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Mission Design for the Innovative Interstellar Explorer Vision Mission

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The Innovative Interstellar Explorer, studied under a NASA Vision Mission grant, examined sending a probe to a heliospheric distance of 200 Astronomical Units (AU) in a "reasonable" amount of time. Previous studies looked at the use of a near-Sun propulsive maneuver, solar sails, and fission reactor powered electric propulsion systems for propulsion. The Innovative Interstellar Explorer's mission design used a combination of a high-energy launch using current launch technology, a Jupiter gravity assist, and electric propulsion powered by advanced radioisotope power systems to reach 200 AU. Many direct and gravity assist trajectories at several power levels were considered in the development of the baseline trajectory, including single and double gravity assists utilizing the outer planets (Jupiter, Saturn, Uranus, and Neptune). A detailed spacecraft design study was completed followed by trajectory analyses to examine the performance of the spacecraft design options.

Nomenclature

AU	Astronomical Unit, the mean distance between the Earth and Sun, 1.497959 x 10^8 km
EP	Electric Propulsion; use of a plasma or ion beam exhaust to propel a space vehicle
IIE	Innovative Interstellar Explorer; an interstellar-precursor, robotic science mission using REP
I_{SP}	Specific impulse; ratio of thrust to the weight flow rate of propellant
NASA	National Aeronautics and Space Administration
NEP	Nuclear Electric Propulsion; the use of a nuclear fission reactor for energy to power an EP system
REP	Radioisotope Electric Propulsion; use of decay of radioisotopes for energy to power an EP system
RPS	Radioisotope Power Source; a source of electricity that converts heat from radioisotope decay
RTG	Radioisotope Thermoelectric Generator; an RPS using the Seebeck effect
SRG	Stirling Radioisotope Generator; an RPS using a Stirling-cycle mechanical generator
TRL	Technology Readiness Level; a measure of technology maturity

I. Introduction

AN "interstellar precursor" mission has been under discussion in the science community for about 30 years[1,2,3,4,5,6,7]. The mission concept is relatively simple, yet difficult to accomplish: leave the solar system as rapidly as possible to reach the interstellar medium as soon as possible, and provide in situ measurements of the outer planetary and near interstellar space along the way. Detailed science objectives have been discussed with appropriate instrumentation[1,8]. The scientific goals of such a mission have varied little over these decades. The most recent formulation includes[1]: (1) explore the interstellar medium and determine directly the properties of the interstellar gas, the interstellar magnetic field, low-energy cosmic rays, and interstellar dust, (2) explore the influence of the interstellar medium on the solar system, its dynamics, and its evolution, (3) explore the impact of the solar system on the interstellar medium as an example of the interaction of a stellar system with its environment, and (4) explore the outer solar system in search of clues to its origin, and to the nature of other planetary systems. Given the desired distances involved for such a mission, here at least 200 Astronomical Units (AU), it is not surprising that the problem of implementing such a mission has always been one of propulsion, especially when one considers that a speed of 1 AU/yr is equivalent to 4.74 km/s.

Past concepts for such a mission have included the use of near-Sun powered gravity assists, with both chemical[6] and advanced high-thrust systems[9,10,11,12,13,14,15], nuclear electric propulsion (NEP)[2,3,4,16,17], and solar sails[18,19]. In this work, we detail the mission design studies for such a mission, dubbed the Innovative Interstellar Explorer (IIE) using radioisotope electric propulsion (REP)[20]. In an REP spacecraft, the power system mass is the major mass driver, and overall miniaturization, where possible, is paramount. The REP system can use any radioactive power supply (RPS) architecture. We assume that plutonium-238, the radioisotope used for the power supplies on Voyager, Galileo, Ulysses, and Cassini, will be used as the power source. A relatively high power output of at least 8 W/kg (specific mass of 125 kg/kW) is required. This performance can, in principle, be obtained with either an advanced radioisotope thermoelectric generator (RTG) or a "next-generation" Stirling radioisotope generator (SRG)[1].

To minimize the flight time to the interstellar medium, the outgoing asymptotic trajectory should be close to the direction of the incoming "interstellar wind"[21,22,23,24,25]. This direction, 252° right ascension and +7° declination in Earth ecliptic coordinates defines the optimal aim point for the trajectory. Given the variability in the interaction region[26], however, a targeted trajectory within ~20° of this point will suffice. In particular, with this less stringent requirement, by remaining close to the plane of the ecliptic, the trajectory can be better optimized and also have a somewhat larger set of backup windows. The final requirement was thus to reach a point within 20° of the incoming interstellar wind direction 200 AU from the Sun "as fast as possible."

II. Mission Architecture

Previous REP trajectory designs[27,28,29,30,31,32] showed that because of the low-acceleration capability of REP a certain mission architecture is optimal for outer solar system missions. This approach, consisting of a high-excess escape energy (C_3) launch from Earth ($C_3 \geq 100$ km^2/s^2) followed by a long period of electric propulsion (EP) thrusting, has been shown to allow rendezvous of a small class spacecraft (dry mass less than 1000 kg) with many bodies throughout the outer solar system[32]. Because of the similarities between the IIE mission and previously studied outer solar system missions, and because the IIE requires the minimum trip time to 200 A.U., a similar mission architecture was chosen.

