Preliminary Design of Nuclear Electric Propulsion Missions to the Outer Planets

Chit Hong Yam,[*] T. Troy McConaghy,[†] K. Joseph Chen[*] and James M. Longuski[‡]
School of Aeronautics and Astronautics, Purdue University, West Lafayette, Indiana 47907-2023

Nuclear electric propulsion is expected to open new doors in deep space exploration. We study direct rendezvous missions to the outer planets which employ a constant-thrust, constant specific-impulse engine. We also consider how gravity assist can augment the capability of nuclear electric propulsion. We present numerical examples of gravity-assist missions to the outer planets, which use an engine similar to that of the Jupiter Icy Moons Orbiter. For example, in an Earth-Venus-Earth-Jupiter-Pluto mission, the spacecraft launches with a V_∞ of 2.2 km/s and rendezvous with Pluto in 10.5 years, with a propellant mass fraction of 50%. We demonstrate the benefit of using intermediate gravity-assist bodies (e.g. Venus, Earth and Mars) to decrease both mission duration and propellant cost.

Nomenclature

a_0	=	initial acceleration of the spacecraft, mm/s^2
g	=	standard acceleration due to gravity, m/s^2
I_{sp}	=	specific impulse, s
$\dot{m}$	=	propellant mass flow rate, mg/s
m_f	=	final spacecraft mass, kg
m_0	=	initial spacecraft mass, kg
P	=	power, kW
R_J	=	radius of Jupiter
T	=	engine thrust, N
V_∞	=	hyperbolic excess speed, km/s
ΔV	=	magnitude of a change in velocity, km/s
η	=	overall engine efficiency

I. Introduction

AN ambitious mission to Jupiter is being planned – the Jupiter Icy Moons Orbiter[1,2] (JIMO) mission. Although the details are uncertain, the JIMO spacecraft will be massive (as much as 20,000 kg after escape from Earth) and will use nuclear electric propulsion[3] (having a total engine power of about 100 kW) with a high specific impulse (about 6,000 s). The launch is expected in 2011.[1] Upon arrival at Jupiter, the spacecraft will spiral into orbit around Callisto, then Ganymede, and finally Europa – the Icy Moons of Jupiter.

If such a potent propulsive system is built, then a natural question to ask is what other missions might be flown. In this paper we will focus not on the Jupiter Icy Moons Orbiter Mission, but instead on innovative missions to the outer planets that are made possible by this new propulsion technology.

Our principal goal is to consider how a massive nuclear electric propulsion (NEP) engine, used in conjunction with gravity assists from the inner planets, can deliver a spacecraft to the outer planets (Jupiter, Saturn, Uranus, Neptune and Pluto). The technique of reaching the outer planets via gravity assists (but without electric propulsion) has been discussed for many decades beginning in the 1960's with the pioneering work of Minovitch,[4,5] Flandro,[6] Deerwester,[7] and Niehoff.[8] In the 1970's, additional contributions and applications of the gravity-assist technique were developed by Hollenbeck,[9] Stancati et al.[10] and Wallace.[11] In the 1980's, Wallace et al.[12] and Diehl and Myers[13] presented detailed trajectory computations that used gravity assist to reach the outer solar system. An

[*] Graduate Student. Student Member AIAA.
[†] Doctoral Candidate. Member AAS, Student Member AIAA.
[‡] Professor. Member AAS, Associate Fellow AIAA.

analysis of the efficacy of delta-V gravity-assist trajectories and a collection of practical results are discussed in the doctoral thesis of Sims.[14]

As early as the 1970's, the use of electric propulsion in conjunction with a gravity-assist from the Earth was investigated by Meissinger,[15] and Atkins et al.[16] In 1979, Sauer[17] described how solar electric propulsion (SEP) combined with an Earth gravity assist could deliver spacecraft to the outer planets. More recent work was published by Kawaguchi,[18] Sauer,[19] Williams and Coverstone-Carroll,[20] and Maddock and Sims.[21] In solar electric propulsion missions, the low thrust trajectory arcs are restricted to a range of only a few astronomical units (AU) from the Sun. Beyond a few AU, the available power drops rapidly and the engine becomes inoperable.

The advantage of NEP is that power is always available although at a cost of additional hardware mass. NEP missions are discussed by Jones and Sauer,[22] Kluever,[23] and Fedotov et al.[24]

In 1999, Sims and Flanagan[25] developed a parameter optimization technique in which a low-thrust gravity-assist (LTGA) trajectory could be modeled as a series of impulsive ΔVs patched together by conic arcs. Their method can be used to find optimal SEP and NEP gravity-assist trajectories with maximum final mass. Based on their technique, an LTGA optimization program was written called the Gravity-Assist, Low-Thrust, Local Optimization Program[26] (GALLOP). Also in 1999, Petropoulos et al.[27] developed a shape-based method to represent low-thrust gravity-assist trajectories in a broad search, propagation program (STOUR-LTGA). Petropoulos and Longuski[28,29] demonstrated how LTGA trajectories could be automatically designed to provide preliminary guesses to GALLOP. This method is described in Petropoulos' doctoral thesis.[30] An extension of the LTGA trajectory design work was presented by McConaghy et al.[26] in 2003. In that same year low-thrust trajectories to the outer solar system were presented by Cupples et al.,[31] Woo et al.,[32] and Vasile et al.[33,34]

In the current work, we employ the concepts of the aforementioned researchers to design rendezvous missions. We assume the spacecraft uses an engine similar in capability to the JIMO engine. We first consider the case that directly launches from the Earth to the outer planets without gravity-assist. The second step is to study LTGA trajectories with different encounter sequences.

II. Assumptions

We consider a spacecraft launched with a fictitious launch vehicle, which we will refer to as "the heavy lifter." The upper stage of the heavy lifter injects the spacecraft into an interplanetary escape orbit with chemical propellant of an I_{sp} of 404 s. We assume the upper stage of the vehicle is able to inject a spacecraft with a mass of 20,000 kg to parabolic escape (i.e. $V_\infty = 0$) or a lesser mass to hyperbolic speed ($V_\infty > 0$). Figure 1 shows the injection capability of this hypothetical launch vehicle. As we can see in Fig. 1, the heavy lifter capability is 115% greater than that of the Boeing Delta 4050H-19[*] launch vehicle.

We first study the case where a spacecraft launches with a zero V_∞ to rendezvous with the outer planets. By rendezvous with the outer planets, we mean the spacecraft's position and velocity match that of the target planet (i.e. the arrival V_∞ is 0). As a potential follow-on mission to the JIMO, we consider the launch years 2014–2025. For the purpose of demonstrating mission feasibility, we select parameters based on the currently developing Nuclear Electric Xenon Ion System[3] (NEXIS). See Table 1. We assume (somewhat arbitrarily) that an acceptable final mass could be as low as 9,000 kg.

Table 1: Nuclear Electric Spacecraft Parameters

Parameter	Values
Power Available to the Thrusters , P	95 kW[a]
Specific Impulse, I_{sp}	6,000 s
Overall Efficiency, η	70 %
Thrust, T	2.26 N
Mass Flow Rate, $\dot{m}$	38.4 mg/s
Injected Mass at Zero Launch V_∞, m_0	20,000 kg

[a] The spacecraft is propelled by five NEXIS thrusters.

