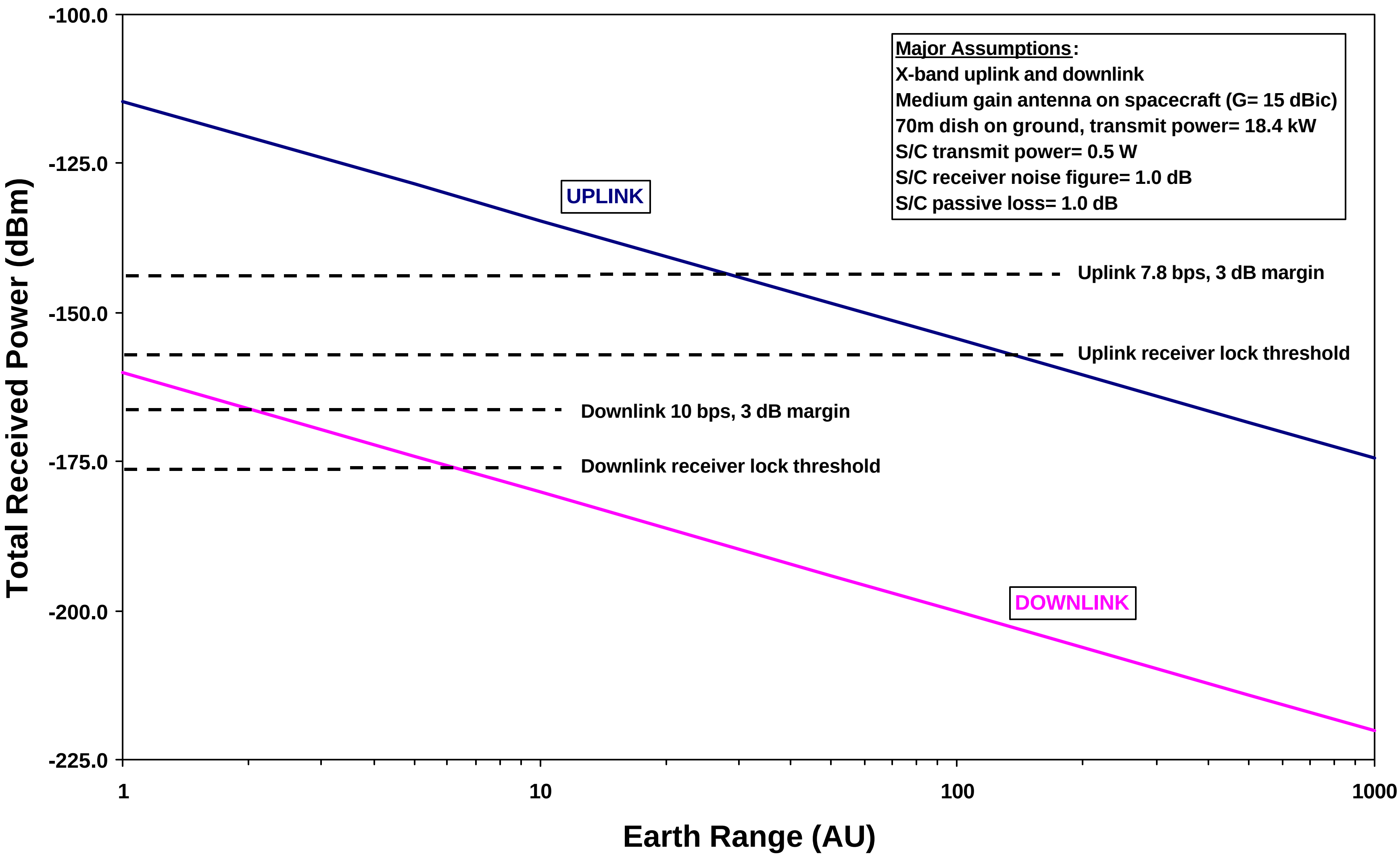
- Good flight reliability history for missions < 10 years long. Why change?
- Ultra low power (ULP) processors, which would enable more redundancy on a S/C, are not flight ready.
- Even if ULP processors where available now, cross-strapping S/C subsystems between >4 processors is cumbersome.
- RF links for inter-processor communication, as well as with S/C subsystems and instruments, enable n-way cross-strapping, but they, too, are not flight-worthy at this point in time

Advantages of Interstellar Probe S/C Architecture over Current "Standard"