The launch architecture chosen for the IIE provides more capability, but at a higher cost, than the previous studies[27,28,29,30,31,32]. The outer solar system trajectories utilized an Atlas V 551 launch vehicle with a Star 48V upper stage to provide the required high C_3. To minimize the IIE trip time, a Delta IV Heavy launch vehicle stacked with two solid propellant upper stages was used. The early studies that explored the mission trade space, utilized a Star 48/Star 37 stack, while the Advanced Project Design Team Studies upgraded to two Star 48A upper stages. Figure 1

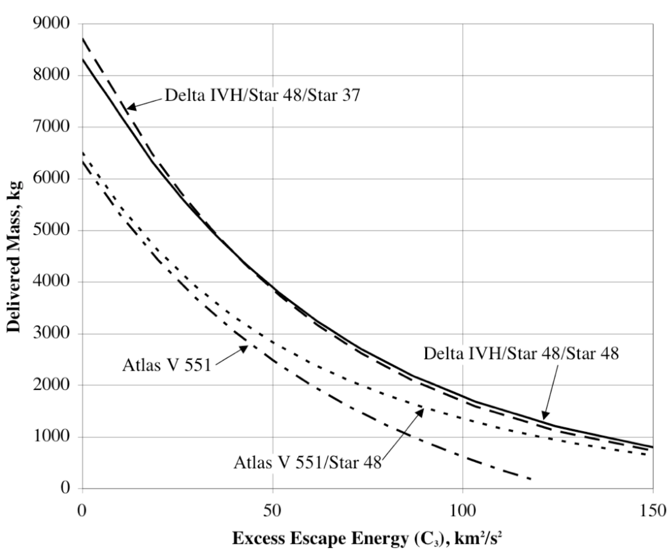

Figure 1. Comparison of Launch Architectures.

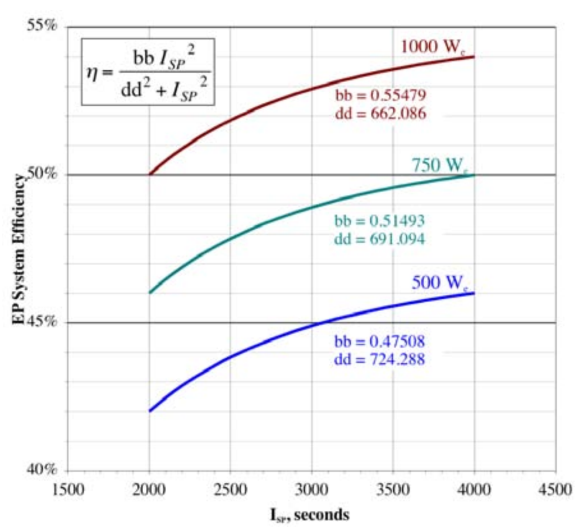

Figure 2. EP System Efficiency vs. I_{SP}.

displays the differences between the capabilities of these launch architectures[33,34,35,36].

The EP system, which provides a significant proportion of the in-space Δv, was included in the optimization by means of a simple EP model. This EP-system model used a theoretical performance model based on current best estimates of performance of low-power EP systems. The performance model, shown in Figure 2, relates efficiency to specific impulse (I_{SP}) at power levels between 500 W_e and 1000 W_e into the EP system. These curves are representative of gridded-ion thrusters or Hall thrusters at these power levels. The specific thruster technology was chosen after an optimal I_{SP} was determined for the mission. In all cases, the gridded-ion thruster was the technology of choice for the IIE mission because the I_{SP} optimized too high for Hall thrusters to be considered feasible. The EP system mass model was based on mass estimates of a low-power gridded-ion propulsion system, with heritage from the NSTAR[37] and NEXT[38] programs, that is currently unfunded and, as a result, not developed. The power level chosen (1000 W_e into the EP system) was chosen based on a trade of power level versus the number of RPS units that could reasonably be placed on a spacecraft. Early trajectory trades showed that power levels around 1000 W_e seemed to be optimal based on mass estimates of the RPS units.

III. Pathfinder Studies

To begin the analysis on a purely trajectory-oriented basis, a study was conducted that analyzed many different trajectory options over many years of launch opportunities. A simple spacecraft model representing the IIE was generated with a dry mass of 519 kg and 1000 W_e of power for propulsion. This dry mass was the final mass target for the trajectory analyses with the trip time minimized for each case. While only a simple mass model of the IIE spacecraft, it allowed analyses of a wide range of trajectories without the need to redesign the spacecraft for each case. This study of the trajectory design space enabled future analyses to be conducted more efficiently because the wide trajectory trade space was understood.

A. Design Space

The trajectory design space included a wide range of launch dates and trajectory types. Because various planetary flyby trajectories were planned for study and the final right ascension and declination were constrained, launch dates between 2010 and 2050 were considered. Minimum trip time trajectories that included single and double flybys of the outer planets (Jupiter, Saturn, Uranus, and Neptune) as well as direct trajectories were designed throughout the range of launch dates. No inner solar system gravity assists were considered to eliminate additional (and potentially massive) thermal requirements on the spacecraft design.

The sidereal period of the flyby planets and the constrained final right ascension and declination limit the launch opportunities for each trajectory type. Figure 3 shows the repetition patterns of each trajectory type studied. Note that the times and numbers of revolutions in Figure 3 are between time-optimized launch opportunities. Non-time-optimal trajectories can be found in the intervening years between time-optimal launch opportunities. Because of its mass, compared to the other outer planets, Jupiter provides the highest Δv to the spacecraft, followed by Saturn, Neptune, and Uranus. The gravity assist maneuvers also bend the spacecraft trajectory depending on the gravity assist altitude and incoming velocity, which adds some flexibility in trajectory design, allowing the gravity assist Δv to be traded for other parameters such as launch date, gravity assist date and total trip time.