Although these parameters are rather specific, we note that they can be scaled to smaller (and even larger) values. Two scaling parameters govern the problem: the initial acceleration of the spacecraft and the specific impulse of the engine. The initial acceleration a_0 and the specific impulse I_{sp} can be written as:

[*] *The Delta IV Payload Planners Guide*, The Boeing Company, Huntington Beach, CA, 2000.

$$a_0 = T / m_0 = (2\eta P)/(m_0 g I_{sp}) \tag{1}$$

$$I_{sp} = T/(\dot{m}g) \tag{2}$$

For missions with the same values of a_0 and I_{sp}, the resulting trajectories will be the same as the ones we present in this paper. (In our problem I_{sp} = 6,000 s and a_0 = 0.11 mm/s^2.)

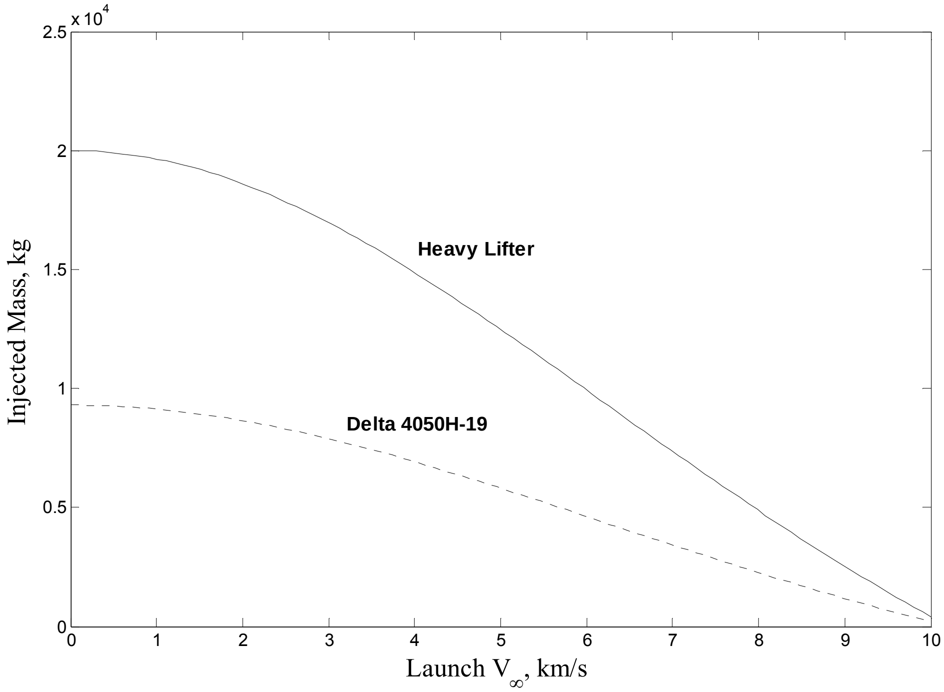

Figure 1: Launch capability of the heavy lifter, a fictitious launch vehicle, compared to the Delta 4050H-19.

III. Direct Missions

Before we consider the problem of LTGA trajectories to the outer planets, we examine the capability of the JIMO propulsion system to deliver spacecraft to these distant destinations without the aid of gravity assist. We first study the case with zero-launch-V_∞ to rendezvous with the target planets.

Our problem is to maximize the final mass of the spacecraft, after transfer between (nearly) circular coplanar orbits using continuous thrust. This type of problem has been discussed since the 1950's;[35-39] analytical or approximate solutions are found in special cases.[40-46] We take a simple approach to assess the potential of this NEP spacecraft. Our initial guess is generated by assuming the spacecraft thrusts continuously along its velocity vector. Figure 2 shows the trajectory plot for a spacecraft with parameters given in Table 1. We estimate the corresponding launch date at Earth by the time and the transfer angle when the trajectory crosses the target planetary orbit. For example, a launch date of July 30, 2018 and the tangential steering law provides an initial guess for a flyby mission to Neptune. An optimal flyby trajectory to Neptune is then found in GALLOP with the estimated departure date. To find an optimal rendezvous trajectory, we decrease the arrival V_∞ (which is large in the initial guess) by extending the flight time of the flyby mission. With a smaller arrival V_∞, the flyby trajectory provides a close initial guess in finding the rendezvous trajectory in GALLOP. Table 2 shows some of the optimal solutions found during the process of finding a rendezvous solution. We found that this process provides a quick way to find an optimal rendezvous trajectory in GALLOP.

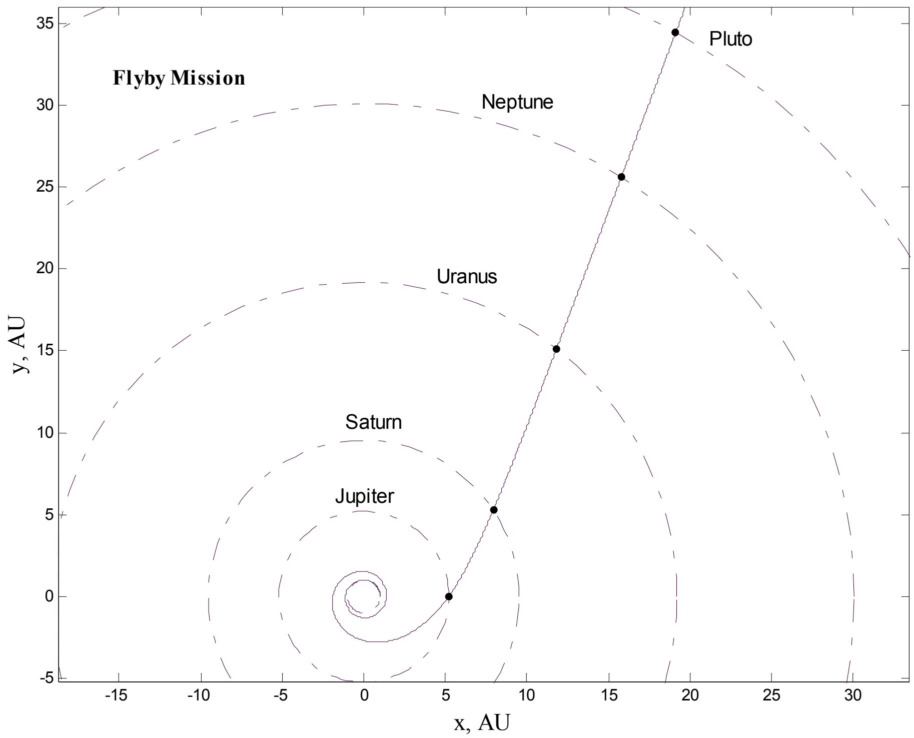

Figure 2: Trajectory plot of a spacecraft thrusting continuously along its tangential direction.

Table 2: Optimal Earth-Neptune Trajectories

Launch Date	Arrival Date	Time of Flight, years	Arrival V_∞, km/s	Final Mass, kg
July 30, 2018[a]	Feb 2, 2028[a]	9.51[a]	34.1	9,628
July 30, 2018[a]	Jan 28, 2030[b]	11.5[a]	17.6	12,640
July 30, 2018[a]	Nov 24, 2030[b]	12.3[a]	0[c]	8,790
Aug 4, 2018[b]	Nov 29, 2030[b]	12.3[a]	0[c]	8,794
Aug 7, 2019[b]	Nov 11, 2030[b]	11.3[a]	0[c]	6,347

[a] Frozen during optimization.
[b] Freed during optimization.
[c] Rendezvous case has a zero arrival V_∞ by construction.

