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Mission Design for the Innovative Interstellar Explorer Vision Mission

Douglas I. Fiehler*
QSS Group, Inc., Cleveland, Ohio, 44135
and
Ralph L. McNutt Jr.†
Johns Hopkins University, Laurel, Maryland, 20723

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The Innovative Interstellar Explorer mission was studied under a NASA Vision Mission grant and its goal is to send a probe to a heliospheric distance of 200 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.

I. Introduction

AN "interstellar precursor" mission has been under discussion in the science community for about 30 yr [1–7]. 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 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 [5,9,10] (AU, the mean distance between the Earth and Sun, 1.497959×10^8 km), 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.

In this work, we detail the mission design studies for an interstellar explorer mission, dubbed the Innovative Interstellar Explorer (IIE) using radioisotope electric propulsion (REP) [11]. In a 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 source (RPS) architecture. We assume that plutonium-238, the radioisotope used for the power supplies on Voyager, Galileo, Ulysses, and Cassini, is 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].

Past concepts for this mission have included the use of near-Sun powered gravity assists, with both chemical [6] and advanced high-thrust systems [12–18], nuclear electric propulsion [2–4,19,20], and solar sails [9,10,21,22]. However, direct comparison of these studies to the REP study discussed here is difficult because of the assumptions made in each study. These other studies examined the use of advanced technologies needing significant development, whereas in this study an attempt was made to limit technology development to those technologies that are enabling for REP (primarily the radioisotope power and electric propulsion systems), and technologies with higher readiness were used for the other systems. Assuming advanced technologies push out the earliest possible launch date of this mission by a decade or more in some cases, and forces the study to estimate the performance of the advanced systems, making them difficult to directly compare to REP.

To minimize the flight time to the interstellar medium, the outgoing asymptotic trajectory should be close to the direction of the incoming interstellar wind [23–27]. This direction, 252 deg right ascension and +7 deg declination in Earth ecliptic coordinates defines the optimal aim point for the trajectory. Given the variability in the interaction region [28], however, a targeted trajectory within ~20 deg 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 deg of the incoming interstellar wind direction 200 AU from the Sun "as fast as possible."

II. Mission Architecture

Previous REP trajectory designs [29–34] 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

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*Mission Systems Engineer, NASA Glenn Research Center, Space Propulsion and Mission Analysis Office, 21000 Brookpark Road, M.S. 500-103. AIAA Member.

†Principle Professional Staff, Applied Physics Laboratory, Space Department, 11100 Johns Hopkins Road, MP3-E116. AIAA Member.

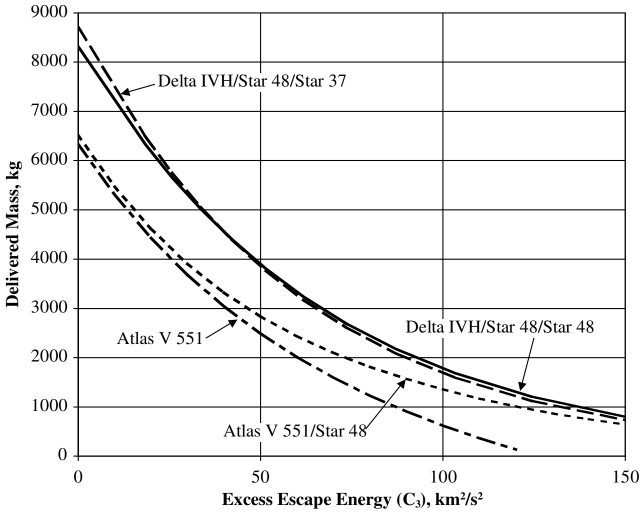

Fig. 1 Comparison of launch architectures.