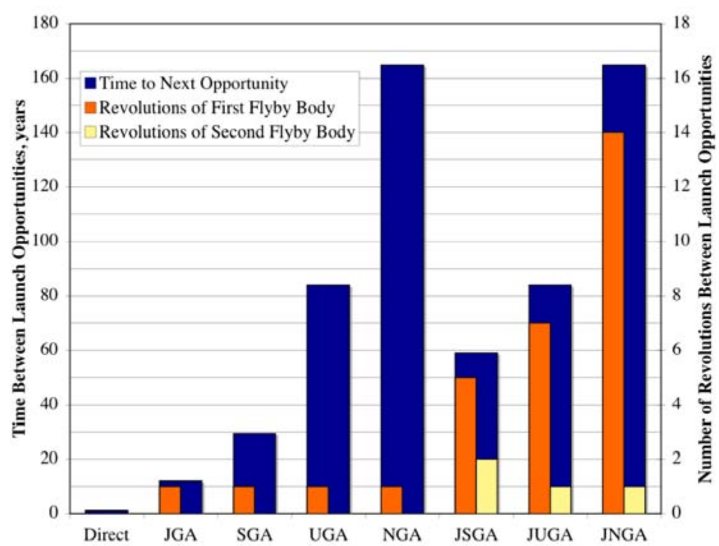

Figure 3. Trajectory Opportunity Repetition.

B. Trajectory Design Trades and Selection

Three types of trajectories were studied; direct trajectories, single gravity assist trajectories, and double gravity assist trajectories. The direct trajectories were characterized by their high-energy launches followed by long thrusting periods that propelled the spacecraft to 200 AU. The single gravity assist trajectories utilized a high-energy launch followed by a gravity assist at Jupiter, Saturn, Uranus or Neptune. The double gravity assist trajectories also launched to a high energy Earth escape that was followed by a Jupiter gravity assist and a gravity assist at Saturn, Uranus, or Neptune. Figure 4 shows the trip times and launch years of the primary minimum-time trajectories. Other non-minimum time trajectories are available throughout the launch windows that extend from each of the launch dates in Figure 4. A summary of pertinent parameters for each of the trajectory types studied is presented in Table 1 and discussed throughout the remainder of this section.

The direct trajectory was the simplest and most flexible of the trajectories studied. It required a C_3 of approximately 100 km^2/s^2, made one revolution around the Sun to achieve solar system escape velocity, and then proceeded to the heliospheric nose (see Figure 5). (Note that in all trajectory plots presented herein, a solid spacecraft trajectory line indicates thrusting while a dashed line indicates a coast, and each plot contains tick marks spaced in time by one-year intervals.) This trajectory was the most flexible because it did not require a planetary gravity assist, which allowed it to be launched in any year. However, this trajectory performed poorly with a trip time to 200 AU of 46.5 years with a lightweight spacecraft model, but its velocity was twice that of the Voyager 2 spacecraft (see the column labeled “Direct” in Table 1).

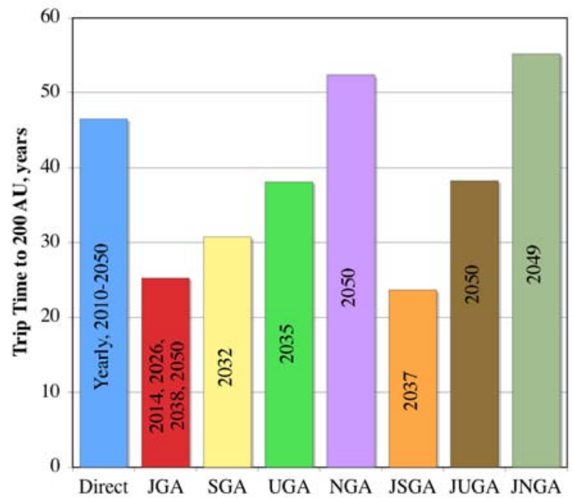

Figure 4. Performance of the Different Trajectories, Superimposed Minimum Trip Time Launch Opportunity Years.

Table 1. Summary of Trajectories Parameters

Trajectory Type	Direct	JGA	SGA	UGA	NGA	JSGA	JUGA	JNGA
Launch Date	January 17, 2010	October 26, 2014	August 26, 2032	August 14, 2035	June 23, 2050	October 3, 2037	January 23, 2050	December 18, 2049
First Gravity Assist Body		Jupiter	Saturn	Uranus	Neptune	Jupiter	Jupiter	Jupiter
Gravity Assist Date		December 7, 2015	April 11, 2035	August 11, 2042	May 2, 2063	November 7, 2038	December 19, 2056	June 6, 2055
Gravity Assist Altitude		45388 km	3027 km	1286 km	1261 km	119726 km	1059149 km	965858 km
Gravity Assist Radius		1.63 Rj	1.05 Rj	1.05 Rj	1.05 Rj	2.67 Rj	15.81 Rj	14.51 Rj
Gravity Assist Δv		28.0 km/s	19.9 km/s	13.8 km/s	13.3 km/s	24.3 km/s	9.6 km/s	10.2 km/s
First Gravity Assist Body						Saturn	Uranus	Neptune
Gravity Assist Date						December 20, 2039	July 6, 2062	January 21, 2071
Gravity Assist Altitude						8393 km	1286 km	1261 km
Gravity Assist Radius						1.14 Rj	1.05 Rj	1.05 Rj
Gravity Assist Δv						23.5 km/s	14.2 km/s	14.5 km/s
Burnout Date	March 31, 2036	November 2, 2029	September 11, 2051	June 22, 2060	September 24, 2085	November 23, 2051	December 3, 2069	March 13, 2103
Burnout Distance	66 AU	103 AU	99 AU	95 AU	82 AU	102 AU	61 AU	101 AU
Burnout Speed	6.6 AU/year	9.5 AU/year	8.8 AU/year	8.1 AU/year	7.5 AU/year	10.3 AU/year	7.7 AU/year	8.6 AU/year
Date 200 AU Reached	July 8, 2056	January 12, 2040	May 23, 2063	September 13, 2073	October 26, 2102	June 1, 2061	April 8, 2088	February 20, 2105
Trip Time to 200 AU	46.5 years	25.2 years	30.8 years	38.1 years	52.4 years	23.7 years	38.2 years	55.2 years
Speed at 200 AU	6.6 AU/year	9.5 AU/year	8.7 AU/year	8.1 AU/year	7.5 AU/year	10.2 AU/year	7.6 AU/year	8.6 AU/year
Azimuth at 200 AU	235.0°	254.8°	231.8°	231.8°	237.5°	263.5°	267.7°	232.2°
Elevation at 200 AU	0.0°	0.6°	7.4°	7.2°	21.3°	1.2°	3.3°	3.4°
Launch Mass	1880 kg	916 kg	1387 kg	1402 kg	1307 kg	900 kg	1803 kg	1541 kg
Propellant Mass	1361 kg	397 kg	868 kg	883 kg	788 kg	381 kg	1284 kg	1022 kg
Final Mass	519 kg	519 kg	519 kg	519 kg	519 kg	519 kg	519 kg	519 kg
Power	1.0 kW	1.0 kW	1.0 kW	1.0 kW	1.0 kW	1.0 kW	1.0 kW	1.0 kW
I_{SP}	2563 s	3654 s	2748 s	3133 s	3981 s	3616 s	2279 s	4304 s
EP System Efficiency	52.0%	53.7%	52.4%	53.1%	54.0%	53.7%	51.2%	54.2%
C_3	103.9 km^2/s^2	152.4 km^2/s^2	124.4 km^2/s^2	123.7 km^2/s^2	128.4 km^2/s^2	153.6 km^2/s^2	106.7 km^2/s^2	117.3 km^2/s^2
EP Δv	32.4 km/s	20.3 km/s	26.5 km/s	30.5 km/s	36.1 km/s	19.5 km/s	27.8 km/s	45.9 km/s
Thrust Time	26.2 years	15.0 years	19.1 years	24.9 years	35.3 years	14.1 years	19.9 years	53.3 years