Of particular interest in our study, we examine the case with a minimum-time transfer between Earth and an outer planet. The minimum-time transfer case is the optimal rendezvous trajectory with the spacecraft thrusting continuously at its maximum level (i.e. without coasting). Launch and arrival dates are freed as variables while constraining the time-of-flight (TOF). Table 3 provides the key characteristics of the minimum TOF trajectory to the outer planets. We see how the minimum TOF varies with the arrival distance from the Sun in Fig. 3. As the arrival distance increases, the minimum TOF approaches an asymptote labeled as "the zero mass limit." The zero mass limit represents the time required for the spacecraft to (literally) exhaust all its mass and is given by $m_0/\dot{m}$ = 16.5 years. In the cases shown in Fig. 3, since the spacecraft is assumed to thrust continuously at its maximum level, we can estimate the required propellant by the product of the mass flow rate and the TOF.

Table 3: Optimal Direct Rendezvous Trajectories

Target Planet	Launch Date	Launch V_∞, km/s	Initial Mass, kg	Time of Flight, years	Final Mass, kg
Launch V_∞ constrained to be zero					
Jupiter	Apr 13, 2024	0	20,000	5.53	13,301
Saturn	Oct 9, 2022	0	20,000	7.20	11,285
Uranus	Oct 20, 2020	0	20,000	9.68	8,268[a]
Neptune	Aug 7, 2019	0	20,000	11.3	6,347[a]
Pluto	May 28, 2015	0	20,000	11.8	5,661[a]
Launch V_∞ unconstrained					
Jupiter	Apr 24, 2024	1.23	19,463	5.53	13,709
Saturn	Jan 24, 2023	1.30	19,398	7.20	12,653
Uranus	June 5, 2021	2.32	18,144	9.68	10,402
Neptune	Apr 21, 2020	2.63	17,640	11.3	8,543[a]
Pluto	Aug 20, 2015	2.75	17,425	11.8	6,890[a]

[a] Violates final mass constraint ($m_f \geq 9{,}000$ kg).

Next, we consider the case where the launch V_∞ is a free variable. Using the minimum-flight-time trajectory as an initial guess, an optimal solution is found with non-zero launch V_∞. Encounter dates are freed while the TOF is constrained to the same value as the zero launch V_∞ case. From Table 3, we observe that for each outer planet mission, when the launch V_∞ is greater than zero, the final mass is greater than that of the zero launch-V_∞ case. Thus the launch V_∞ provides an initial boost to the trajectories, which can replace the initial thrusting phase of the low-thrust engine. (It is better to use an impulsive ΔV with a lower I_{sp} in these cases.) Also, the further the target planet is, the higher the initial boost required. In addition, we notice from Fig. 4 that the non-zero launch V_∞ trajectories are no longer thrusting continuously as in the case of zero-launch-V_∞ (for a fixed TOF). Unfortunately, in the cases of Neptune and Pluto the final mass constraint ($m_f \geq 9{,}000$ kg) cannot be met.

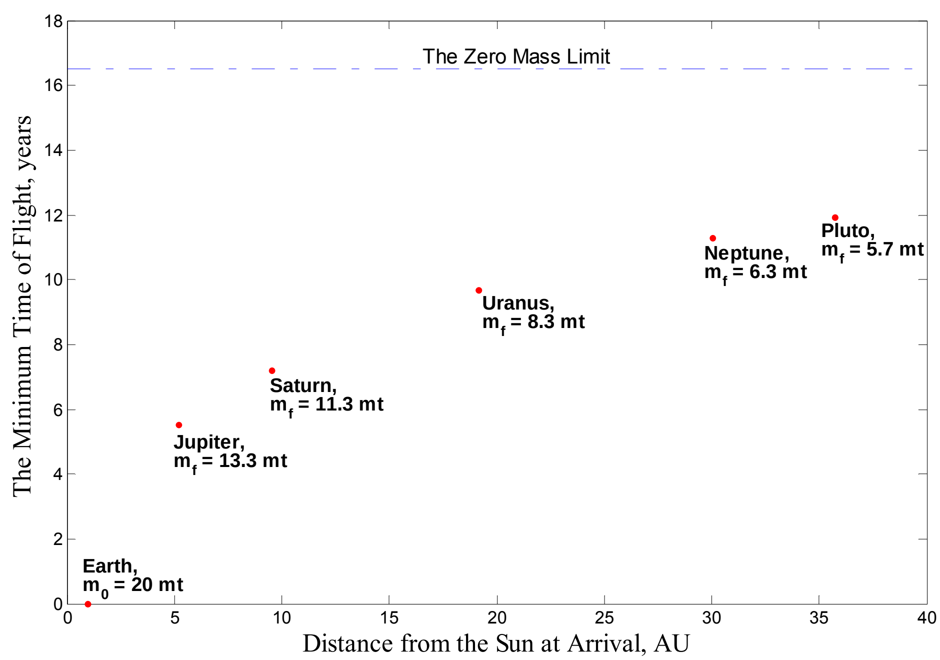

Figure 3: The Minimum time of flight as a function of the arrival distance for zero-launch V_∞; final mass in metric tons, mt.

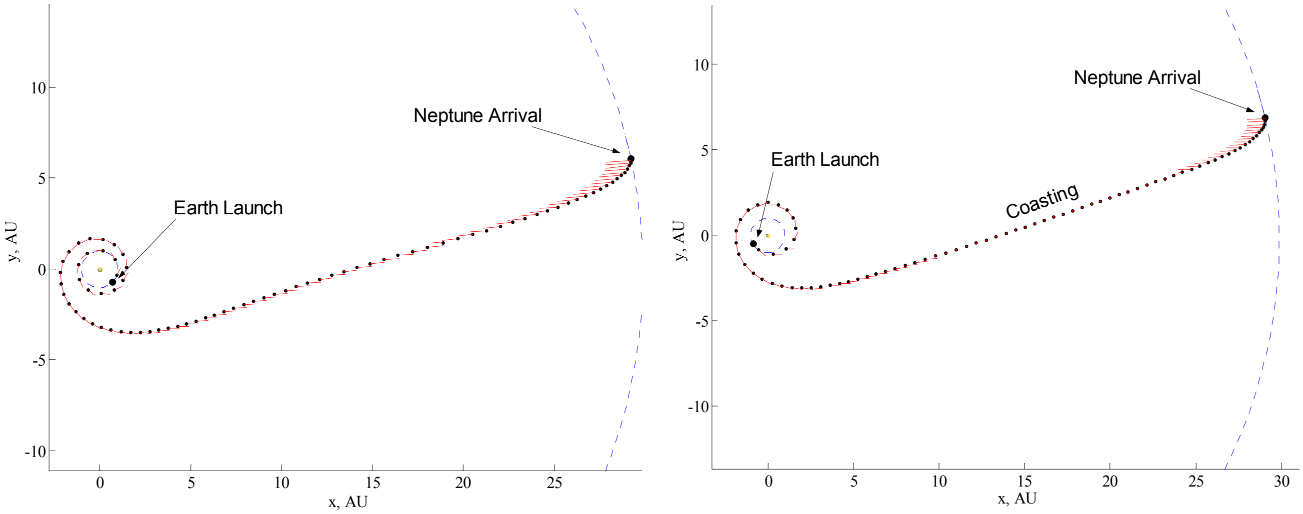

Figure 4: Optimal Earth-Neptune trajectories; left: launch V_∞ constrained to be zero, right: launch V_∞ freed for the same TOF.