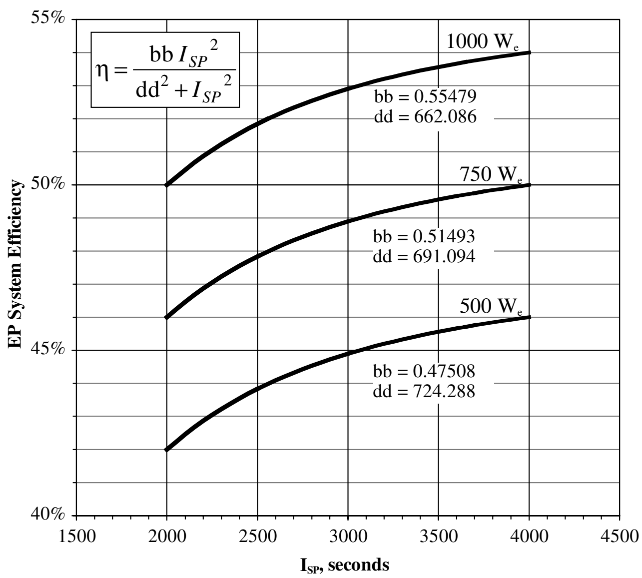

Fig. 2 EP-system efficiency vs specific impulse.

spacecraft (dry mass less than 1000 kg) with many bodies throughout the outer solar system [34]. 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 AU, 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 [29–34]. The outer solar system trajectories used 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, used a Star 48/Star 37 stack, while the Advanced Project Design Team Studies upgraded to two Star 48A upper stages. Figure 1 displays the differences between the capabilities of these launch architectures [35–38].

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 uses a theoretical performance model based on current best estimates of performance of low-power EP systems. The performance model, shown in Fig. 2, relates efficiency to specific impulse (I_{SP}) at power levels between $500W_e$ and $1000W_e$ 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 is 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 NASA Solar Electric Propulsion Technology Application Readiness (NSTAR) [39] and NASA Evolutionary Xenon Thruster (NEXT) [40] programs. The power level chosen ($1000W_e$ 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 $1000W_e$ seemed to be optimal based on mass estimates of the RPS units. The EP-system lifetime required is approximately 15 yr for the minimum trip time trajectories. While this lifetime requirement exceeds current thruster lifetimes, it is achievable by means of operating several thrusters serially, derating the thruster operating points, and modifications of the thruster design, primarily using more sputter-resistant materials in critical components.

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 $1000W_e$ 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. A Venus flyby was not considered because it impacts the thermal design of the spacecraft as the thermal heat flux input at Venus's orbit is ~1.9 times that at Earth. This raises the radiator temperatures of the RTGs, increasing the complexity of the power subsystem design while also introducing constraints for additional thermal radiating potential for waste heat around the spacecraft. As this mission is designed to go away from the Sun and mass is constrained, the design of the thermal system is already complex. Earth gravity assists were not considered because one of

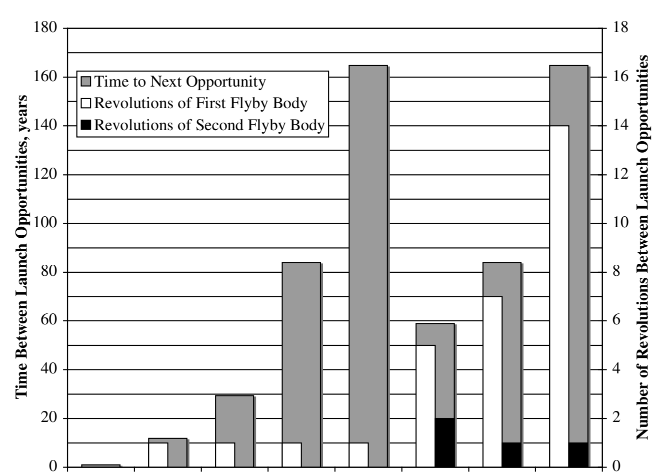

Fig. 3 Trajectory opportunity repetition.

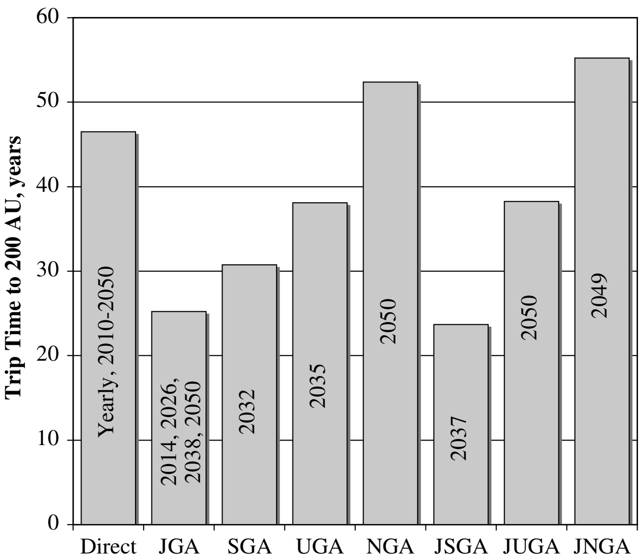

Fig. 4 Performance of the different trajectories, superimposed minimum trip time launch opportunity years.