Of the single gravity assist trajectories, the Jupiter Gravity Assist (JGA) trajectory had the shortest trip time, and second shortest trip time of all of the trajectories studied (see the columns labeled “JGA”, “SGA”, “UGA”, and “NGA” in Table 1). For the Jupiter, Saturn, Uranus, and Neptune gravity assist trajectories, the trip times were 25.2, 30.8, 38.1 and 52.4 years, respectively. These trajectories all require a high-C_3 launch (between 120 and 150 km^2/s^2), followed by a thrusting period to reach the gravity-assist body, followed first by the planetary gravity assist that provides a Δv of between 13.3 and 28.0 km/s, then by a long thrusting period until the propellant supply is depleted, and finally by a long coast to 200 AU (see Figure 6). Only the JGA trajectory could accommodate more than one minimum-time launch opportunity during the 2010 – 2050 study window due to Jupiter’s relatively short sidereal period of 11.9 years. The JGA trajectory can be launched in 2014, 2026, 2038, or 2050 and achieve performance

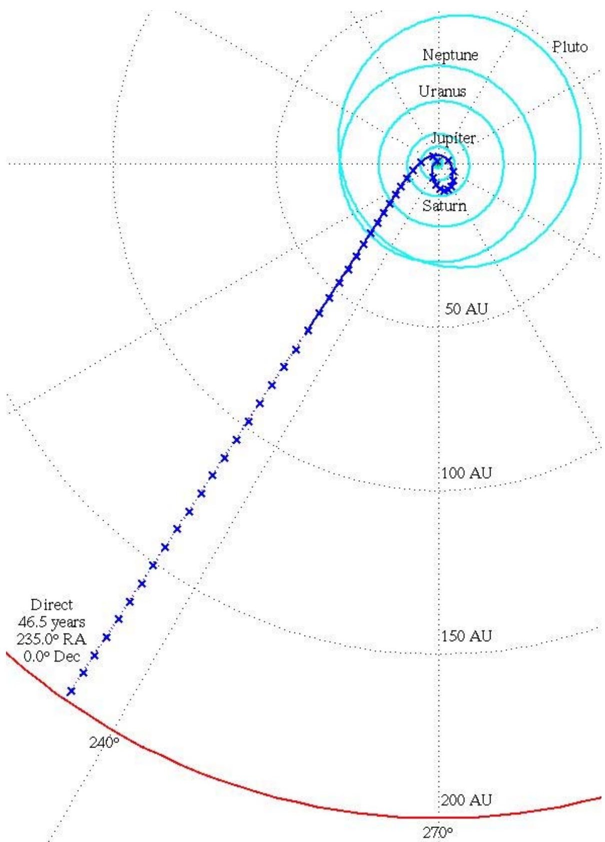

Figure 5. Direct Trajectory.

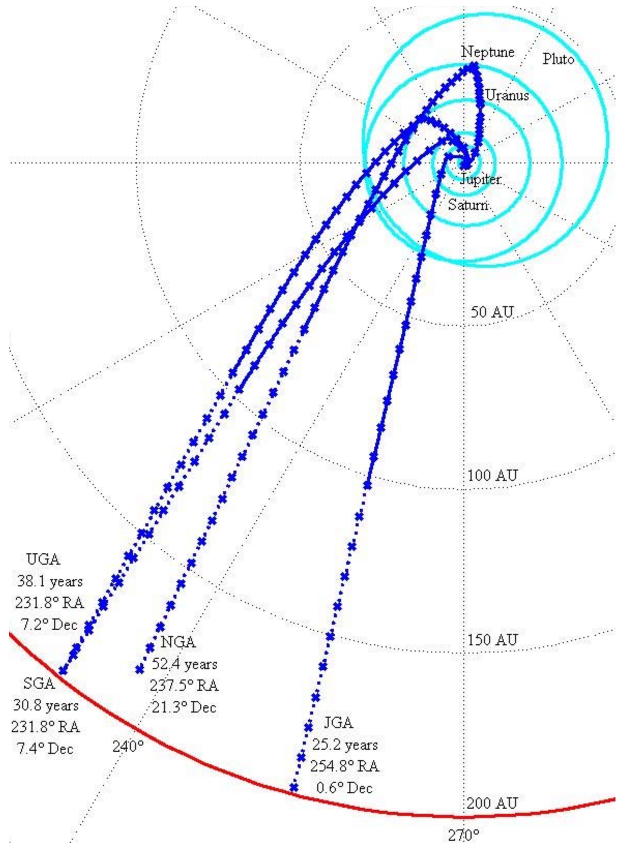

Figure 6. Single Gravity Assist Trajectories.