IV. Methodology for LTGA Trajectory Design

A. Broad Search

As a first step, we perform broad searches of low-thrust trajectories using a shaped-based, automated approach as described in Refs. 27-30. The shaped-based method provides an analytic representation of the low-thrust trajectory arc, which allows rapid searches over a wide range of the design space. We assume a two-body model, with coast and thrust arcs patched together betweens flybys of celestial bodies. We employ conic sections for coasting arcs and exponential sinusoids[27] for thrusting arcs. The polar equation of an exponential sinusoid is given by $r = k_0 \exp[k_1 \sin(k_2\theta + \varphi)]$; where k_0, k_1, k_2, and φ are constants. The required thrust acceleration at each point of the trajectory can be determined from the trajectory shape.

The shape-based approach is implemented as software named STOUR-LTGA. To perform a broad search in STOUR-LTGA, the user specifies the encounter sequence together with the launch date and launch V_∞ ranges and step sizes. Numerous trajectories can be analyzed and selected by a MATLAB toolbox called STOUR Interactive Toolbox (SIT). In general, we select trajectories with low propellant mass fraction, low TOF, low launch V_∞ and acceptable thrust acceleration.

B. Optimization

Candidate trajectories selected from the broad search are used as the initial guesses for a low-thrust gravity assist trajectory optimization program called GALLOP.[26] The trajectory model in GALLOP divides each planet-planet leg of the trajectory into segments of equal duration. (For a direct mission without gravity assist, there is only one leg.) The thrusting on each segment is modeled by an impulse at the midpoint of the segment, with conic arcs between the impulses. The leg is propagated forward from the launch body and backward. In order to have a feasible trajectory, the forward- and backward-propagated half-legs must meet at a match-point time in the middle of the leg.

The optimization variables in GALLOP include the following: 1) the impulsive ΔV on each segment, 2) the Julian dates at the launch, flyby, and destination bodies, 3) the launch V_∞, 4) the incoming inertial velocity vectors at all of the postlaunch bodies, 5) the spacecraft mass at each body, 6) the flyby periapsis altitude at the gravity-assist bodies, and 7) the B-plane angle at the gravity-assist bodies.

The optimization program can alter these variables to find a feasible and optimal solution of the given problem. A feasible solution means the variables satisfy the constraints. These constraints include upper bounds on the impulsive ΔV on each of the segments, the launch-V_∞ magnitude and the encounter dates at the bodies. Within the feasible set of solutions, the optimizer can find a solution which maximizes the final mass of the spacecraft.

V. Numerical Results

A. LTGA Trajectory to Jupiter via Mars Gravity Assist

To illustrate the shaped-based approach, we study an Earth-Mars-Jupiter mission with a zero-launch-V_∞ constraint. We begin with a broad search in STOUR, with launch dates from Jan 2011 to Dec 2023 and a 20-day step size. Since we are interested in finding a candidate trajectory with thrust acceleration close to the parameters given in Table 1, we focus on trajectories with a small acceleration. For missions with the zero-launch-V_∞ constraint, we also select trajectories based on their launch V_∞. Figure 5 shows 5,424 trajectories with maximum acceleration less than 0.15 mm/s^2. (For the parameters in Table 1, the initial acceleration is 0.11 mm/s^2.) Trajectories with the smallest launch V_∞ (0.2 km/s) are also indicated by circles in Fig. 5. (We note that STOUR cannot constrain the launch V_∞ to be exactly zero.) We observe promising groups of trajectories that launch around 2019 to 2022.

Next, we perform a refined search of E-M-J trajectories from Jan 2019 to Dec 2022, with a step size of 10 days. Figure 6 shows the resulting trajectories with maximum acceleration less than 0.15 mm/s^2. A candidate trajectory with low propellant mass fraction and reasonable TOF (indicated by the arrow in Fig. 6) is selected as an initial guess for optimization in GALLOP. From Fig. 7, we can see that the trajectory of the initial guess and the optimal solution are similar in their shape. Table 4 gives a comparison of the trajectory characteristics of the STOUR initial guess and the optimized solution in GALLOP. We notice that the optimizer moved the Mars encounter date almost four months later.

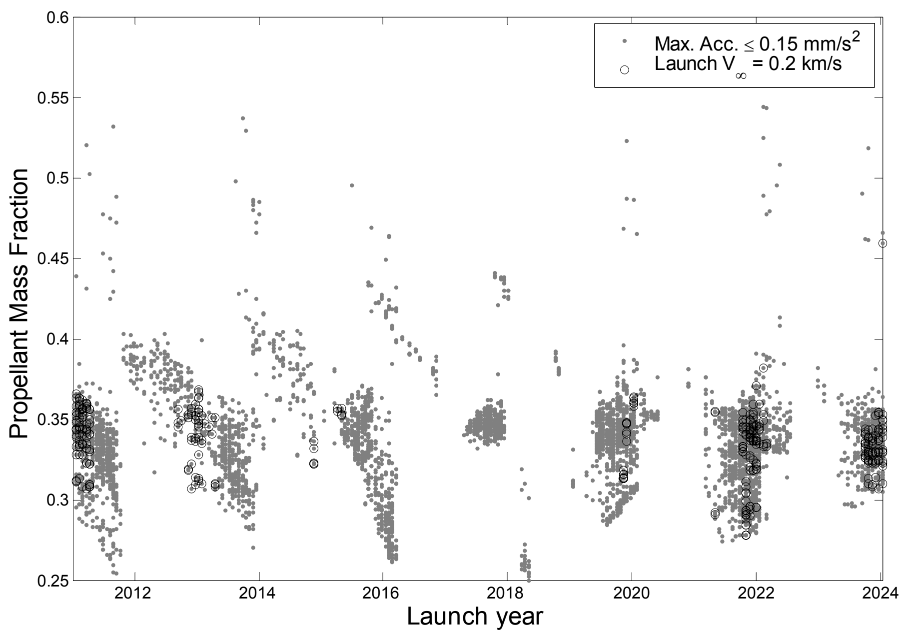

Figure 5: STOUR-LTGA broad search for Earth-Mars-Jupiter trajectories; launch date ranges from the year 2011 to 2023.