the top-level requirements was to exclude any Earth gravity assists due to the radioisotope material onboard. Although National Aeronautics and Space Administration (NASA) spacecraft with RPSs on board have performed Earth gravity assists in the past and NASA will consider them in the future, this is a political topic that can change liens against a mission much more rapidly than designs can be changed. With no Earth gravity assist, multiple Venus flybys are required to significantly add final speed. This will increase further the dwell time in the inner solar system while increasing mission complexity and further constrain a window for a Jupiter flyby. Neither Mars nor Mercury flybys offer any obvious advantage while compounding the complexities already inherent in Venus flybys. Hence, gravity assists from Earth and/or Venus, while potentially improving the trajectory, were not studied further due to potential added complexity and increased risk for the mission overall.

The sidereal period of the outer solar system 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 Fig. 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 with 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.

B. Trajectory Design Trades and Selection

Trajectory design and optimization was completed using the direct trajectory optimization method (DTOM) code. As the name suggests, the DTOM is a direct method for obtaining optimal, low-thrust, interplanetary trajectories [41]. The DTOM numerically integrates the three-dimensional equations of motion using modified equinoctial orbital elements to accommodate circular orbits (eccentricity of 0) [42]. The parameterized continuous-time control, thrust and coast lengths, launch date scaling factor, and Earth-escape parameters define the generic design space. More specialized problems can be defined with planetary gravity assists, loiter periods at the target body (used for sample-return missions), optimization of power level and specific impulse (either single value or parameterized continuous-time profile), and specialized thruster system models. Previous REP trajectories have been verified with the more widely used VARITOP trajectory optimization code [43].

Three types of trajectories were studied; direct trajectories, single-gravity-assist trajectories, and double-gravity-assist trajectories. The direct trajectories are characterized by their high-energy launches

Table 1 Summary of trajectory parameters

Trajectory type	Direct	JGA	SGA	UGA	NGA	JSGA	JUGA	JNGA
Launch date	Jan. 17, 2010	Oct. 26, 2014	Aug. 26, 2032	Aug. 14, 2035	June 23, 2050	Oct. 3, 2037	Jan. 23, 2050	Dec. 18, 2049
First gravity-assist body		Jupiter	Saturn	Uranus	Neptune	Jupiter	Jupiter	Jupiter
Gravity-assist date		Dec. 7, 2015	Apr. 11, 2035	Aug. 11, 2042	May 2, 2063	Nov. 7, 2038	Dec. 19, 2056	June 6, 2055
Gravity-assist altitude		45388 km	3027 km	1286 km	1261 km	119,726 km	1,059,149 km	965,858 km
Gravity-assist radius		$1.63R_J$	$1.05R_S$	$1.05R_U$	$1.05R_N$	$2.67R_J$	$15.81R_J$	$14.51R_J$
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
Second gravity-assist body						Saturn	Uranus	Neptune
Gravity-assist date						Dec. 20, 2039	July 6, 2062	Jan. 21, 2071
Gravity-assist altitude						8393 km	1286 km	1261 km
Gravity-assist radius						$1.14R_S$	$1.05R_U$	$1.05R_N$
Gravity assist Δv						23.5 km/s	14.2 km/s	14.5 km/s
Burnout date	Mar. 31, 2036	Nov. 2, 2029	Sept. 11, 2051	June 22, 2060	Sept. 24, 2085	Nov. 23, 2051	Dec. 3, 2069	Mar. 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/yr	9.5 AU/yr	8.8 AU/yr	8.1 AU/yr	7.5 AU/yr	10.3 AU/yr	7.7 AU/yr	8.6 AU/yr
Date 2000 AU reached	July 8, 2056	Jan. 12, 2040	May 23, 2063	Sept. 13, 2073	Oct. 26, 2102	June 1, 2061	Apr. 8, 2088	Feb. 20, 2105
Trip time 200 AU	46.5 yr	25.2 yr	30.8 yr	38.1 yr	52.4 yr	23.7 yr	38.2 yr	55.2 yr
Speed at 200 AU	6.6 AU/yr	9.5 AU/yr	8.7 AU/yr	8.1 AU/yr	7.5 AU/yr	10.2 AU/yr	7.6 AU/yr	8.6 AU/yr
Azimuth at 200 AU	235.0 deg	254.8 deg	231.8 deg	231.8 deg	237.5 deg	263.5 deg	267.7 deg	232.2 deg
Elevation at 200 AU	0.0 deg	0.6 deg	7.4 deg	7.2 deg	21.3 deg	1.2 deg	3.3 deg	3.4 deg
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	$1.0kW_e$	$1.0kW_e$	$1.0kW_e$	$1.0kW_e$	$1.0kW_e$	$1.0kW_e$	$1.0kW_e$	$1.0kW_e$
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 yr	15.0 yr	19.1 yr	24.9 yr	35.3 yr	14.1 yr	19.9 yr	53.3 yr