similar to that presented in Table 1. The other trajectories can accommodate launch opportunities in several years before and after the minimum-time launch year, due to the long periods of these outer planets, at the expense of longer trip time. Note that during the 2010-2050 time period, Neptune is in the wrong part of its orbit to provide any benefit. To affect a Neptune gravity assist, the spacecraft must travel approximately 30 AU away from the heliospheric nose, perform the Neptune gravity assist, and then traverse back through the solar system to reach 200 AU within the tolerance of the right ascension and declination targets.

The double gravity assist trajectories performed equivalently to the single gravity assist trajectories, except in the case of the Jupiter-Saturn Gravity Assist (JSGA) trajectory (see the columns labeled "JSGA", "JUGA", and "JNGA" in Table 1). Again, high energy launches were required for all trajectories (106 km^2/s^2 < C_3 < 154 km^2/s^2), followed by thrusting periods to achieve each of the gravity assists, a long period of thrusting until all propellant is exhausted, and finally a long coast period to reach 200 AU (see Figure 7). The trip times for the minimum-time double-gravity assist-trajectories are 23.7, 38.2, and 55.2 years for the Jupiter-Saturn, Jupiter-Uranus, and Jupiter-Neptune gravity-assist trajectories, respectively. The JSGA trajectory has the shortest trip time of all trajectories studied, however only one opportunity exists to launch on this trajectory during the 40-year window that was examined. The Jupiter-Uranus and Jupiter-Neptune gravity assist trajectories' launch dates could be adjusted by Jupiter's sidereal period, because of the long sidereal periods of Uranus and Neptune, but penalties in trip time would be realized. Again, as with the single gravity assist trajectories, Neptune is in the wrong part of its orbit to provide any benefit in trip time during the 40-year study window.

The JGA trajectory was chosen for further study because of its near-minimum trip time, as compared to the other trajectories studied, and its minimum trip time launch window repetition every 11.9 years (Jupiter's sidereal period). Each minimum trip time launch window is approximately 20-25 days long (±10-12 days) for a 6-month trip time penalty, and could be extended for additional trip time penalties. Around each minimum trip time launch opportunity are other launch opportunities with minimal trip time penalties each spaced by approximately 13 months (see Figure 9). These characteristics made the JGA trajectory the most feasible choice for further study.

Some room for improvement exists in the design of these trajectories without stretching the realm of technologies that could be available. One improvement would be a liquid upper stage to replace the two solid Star motors. This upper stage would have a higher I_{SP} and thus provide more mass to the same required C_3. Spacecraft

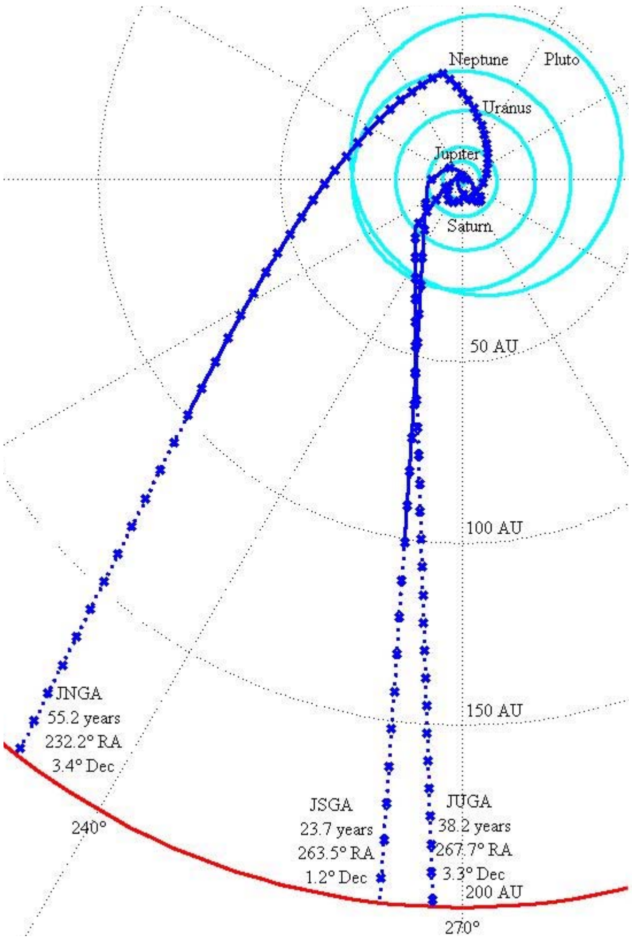

Figure 7. Double Gravity Assist Trajectories.