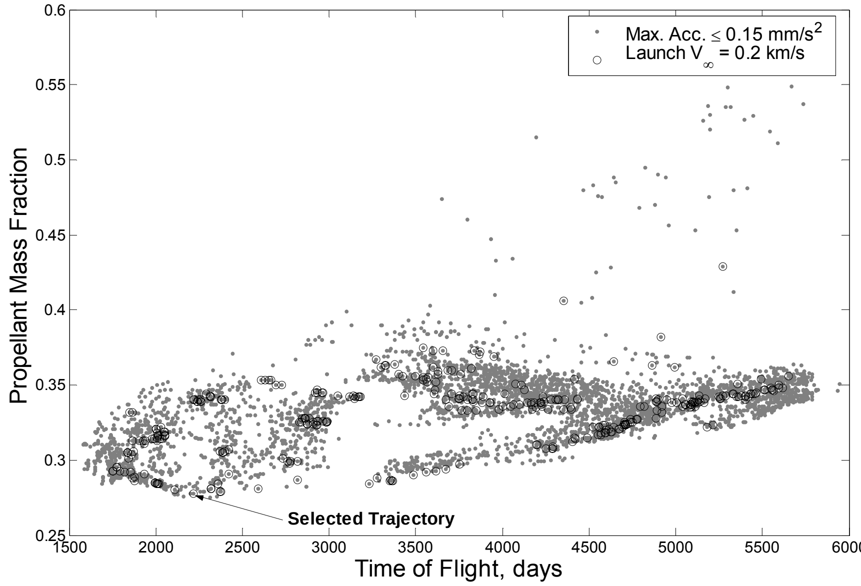

Figure 6: STOUR-LTGA refined search for Earth-Mars-Jupiter trajectories; launch date ranges from the year 2019 to 2022.

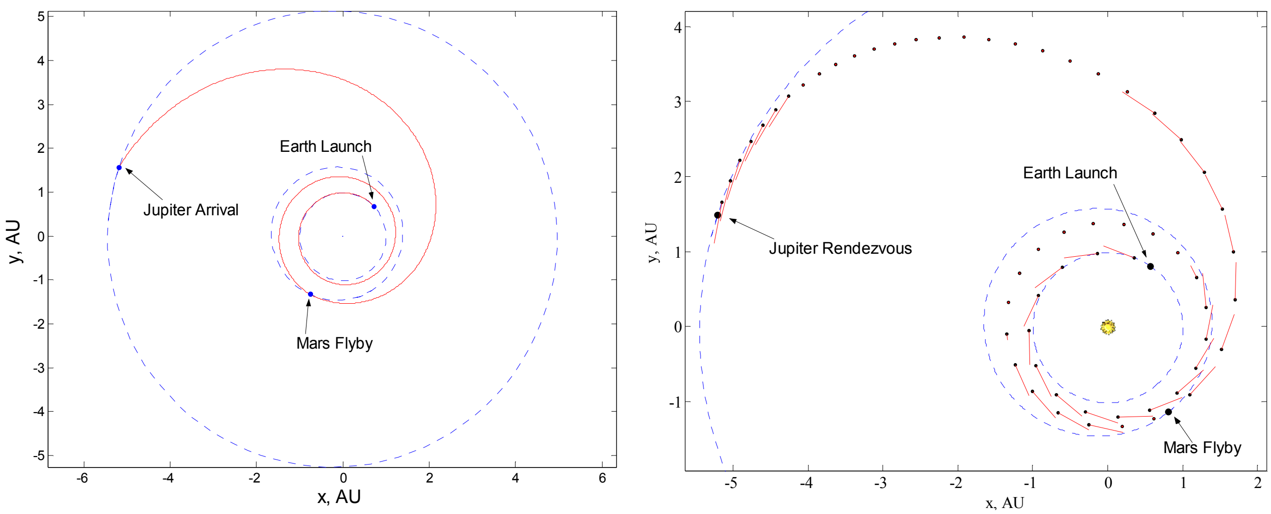

Figure 7: Earth-Mars-Jupiter trajectory; left: STOUR initial guess, right: optimal solution in GALLOP.

Table 4: E-M-J rendezvous trajectory

Characteristics	**STOUR-LTGA**	**GALLOP**
E-1 Launch date	Nov 6, 2021	Nov 16, 2021
E-1 Launch V_∞, km/s	0.2	0
M-2 Flyby date	Nov 28, 2023	Mar 20, 2024
M-2 Flyby V_∞, km/s	2.95	3.61
M-2 Flyby altitude, km	410	500[b]
M-2 Flyby B-plane angle[a]	-21.6^o	0^o
J-3 Arrival date	Nov 15, 2027	Nov 26, 2027
J-3 Arrival V_∞, km/s	2.89	0
Total TOF, years	6.02	6.02
Initial mass, kg	20,000	20,000
Final mass, kg	14,420	16,026
Propellant mass fraction	27.9%	19.9%

[a] Fundamental plane taken as ecliptic of J2000.
[b] On lower bound.

To illustrate the benefit of the Mars gravity assist, we vary the TOF (while encounter dates are freed) of the Jupiter direct mission and the Earth-Mars-Jupiter mission. Figure 8 shows how the propellant consumption of Jupiter rendezvous missions changes with the flight time. We observe that with a Mars gravity assist, the minimum TOF to Jupiter can be shortened to 5 years. In addition, the Earth-Mars-Jupiter trajectory requires less propellant than the direct mission for a specific TOF.

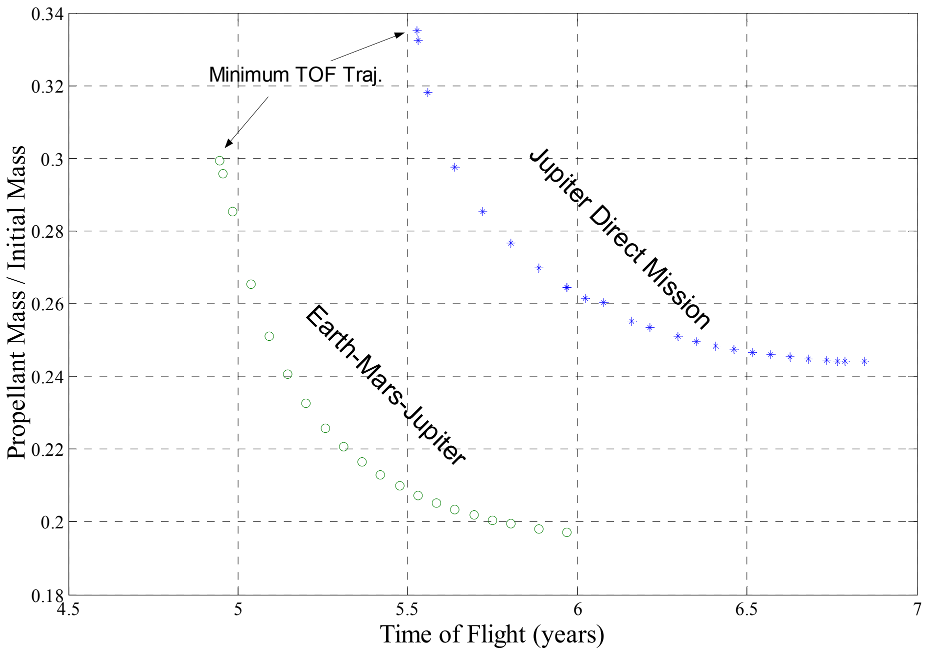

Figure 8: Trade Study of Jupiter Rendezvous Mission, with launch V_∞ constrained to be zero.