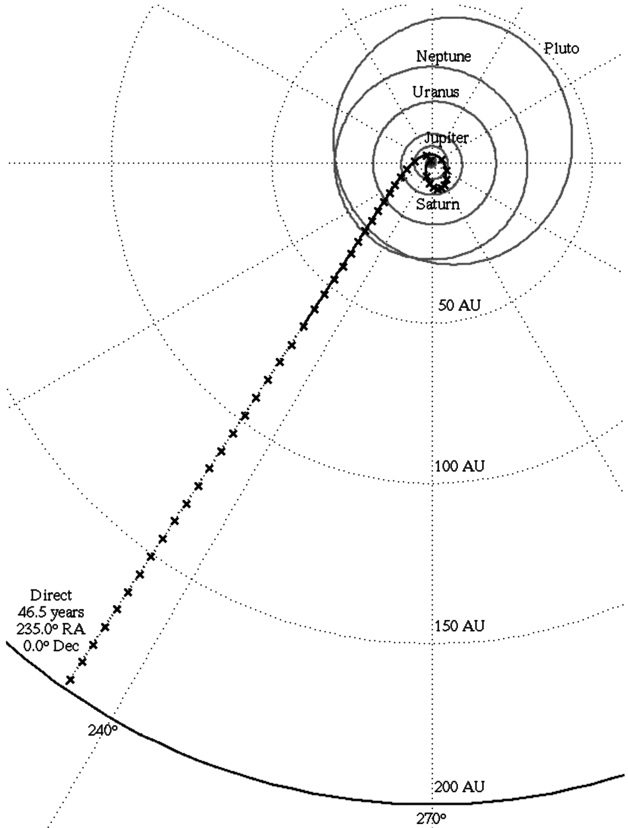

Fig. 5 Direct trajectory.

followed by long thrusting periods that propelled the spacecraft to 200 AU. The single-gravity-assist trajectories use a high-energy launch followed by a gravity assist at Jupiter, Saturn, Uranus or Neptune. The double-gravity-assist trajectories also launch to a high-energy Earth escape that is 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 Fig. 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 is the simplest and most flexible of the trajectories studied. It requires a C_3 of approximately 100 $\mathrm{km^2/s^2}$, makes one revolution around the Sun to achieve solar system escape velocity, and then proceeds to the heliospheric nose (see Fig. 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 is the most flexible because it does not require a planetary gravity assist, which allows it to be launched in any year. However, this trajectory performs poorly with a trip time to 200 AU of 46.5 yr with a lightweight spacecraft model, even though its velocity is twice that of the Voyager 2 spacecraft (see the column labeled "Direct" in Table 1).

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 are 25.2, 30.8, 38.1, and 52.4 yr, respectively. These trajectories all require a high-C_3 launch (between 120 and 150 $\mathrm{km^2/s^2}$), 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 Fig. 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 yr. The JGA trajectory can be launched in 2014, 2026, 2038, or 2050 and achieve performance 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 perform 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 ($106\ \mathrm{km^2/s^2} < C_3 < 154\ \mathrm{km^2/s^2}$), 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 Fig. 7). The trip times for the minimum-time double-gravity-assist trajectories are 23.7, 38.2, and 55.2 yr 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 with the other trajectories studied, and its minimum trip time launch window repetition every 11.9 yr (Jupiter's sidereal period). Each minimum trip time launch