- **Each ULP processor on IP is powerful enough to run S/C operations by itself, so if IP has "N" processors then IP has true "N"-fold redundancy**
  - **Fault Protection Processors on classic systems provide opportunity for Ground Operations to fix problems with main flight processor. However, they are not powerful enough to run S/C by themselves, so not very useful for missions that must operate autonomously.**
- **All processors not assigned to be the master act as "watchers", hence more oversight than with a single FPP per flight processor**
- **RF links between processors allows for N-fold redundancy**
- **RF links between processors and subsystems allows for easier implementation of subsystem redundancy**

A Realistic Interstellar Explorer

Link Analysis Results

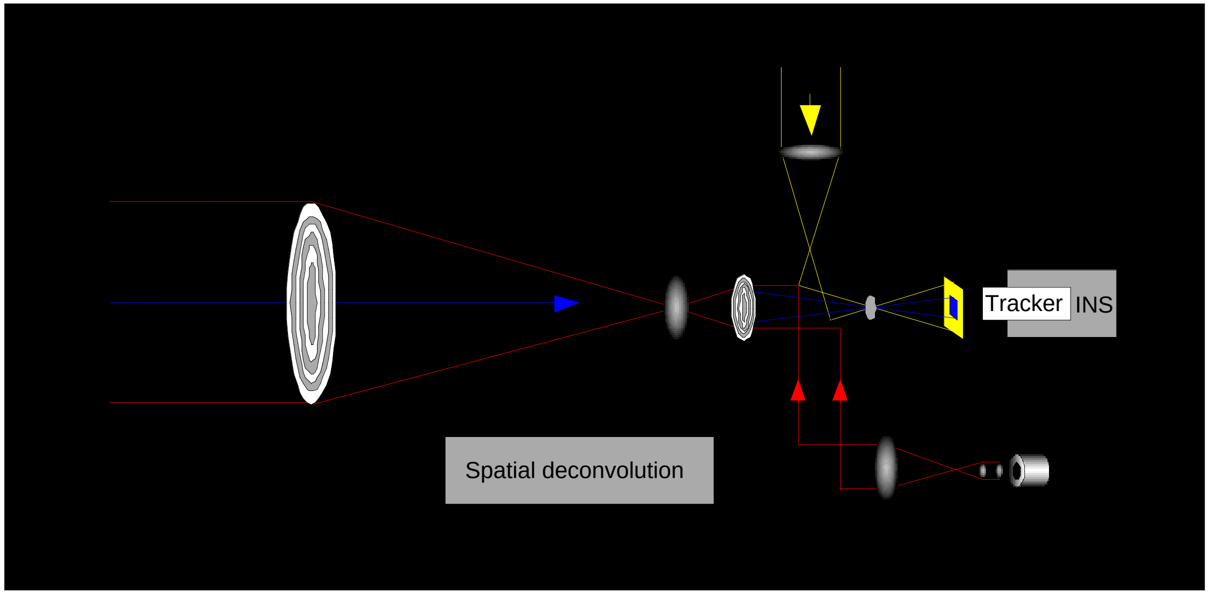


Optical Communication System

System requirements:

- **Average transmit power > 20 W**
- **Aperture: 1 meter**
- **Burst data rate: 500 bps @ 1000 A.U.**
- **Pointing accuracy ~ 300 nrad**
- **Intensity modulation - direct detection**
- **Co-boresighted fine guidance tracker**
- **Off-axis coarse tracker**

JHU/APL concept incorporates advanced technologies (VCSELs, MEMs, and diffractive optics) to minimize mass and prime power



Schedule

2000-2002 Advanced Technology Development study(ies)
2000-2002 Continued definition studies of the solar sail concept for IP at JPL
2002-2003 Update of OSS strategic plan with study for a "New Millennium"-like mission
2003-2007 Focused technology development for small probe technologies
2004-2007 Development of sail demonstration mission
2004-2007 Development of Solar Probe mission (test for perihelion propulsion)
[2006-2007 Hardware tests for radioisotope sail feasibility]
[2006-2007 Hardware tests for antimatter propulsion schemes]
2006-2007 Monitor DoD STP effort and conduct NASA-specific hardware tests
[2002-2007 Development of space-qualified nuclear thermal reactor]
2007-2010 Focused technology development for an Interstellar Probe
2009-2012 Design and launch of first generation solar-sail probe
2010 Test of Solar Probe performance in the perihelion pass of October 2010
2012-2015 Design and launch second generation probe 1000 AU goal in 50 years
2015-2065 Data return from 1000 AU and "beyond the infinite..."