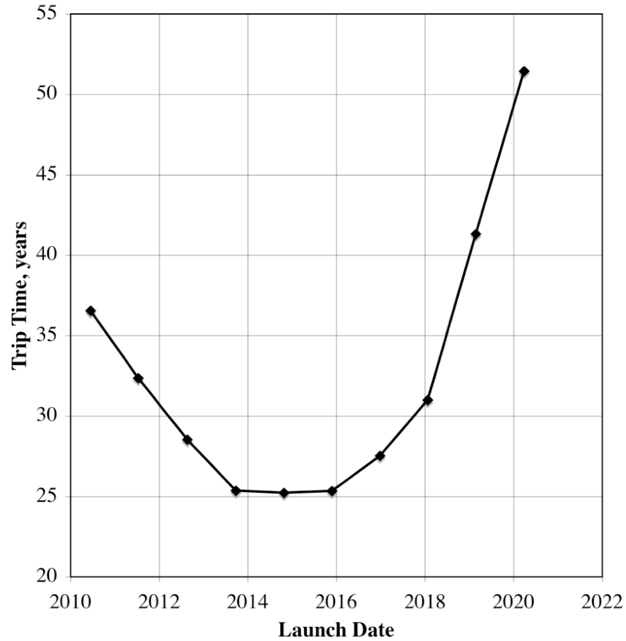

Figure 9. Jupiter Gravity Assist Trajectory Trip Time vs. Launch Date

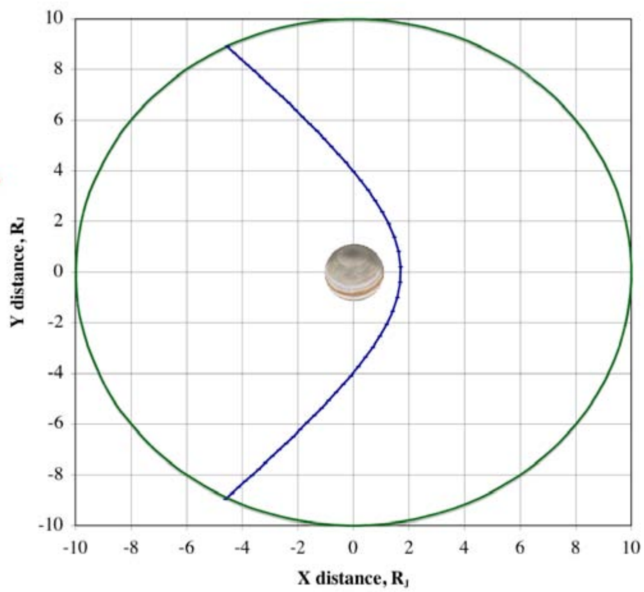

Figure 8. Jupiter Flyby Trajectory Inside 10 R_J with 15-minute Interval Tick-Marks

mass reductions would also improve the performance of these trajectories. Mass reductions could come in the form of technology improvements (higher RPS specific power) or design improvements (lighter structure, innovative spacecraft design).

C. Jupiter Gravity Assist Design and Considerations

The Jupiter gravity assist, used in many of these trajectories, adds complexity to the design. Not only does the JGA add constraints to the launch windows, but it also introduces a very inhospitable radiation environment to be traversed by the spacecraft[39]. To attain the maximum effect of the gravity assist, the minimum flyby radius was constrained at 1.05 R_J, at the expense of time in the radiation belts. Figure 8 shows the JGA trajectory's Jupiter encounter within 10 R_J (0.5 R_J outside of Europa) with tick-marks indicating time intervals of 15 minutes. In this case, the spacecraft reaches approximately 1.7 R_J, receives a Δv of approximately 28 km/s, and has its trajectory turned by approximately 90°. For this Jupiter encounter, a total of approximately 13 hours are spent inside 10 R_J, where most of the radiation exposure would occur. No analysis was performed to estimate the radiation dose or the amount of shielding needed on the spacecraft to protect sensitive components during the Jupiter encounter as part of this study. Note that this additional radiation shielding design consideration is not pertinent to the other outer planets due to their lack of significant radiation environments.

IV. Advanced Project Design Team Studies

To add more detail to the spacecraft design used in these pathfinder studies, a study was conducted with the Jet Propulsion Laboratory's (JPL's) Advanced Project Design Team (Team-X). The goal of this study was to create detailed spacecraft designs using existing and in-development technologies while adding sufficient margins/reserves according to JPL's Design Principles. The technology cut-off date for these studies is 2010 (technology must be at a technology readiness level (TRL) of 6), sufficient to provide the necessary technology for a 2014 launch date.

A. Spacecraft Design Options

Four options were investigated during the study. All options used the same architecture configuration, subsystem design, and baseline (JGA) mission design. Different technology and data rate assumptions drove the design of the 4 options. The baseline design (Option 1) relies on current state-of-the-art technology and does not make any aggressive technology assumptions, except for the power system – an advanced, high-temperature RTG. A downlink data rate of 5.8 kbps from 200 AU is assumed. This rate is sufficient to downlink data collected continuously at a rate of 500 bps with two downlinks of ~7 hours per week to 180 phased 12-m antennas operating at Ka-band. The spacecraft has a 2.1-m diameter high gain antenna and carries three 1-kW ion thrusters, one being a spare. Four, fully redundant command and data subsystems (CDS) are used to deal with reliability questions for a ~30-year flight time.

The second study option is a delta from the baseline that investigates more aggressive technology and redundancy assumptions. Only two CDS strings and two thrusters are included. The high-gain antenna is increased to a 3-m diameter to compensate for other (mass and power reducing) system changes. The option 2 spacecraft is presented in Figure 10[1]. The spacecraft design of the other options is based on this spacecraft with modifications based on the technology and redundancy choices made for each option.

Option 3 studies whether reducing the return data rate to 500 bps (from 5.8 kbps in Options 1 and 2) saves significant mass and power and, hence, reduces trip time. The decreased data rate only saves around 20 kg of dry mass on the spacecraft from the baseline design.