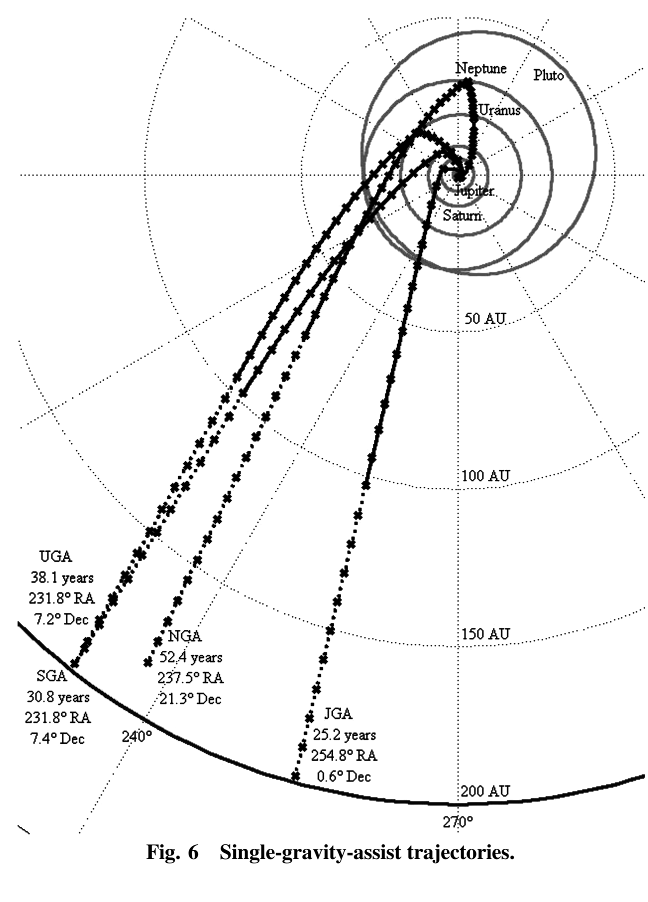

Fig. 6 Single-gravity-assist trajectories.

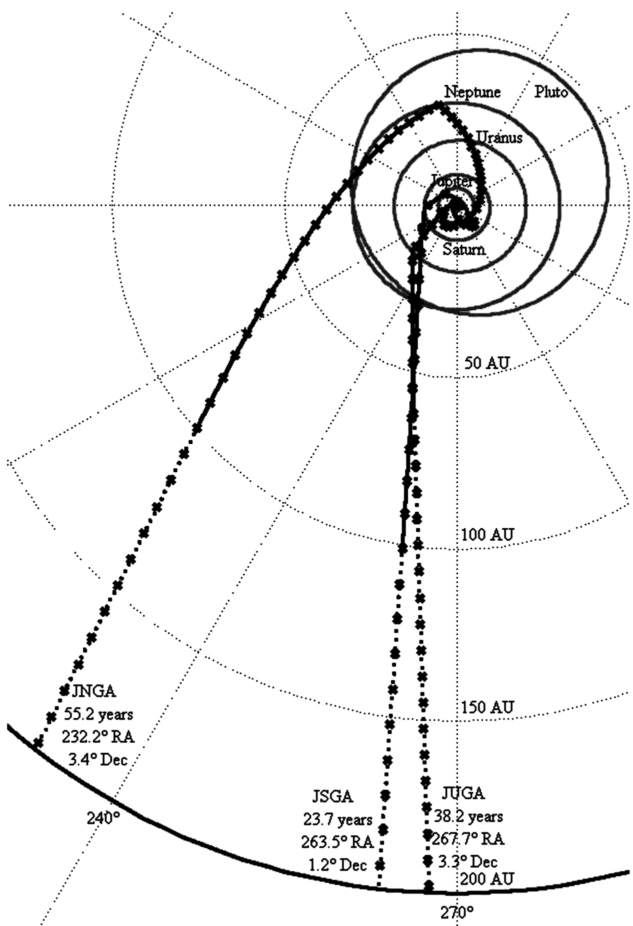

Fig. 7 Double-gravity-assist trajectories.