B. Other LTGA Trajectories to Jupiter

We next study cases with an Earth gravity assist and a Venus-Earth gravity assist. Table 5 summarizes the trajectory characteristics of an Earth-Earth-Jupiter rendezvous mission. From Fig. 9, we observe that between launch and the Earth flyby, the spacecraft thrusts (on average) towards the inertial positive Y direction. The Earth V_∞ increases from 0.71 km/s to 7.5 km/s in 1.2 years during this Earth-Earth transfer. The total mission takes 4.5 years with a final mass of 16.2 mt (metric tons). We notice that the Earth-to-Earth transfer in our trajectory is very similar to that suggest by Kawaguchi in Ref. 18.

An Earth-Venus-Earth-Jupiter rendezvous trajectory is shown in Fig. 10. The trajectory has three phases of thrusting. Launching from the Earth with a V_∞ of 2.2 km/s, the energy of the orbit is decreased via the spacecraft thrusting against its velocity. After the Venus flyby, the spacecraft coasts back to the Earth with a flyby V_∞ of 10 km/s. The second thrusting phase occurs after the Earth flyby to further increases the energy of the orbit. Finally, the spacecraft rendezvous with Jupiter during its third thrusting phase. To compare the mission performance, the TOF of the E-V-E-J trajectory is kept the same as the E-E-J (4.5 years). We notice that with the same TOF, the E-E-J mission has a final mass of about 600 kg greater than that of the E-V-E-J mission.

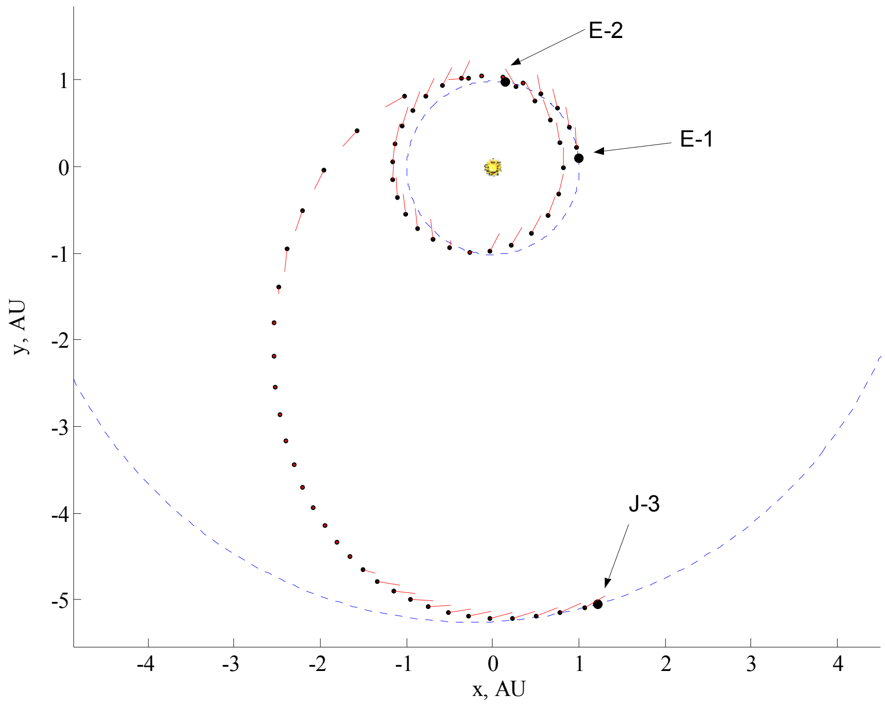

Figure 9: Trajectory plot of an Earth-Earth-Jupiter rendezvous mission.

Table 5: E-E-J rendezvous trajectory

Characteristics	Value
E-1 Launch date	Sep 29, 2015
E-1 Launch V_∞, km/s	0.71
E-2 Flyby date	Dec 13, 2016
E-2 Flyby V_∞, km/s	7.49
E-2 Flyby altitude, km	500[b]
E-2 Flyby B-plane angle[a]	0^o
J-3 Arrival date	Apr 5, 2020
Total TOF, years	4.52
Initial mass, kg	19,820
Final mass, kg	16,181
Propellant mass fraction	18%

[a] Fundamental plane taken as ecliptic of J2000.
[b] On lower bound.

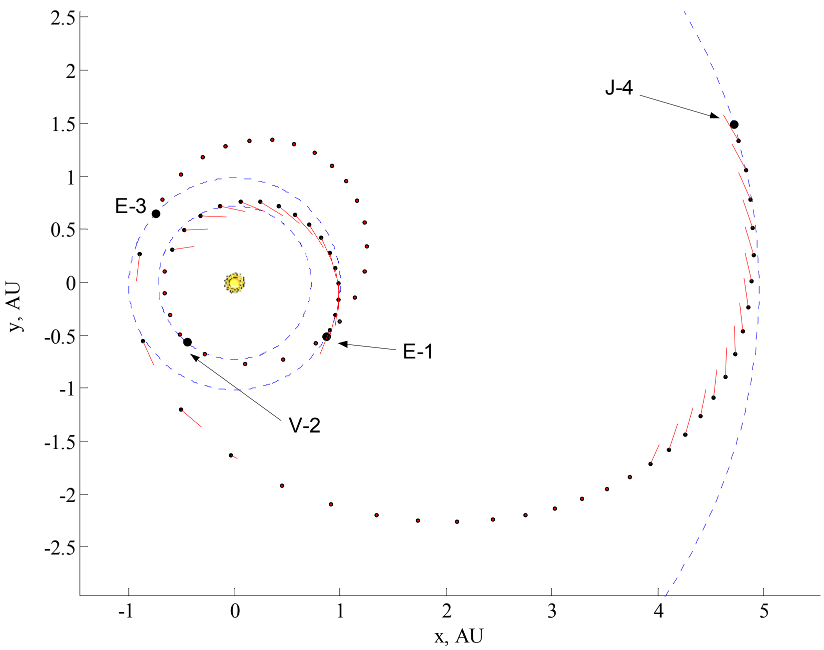

Figure 10: Trajectory plot of an Earth-Venus-Earth-Jupiter rendezvous mission.

Table 6: E-V-E-J rendezvous trajectory

Characteristics	**Value**
E-1 Launch date	Aug 22, 2018
E-1 Launch V_∞, km/s	2.19
V-2 Flyby date	Feb 26, 2019
V-2 Flyby V_∞, km/s	5.75
V-2 Flyby altitude, km	500[b]
V-2 Flyby B-plane angle[a]	14°
E-3 Flyby date	Feb 8, 2020
E-3 Flyby V_∞, km/s	10.3
E-3 Flyby altitude, km	500[b]
E-3 Flyby B-plane angle[a]	177°
J-4 Arrival date	Feb 27, 2023
Total TOF, years	4.52
Initial mass, kg	18,333
Final mass, kg	15,606
Propellant mass fraction	15%

[a] Fundamental plane taken as ecliptic of J2000.
[b] On lower bound.

C. LTGA Trajectories to Pluto

A mission to Pluto presents many challenges. Due to the great distance (Pluto is at 36 AU in 2025), it is extremely difficult to have a reasonable TOF for a practical propellant cost. While a direct Earth-Pluto mission can achieve a moderate TOF of just under 12 years, its propellant mass fraction of 72% is far from realistic. It is thus clear that an acceptable mission to Pluto must employ gravity assists (for our system).