Option 4 combines the aggressive technology in Option 2, the reduced data rate of Option 3, and a reduced ion thruster power that ultimately results in a dry mass 170 kg less than that of the baseline design.

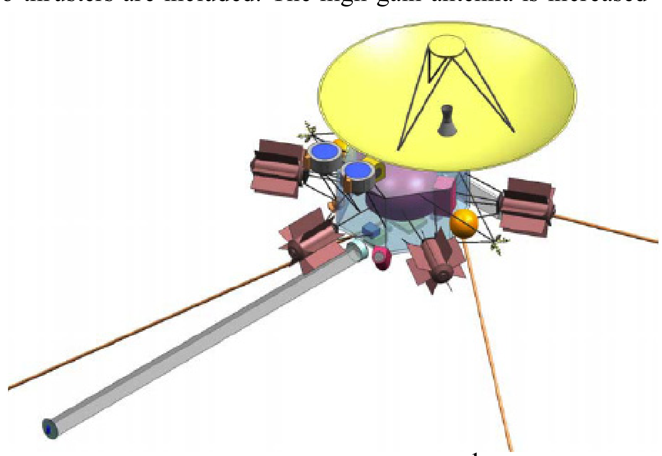

Figure 10. Option 2 Spacecraft Design[1].

Table 2. Option Trades for Spacecraft System Design.

	Option 1 Mass	Option 2 Mass	Option 3 Mass	Option 4 Mass	Subsystem Contingency	CBE+ Contingency	Mode 1 Power	Mode 2 Power	Mode 3 Power	Mode 4 Power	Mode 5 Power	Mode 6 Power
Description	**Baseline**	**Aggressive Technology Assumptions**	**Reduced Downlink Rate**	**Combination of Options 2 and 3**			**Safing**	**Telecom after EP Burnout**	**Engine-off Cruise**	**Engine-on Cruise 10 AU to EP Burnout**	**Launch**	**Telecom before EP Burnout**
Payload												
Instruments	35 kg	35 kg	35 kg	35 kg	30%	46 kg	9.1 W	29.4 W	29.4 W	29.4 W	0.0 W	29.4 W
Payload Total	**35 kg**	**35 kg**	**35 kg**	**35 kg**	**30%**	46 kg	**9.1 W**	**29.4 W**	**29.4 W**	**29.4 W**	**0.0 W**	**29.4 W**
Bus												
Attitude Control	15 kg	7 kg	15 kg	7 kg	21%	18 kg	9.0 W	36.0 W	36.0 W	36.0 W	40.0 W	36.0 W
Command & Data	26 kg	14 kg	26 kg	14 kg	30%	34 kg	43.0 W	43.0 W	43.0 W	43.0 W	43.0 W	43.0 W
Power	182 kg	182 kg	182 kg	155 kg	30%	237 kg	10.1 W	46.0 W	10.4 W	11.4 W	8.2 W	46.5 W
Propulsion1	81 kg	62 kg	81 kg	59 kg	20%	97 kg	0.7 W	0.7 W	0.7 W	0.7 W	0.7 W	0.7 W
Propulsion2	10 kg	10 kg	10 kg	10 kg	18%	12 kg	41.0 W	41.0 W	1.0 W	1.0 W	1.0 W	41.0 W
Structures & Mechanisms	126 kg	110 kg	124 kg	99 kg	30%	164 kg	0.0 W	0.0 W	0.0 W	0.0 W	0.0 W	0.0 W
Cabling	38 kg	30 kg	37 kg	28 kg	30%	49 kg						
Telecomm	23 kg	24 kg	21 kg	21 kg	20%	28 kg	17.0 W	522.6 W	17.0 W	17.0 W	17.0 W	517.0 W
Thermal	48 kg	42 kg	38 kg	35 kg	30%	62 kg	34.5 W	34.5 W	32.0 W	47.5 W	23.8 W	47.5 W
Bus Total	**549 kg**	**481 kg**	**535 kg**	**428 kg**	**28%**	701 kg	**155.2 W**	**723.8 W**	**140.0 W**	**156.5 W**	**133.6 W**	**731.6 W**
Spacecraft Total (Dry)	**585 kg**	**516 kg**	**570 kg**	**463 kg**	**28%**	747 kg	**164.3 W**	**753.2 W**	**169.4 W**	**185.9 W**	**133.6 W**	**761.0 W**
Subsystem Heritage Contingency	162 kg	145 kg	158 kg	129 kg								
System Contingency	13 kg	10 kg	13 kg	10 kg			49.3 W	225.9 W	50.8 W	55.8 W	40.1 W	228.3 W
Spacecraft with Contingency	**760 kg**	**671 kg**	**741 kg**	**602 kg**			**213.6 W**	**979.1 W**	**220.3 W**	**241.7 W**	**173.6 W**	**989.4 W**
Xenon Propellant	459 kg	450 kg	461 kg	394 kg								
Hydrazine Propellant	31 kg	31 kg	31 kg	31 kg								
Spacecraft Total (Wet)	**1250 kg**	**1151 kg**	**1232 kg**	**1026 kg**								

The high gain antenna is 2.1-m in diameter, 2 CDS strings are used, and two 750 W ion thrusters are used for propulsion.

In each case, power requirements for six operational power modes were evaluated and design reserves/margins were applied in accord with the technology readiness levels and design rules used by JPL's Team-X. Also, the overall mission design was re-optimized in each case assuming a "best launch date" in 2014 and a Jupiter gravity assist. Details for each option at the system level are given in Table 2.