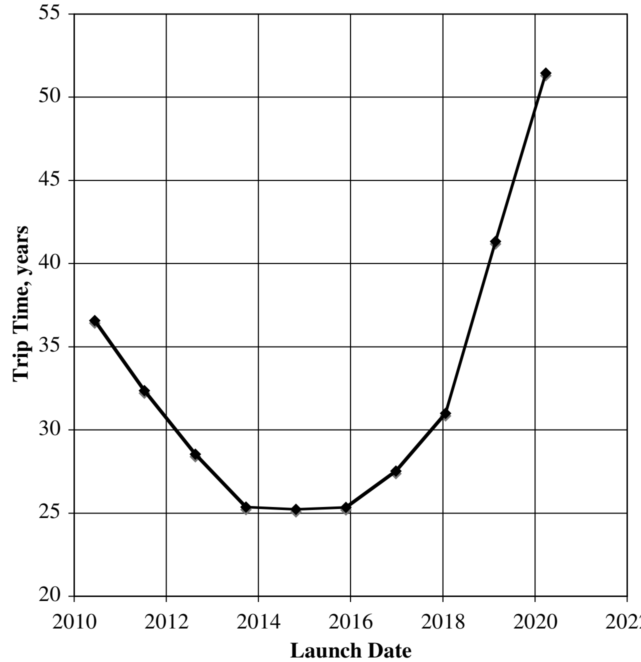

Fig. 8 Jupiter gravity-assist trajectory trip time vs launch date.

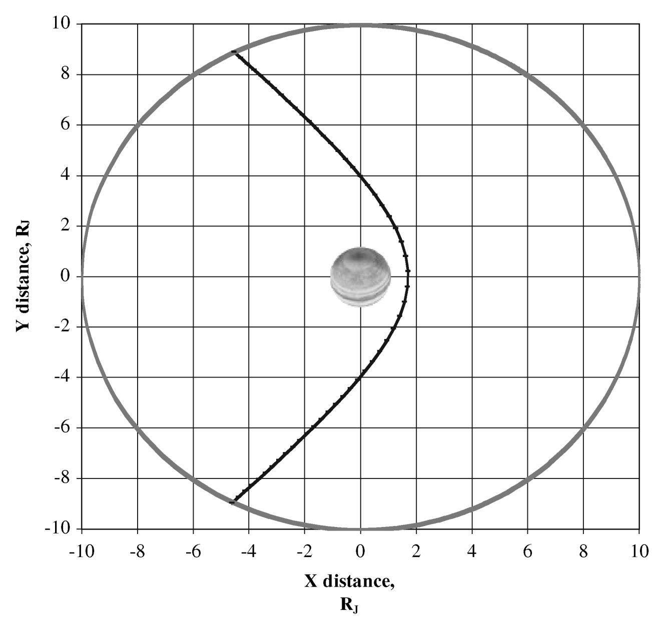

Fig. 9 Jupiter flyby trajectory inside $10R_J$ with 15-min interval tick marks.

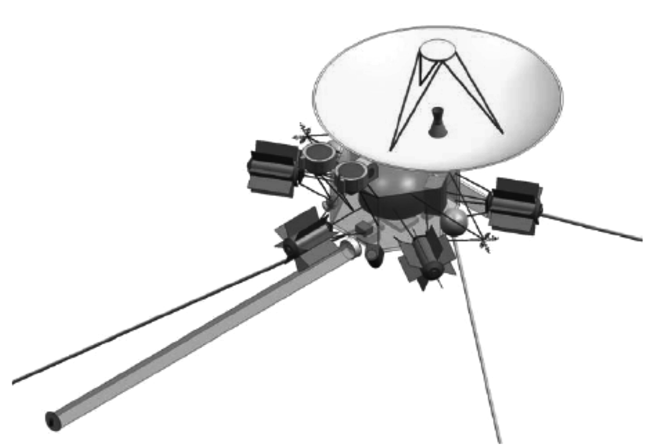

Fig. 10 Option 2 spacecraft design [1].