One of the most promising trajectories we have designed is one that uses gravity assist from Venus, Earth, and Jupiter. Finding good initial guesses for the trajectories to Pluto is difficult. The exponential sinusoid model provides reasonable shapes for low- to medium-energy trajectories.[30] For missions to the far outer planets (from Uranus to Pluto), however, the orbital energies of the transfer orbits can be quite high (nearly hyperbolic). In order to find a reasonable initial guess in STOUR, we introduce a coasting arc (conic) for the Jupiter-Pluto leg. Figure 11 shows the trajectory plot of the initial guess and the optimal solution for the E-V-E-J-P mission. The solid line in the STOUR plot represents a pure thrusting arc while the dashed line indicates a coasting arc. Table 7 provides the trajectory itinerary. In our experience, when we introduce a coasting arc, STOUR usually overestimates the TOF for trajectories to the far outer planets. We attempt to decrease the flight time from 40 years to 10 years by gradually forcing the Pluto arrival date earlier (while keeping other encounter dates frozen). By doing so, GALLOP turns the coasting arc into a thrusting arc which rendezvous with Pluto. A final step of freeing all the encounter dates (with a TOF upper bound of 10.5 years) provides the optimized trajectory shown in Fig. 11. In the figure, we notice the dramatic bending that Jupiter provides.

Figure 12 shows the E-V-E-J-P trajectory in the inner solar-system in greater detail. Starting from the Earth (thrusting in an inertially fixed direction), the spacecraft encounters Venus after about half a rev around the Sun. After a flyby at Venus and encountering the Earth again, a close swingby of Jupiter boosts the spacecraft to Pluto. The total TOF of this trajectory is 10.5 years, and the PMF is 50% (significantly better than the direct mission).

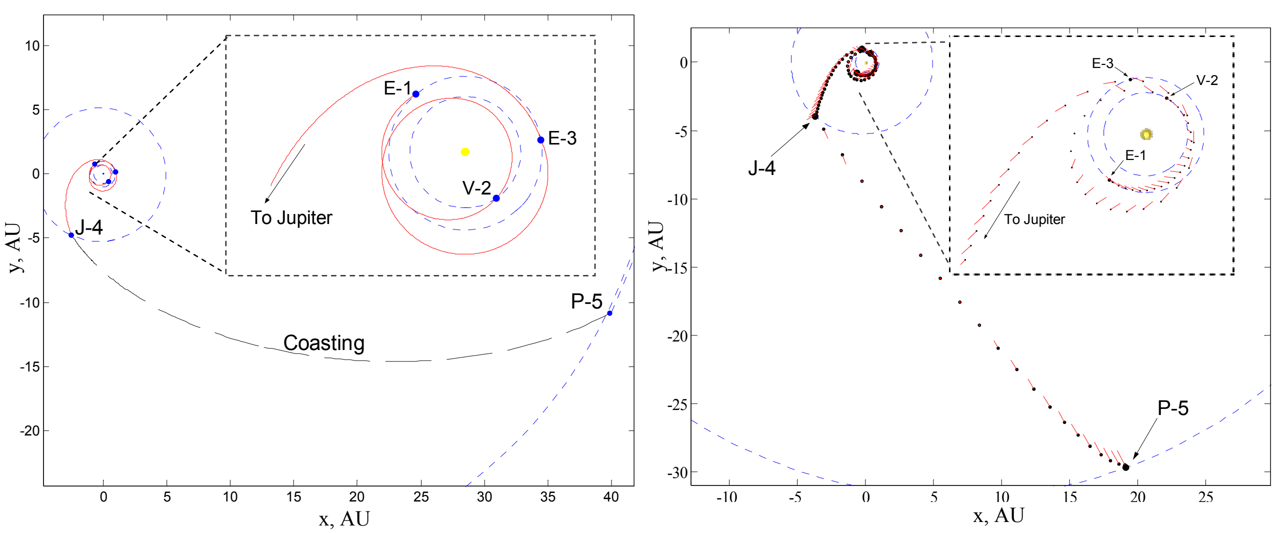

Figure 11: E-V-E-J-P trajectory; left: STOUR initial guess, right: optimal solution in GALLOP.

Table 7: E-V-E-J-P rendezvous trajectory

Characteristics	STOUR-LTGA	GALLOP
E-1 Launch date	Jan 31, 2015	May 13, 2015
E-1 Launch V_∞, km/s	6.0	2.19
V-2 Flyby date	Aug 4, 2015	Oct 16, 2015
V-2 Flyby V_∞, km/s	9.32	5.41
V-2 Flyby altitude, km	92,214	6,422
V-2 Flyby B-plane angle[a]	-95°	109°
E-3 Flyby date	Oct 2, 2016	Jan 5, 2017
E-3 Flyby V_∞, km/s	7.73	11.2
E-3 Flyby altitude, km	815	500[b]
E-3 Flyby B-plane angle[a]	-148°	-1°
J-4 Flyby date	Nov 8, 2018	Apr 29, 2018
J-4 Flyby V_∞, km/s	9.64	16.7
J-4 Flyby altitude, R_J [c]	41.2	2.41
J-4 Flyby B-plane angle[a]	-63°	-4°
P-5 Arrival date	July 11, 2055	Nov 26, 2025
P-5 Arrival V_∞, km/s	2.70	0
Total TOF, years	40.4	10.5
Initial mass, kg	9,928	18,340
Final mass, kg	5,391	9,196
Propellant mass fraction	46%	50%

[a] Fundamental plane taken as ecliptic of J2000.
[b] On lower bound.
[c] Assuming a Jupiter radius of 71,492 km.

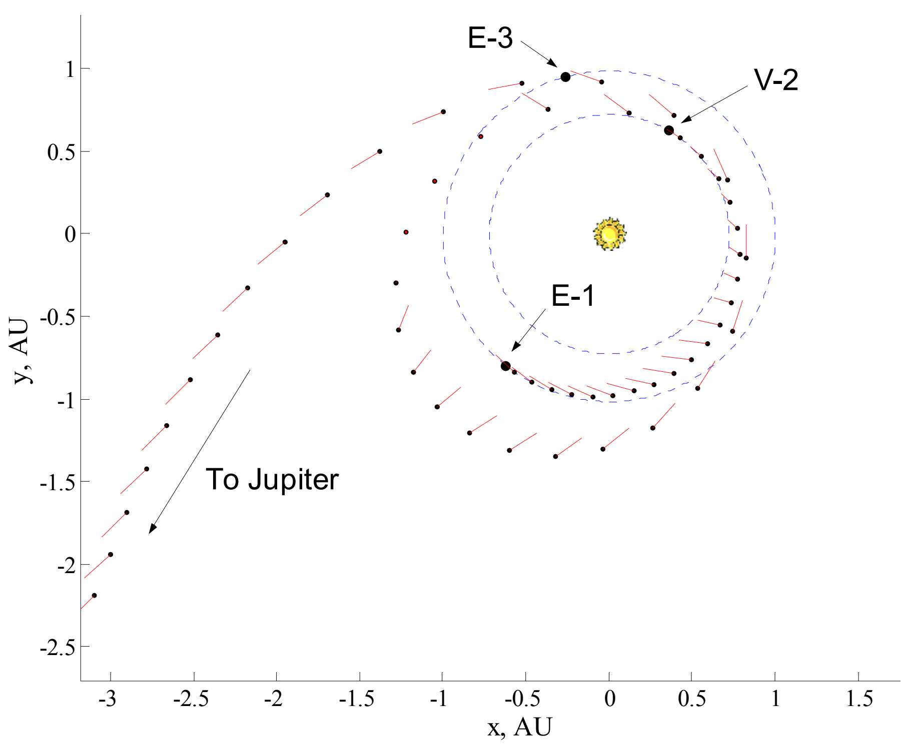

Figure 12: The Earth-Venus-Earth leg of the E-V-E-J-P rendezvous trajectory.