B. Trajectory Performance Using Advanced Project Design Team Spacecraft Design

Because more mass is required to be delivered to 200 AU than in the pathfinder studies, the trip times for the design options in Table 2 are longer. Using the spacecraft designs and technology assumptions in Options 1, 2, 3, and 4, the respective trip times are 31.1 years, 29.7 years, 30.7 years and 29.9 years (see Table 3). Optimization of each system-design option yields trajectories with a launch date in October of 2014 and that arrive at 200 AU between June 2044 and November of 2045 with Jupiter gravity assists in either January or February of 2016. The parameters that affect the trajectory were all held constant for options 1 through 3, except for the final mass (dry mass) delivered to 200 AU, explaining the trip time differences between these options. Option 4 also decreased the power available to the EP system. This change decreased the level of acceleration the EP system could provide and also decreased the dry mass of the spacecraft, which resulted in a similar trip time to option 2. Each of these options result in a final velocity relative to the Sun of 7.6 to 8.0 AU/year (nearly 2.5 times the speed of Voyager 2, now 3.3 AU/yr and more than twice that of Voyager 1, now about 3.6 AU/yr) and all within 15° of the target right ascension and declination.

Table 3. Trajectory Trades for Spacecraft System Designs.

	Option 1	Option 2	Option 3	Option 4
Launch Date	October 22, 2014	October 22, 2014	October 22, 2014	October 23, 2014
Gravity Assist Body	Jupiter	Jupiter	Jupiter	Jupiter
Gravity Assist Date	February 13, 2016	January 29, 2016	February 10, 2016	January 16, 2016
Gravity Assist Altitude	79131 km	71970 km	77736 km	65629 km
Gravity Assist Radius	2.11 R_J	2.01 R_J	2.09 R_J	1.92 R_J
Gravity Assist Δv	23.3 km/s	24.2 km/s	23.5 km/s	25.0 km/s
Burnout Date	April 15, 2033	June 19, 2032	February 4, 2033	October 16, 2032
Burnout Distance	105 AU	104 AU	104 AU	107 AU
Burnout Speed	7.6 AU/year	8.0 AU/year	7.7 AU/year	7.9 AU/year
Date 200 AU Reached	November 4, 2045	June 11, 2044	July 11, 2045	September 10, 2044
Trip Time to 200 AU	31.1 years	29.7 years	30.7 years	29.9 years
Speed at 200 AU	7.6 AU/year	8.0 AU/year	7.7 AU/year	7.8 AU/year
Azimuth at 200 AU	265.1°	262.8°	264.7°	261.2°
Elevation at 200 AU	0.0°	0.0°	0.0°	0.0°
Launch Mass	1281 kg	1193 kg	1265 kg	1068 kg
Xenon Propellant Mass	459 kg	450 kg	461 kg	394 kg
Final Mass	843 kg	758 kg	824 kg	686 kg
Power	1.0 kW	1.0 kW	1.0 kW	0.75 kW
I_{SP}	3862 s	3789 s	3830 s	3524 s
EP System Efficiency	53.9%	53.8%	53.9%	49.6%
Total Stack C_3	120.6 km²/s²	125.8 km²/s²	121.5 km²/s²	132.7 km²/s²
Delta IV H C_3	16.3 km²/s²	17.1 km²/s²	16.5 km²/s²	18.1 km²/s²
Delta IV H Launch Mass	6906 kg	6803 kg	6887 kg	6678 kg
EP Δv	15.9 km/s	16.8 km/s	16.1 km/s	15.3 km/s
Thrust Time	18.5 years	17.7 years	18.3 years	18.0 years

V. Conclusion

A mission beyond the edge of the solar system to interstellar space has been a desire of the science community for decades, and achieving the science goals in a "reasonable" amount of time has been a challenge as shown in previous studies. This study explores the trajectory trade space through analysis of direct, single-gravity-assist, and double-gravity-assist trajectories to 200 AU. The trajectory chosen as the baseline for this study, the Jupiter gravity-assist trajectory, has one of the shortest trip times of those trajectories studied and the most flexible launch opportunities. This baseline trajectory, flown with a light small spacecraft (~520 kg dry mass), could reach 200 AU within the right ascension and declination constraints in approximately 25 years. A spacecraft design study was conducted to add considerations of technology readiness, margins, and physical layout of the spacecraft systems with a technology cut-off date of 2010 (TRL 6). This design study resulted in spacecraft dry masses approximately 140-320 kg higher than used in the initial trajectory trade study. This mass increase results in trip times approximately 5 years longer for a total of approximately 30 years to reach 200 AU. Whether this trip time is "reasonable" will have to be determined by the science community and programmatic considerations. The twin Voyager spacecraft have been flying for over 25 years and have a potential lifetime of 15 years more until their decaying RTG power output can no longer run them. Following their "Grand Tour" of the outer planets, the spacecraft have remained at the scientific cutting edge while continuing to excite the public imagination. The Innovative Interstellar Explorer would be a worthy successor to the Voyagers, and their predecessors Pioneers 10 and 11, in taking the first scientific step to the stars. The required technology to reach 200 AU and the interstellar medium either exists or can be developed in time for a 2014 launch with the proper interest, funding, and commitment to scientific discovery and the next generation of space explorers.

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The Innovative Interstellar Explorer, studied under a NASA Vision Mission grant, examined sending a probe to a heliospheric distance of 200 Astronomical Units (AU) in a "reasonable" amount of time. Previous studies looked at the use of a near-Sun propulsive maneuver, solar sails, and fission reactor powered electric propulsion systems for propulsion. The Innovative Interstellar Explorer's mission design used a combination of a high-energy launch using current launch technology, a Jupiter gravity assist, and electric propulsion powered by advanced radioisotope power systems to reach 200 AU. Many direct and gravity assist trajectories at several power levels were considered in the development of the baseline trajectory, including single and double gravity assists utilizing the outer planets (Jupiter, Saturn, Uranus, and Neptune). A detailed spacecraft design study was completed followed by trajectory analyses to examine the performance of the spacecraft design options.

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