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 Fig. 8). 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 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 [44]. To attain the maximum effect of the gravity assist, the minimum flyby radius was constrained at $1.05R_J$, at the expense of time in the radiation belts. Figure 9 shows the JGA trajectory's Jupiter encounter within $10R_J$ ($0.5R_J$ outside of Europa) with tick marks indicating time intervals of 15 min. In this case, the spacecraft reaches approximately $1.7R_J$, receives a Δv of approximately 28 km/s, and has its trajectory turned by approximately 90 deg. For this Jupiter encounter, a total of approximately 13 h are spent inside $10R_J$, 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

Table 2 Option trades for spacecraft system design

Description	Option 1 mass Baseline	Option 2 mass Aggressive technology assumptions	Option 3 mass Reduced downlink rate	Option 4 mass Combination of options 2 and 3	Subsystem contingency	CBE+ contingency	Mode 1 power Safing	Mode 2 power Telecom after EP burnout	Mode 3 power Engine-off cruise	Mode 4 power Engine-on cruise 10 AU to EP burnout	Mode 5 power Launch	Mode 6 Power 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	36.0 W	36.0 W
Command and 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
Propulsion 1	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
Propulsion 2	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 and 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	761.6 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								

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. If the minimum distance from Jupiter's center is limited to $10R_J$, the trip time to 200 AU increases by approximately 3 yr.

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 cutoff 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 four 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 h 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-yr 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 Fig. 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. 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 reoptimized 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.

The power analysis was made on the basis of six defined operational modes for the spacecraft and the use of next-generation, high-temperature RTGs with a specific mass of $10.9W(e)/\mathrm{kg}$, mass per unit of 27.5 kg, and an operational power level of $228W(e)$ output at 15 yr, i.e., at the end of the thrusting phase. However, the engines are powered off during downlinks and the power level available on the probe must support the required data linkage as well. Hence, the power does not directly trade against the RPS units. For option 4 (see Table 2), that used a reduced data rate, dropping back to five RPS units is a possibility. Using more than six units makes placement on the spacecraft problematic as well as placement within the launch vehicle shroud, while also cutting into the mass margin for the launch vehicle. Six units meet the requirements for downlink while also allowing for reasonable propulsion capability and power and launch vehicle mass lift margins. A comparison of option 4 to the other options shows that at a systems level, propulsion power level does not trade directly against trip time. No other system-level designs were completed at different power levels.

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, 29.7, 30.7, and 29.9 yr (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 are all held constant for options 1

Table 3 Trajectory trades for spacecraft system designs

	Option 1	Option 2	Option 3	Option 4
Launch date	Oct. 22, 2014	Oct. 22, 2014	Oct. 22, 2014	Oct. 23, 2014
Gravity-assist body	Jupiter	Jupiter	Jupiter	Jupiter
Gravity-assist date	Feb. 13, 2016	Jan. 29, 2016	Feb. 10, 2016	Jan. 16, 2016
Gravity-assist altitude	79,131 km	71,970 km	77736 km	65,629 km
Gravity-assist radius	$2.11R_J$	$2.01R_J$	$2.09R_J$	$1.92R_J$
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	Feb. 4, 2033	Oct. 16, 2032
Burnout distance	105 AU	104 AU	104 AU	107 AU
Burnout speed	7.6 AU/yr	8.0 AU/yr	7.7 AU/yr	7.9 AU/yr
Date 2000 AU reached	Nov. 4, 2005	June 11, 2044	July 11, 2045	Sept. 10, 2044
Trip time 200 AU	31.1 yr	29.7 yr	30.7 yr	29.9 yr
Speed at 200 AU	7.6 AU/yr	8.0 AU/yr	7.7 AU/yr	7.8 AU/yr
Azimuth at 200 AU	265.1 deg	262.8 deg	264.7 deg	261.2 deg
Elevation at 200 AU	0.0 deg	0.0 deg	0.0 deg	0.0 deg
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 km^2/s^2	125.8 km^2/s^2	121.5 km^2/s^2	132.7 km^2/s^2
Delta IV H C_3	16.3 km^2/s^2	17.1 km^2/s^2	16.5 km^2/s^2	18.1 km^2/s^2
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 yr	17.7 yr	18.3 yr	18.0 yr

through 3, except for the final mass (dry mass) delivered to 200 AU, explaining the trip time differences between these options. Option 4 also decreases the power available to the EP system. This change decreases the level of acceleration the EP system can provide and also decreases the dry mass of the spacecraft, which results 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.

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 ($\sim$520 kg dry mass), could reach 200 AU within the right ascension and declination constraints in approximately 25 yr. A spacecraft design study was conducted to add considerations of technology readiness, margins, and physical layout of the spacecraft systems with a technology cutoff 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 yr longer for a total of approximately 30 yr 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 yr and have a potential lifetime of 15 yr 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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J. Martin
Associate Editor