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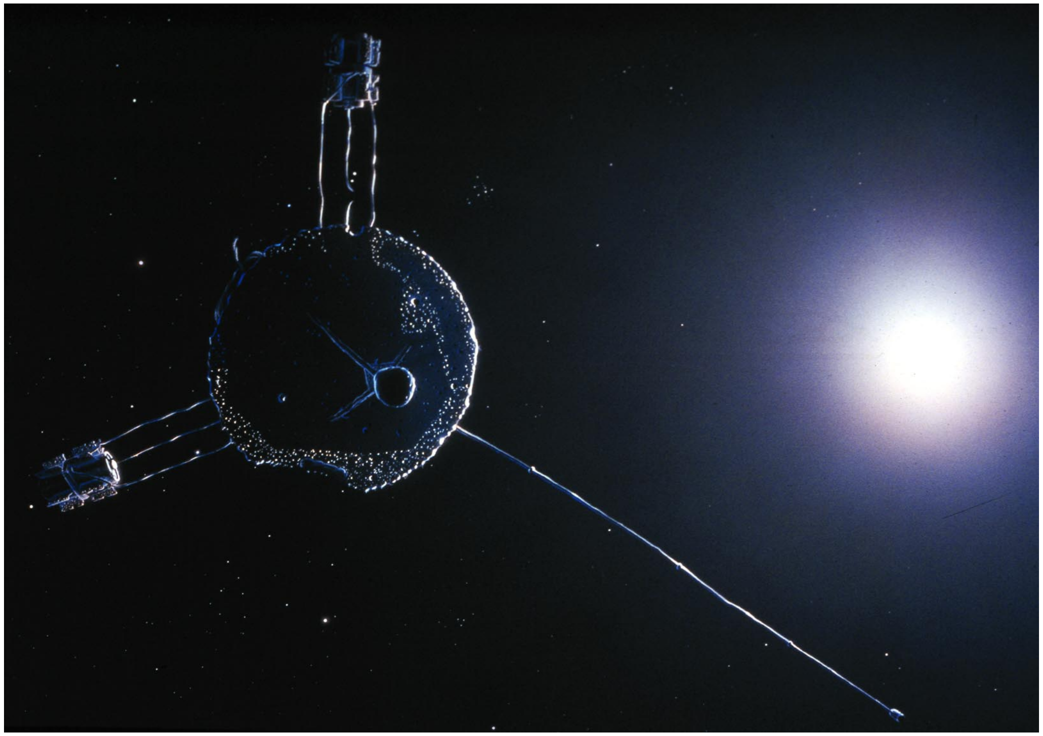
Probes are Already En Route to Distant Stars



Pioneer 10 as a relic, adrift and cold, passing through by a random star in the Milky Way

(© Astronomy Magazine)

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The Next Step ...

What is still needed next is another factor of 10 in speed, to ...

200 AU yr^{-1}, at which the first targeted interstellar crossing to Epsilon Eridani will take ~3400 years, the age of the Colossi of Memnon (Amehotep III - 18th dyn)

Though not ideal, the stars would be within our reach

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Implementing the Next Step

The target terminal speed is 200 AU/yr = 948 km/s

At an initial propellant fraction of 60%, the mass ratio is 2.5, the required specific impulse is 1.05×10^5s

To maximize the specific impulse, the propellant of choice is again LH_2

The specific impulse corresponds to an exhaust speed of 1035 km/s or H^+ accelerated through ~5.6 kV

$$x = g I_{sp} \frac{m_0}{\dot{m}} \left[1 - \frac{m_{final}}{m_0} \left(\ln \frac{m_0}{m_{final}} + 1 \right) \right]$$

X is the distance traveled

Issue is the sizing and specific mass of the power plant

Some type of nuclear energy is required

Example System

Assume 10 mg/s of H^+ = 960 A of current

=> 5.35 MW of electrical power required

Assume 50 year acceleration time

=> $m_{propellant}$ = 15,800 kg

m_0 = 26,300 kg

m_{final} = 10,500 kg

Assume 1.5 kg/kW => $m_{powerplant}$ = 8000 kg

=> $m_{payload}$ = 2500 kg

During acceleration (50 years), probe travels 4250 AU

Minimum size is set by reactor criticality, power processing, engines, propellant tansk

Required power and H_2 amounts comparable to manned Mars mission requirements

Ad Astra!