D. Representative LTGA Trajectories to Saturn, Uranus and Neptune

We found several trajectories to Saturn, Uranus and Neptune. Here we present our best cases to reach these three planets: Earth-Venus-Earth-Saturn (E-V-E-S), Earth-Earth-Jupiter-Uranus (E-E-J-U) and Earth-Venus-Earth-Jupiter-Neptune (E-V-E-J-N). Trajectory plots are shown in Fig. 13-15 and itineraries are provided in Table 8-10. The mission performance of these cases is discussed in the next section.

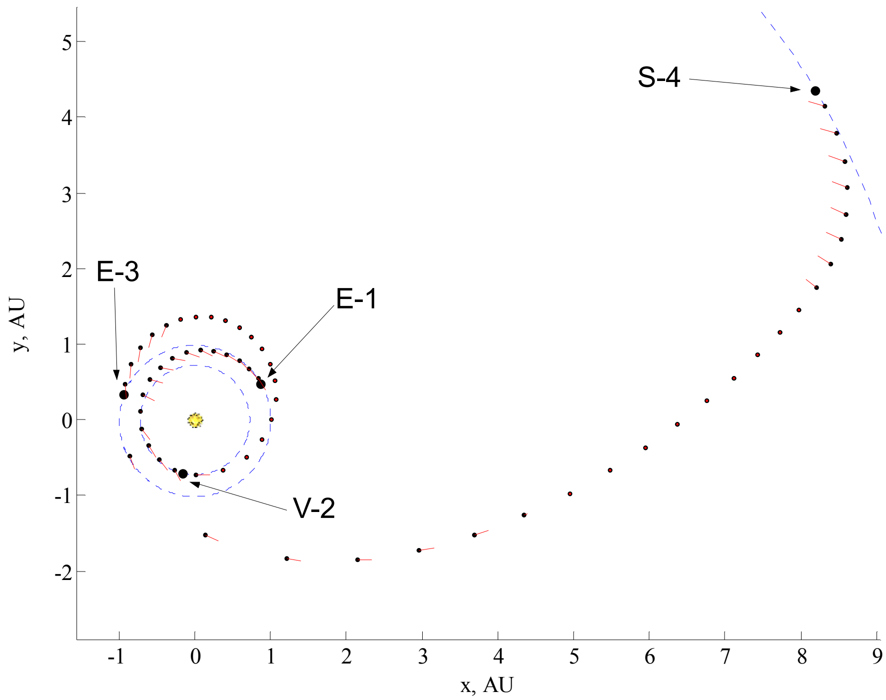

Figure 13: Trajectory plot of an Earth-Venus-Earth-Saturn rendezvous mission.

Table 8: E-V-E-S rendezvous trajectory

Characteristics	**Value**
E-1 Launch date	Oct 21, 2021
E-1 Launch V_∞, km/s	1.95
V-2 Flyby date	Apr 11, 2022
V-2 Flyby V_∞, km/s	5.30
V-2 Flyby altitude, km	8,265
V-2 Flyby B-plane angle[a]	5°
E-3 Flyby date	Mar 1, 2023
E-3 Flyby V_∞, km/s	9.47
E-3 Flyby altitude, km	500[b]
E-3 Flyby B-plane angle[a]	179°
S-4 Arrival date	Feb 7, 2028
Total TOF, years	6.30
Initial mass, kg	18,671
Final mass, kg	14,306
Propellant mass fraction	23%

[a] Fundamental plane taken as ecliptic of J2000.
[b] On lower bound.

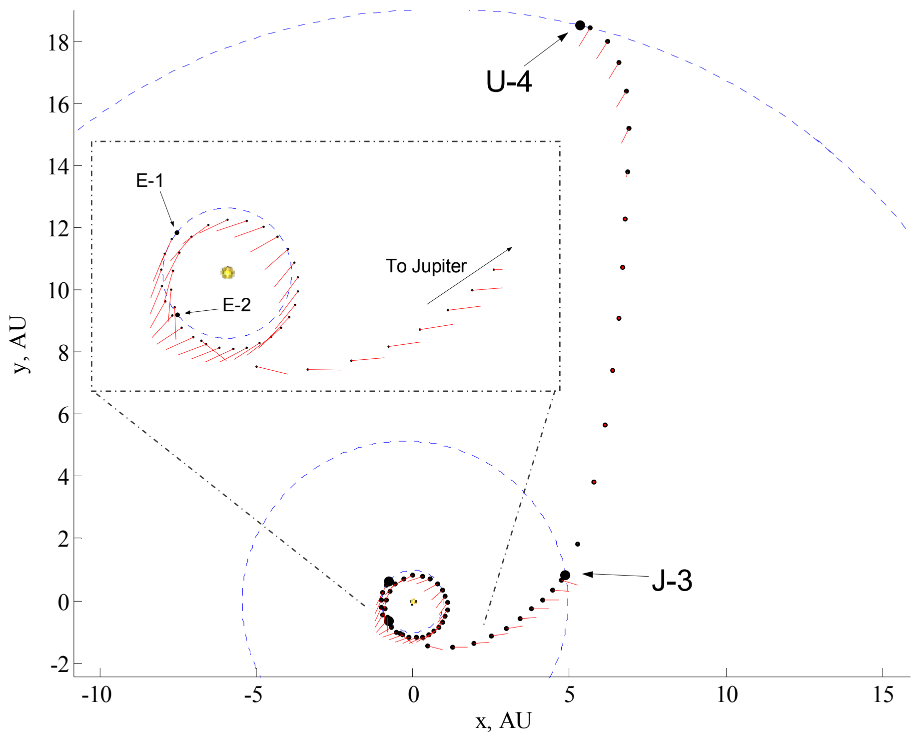

Figure 14: Trajectory plot of an Earth-Earth-Jupiter-Uranus rendezvous mission.

Table 9: E-E-J-U rendezvous trajectory

Characteristics	**Value**
E-1 Launch date	Feb 11, 2020
E-1 Launch V_∞, km/s	0.94
E-2 Flyby date	Mar 1, 2021
E-2 Flyby V_∞, km/s	8.25
E-2 Flyby altitude, km	500[b]
E-2 Flyby B-plane angle[a]	-1°
J-3 Flyby date	Dec 3, 2022
J-3 Flyby V_∞, km/s	11.8
J-3 Flyby altitude, R_J [c]	16.7
J-3 Flyby B-plane angle[a]	2°
U-4 Arrival date	June 3, 2029
Total TOF, years	9.31
Initial mass, kg	19,682
Final mass, kg	13,029
Propellant mass fraction	34%

[a] Fundamental plane taken as ecliptic of J2000.
[b] On lower bound.
[c] Assuming a Jupiter radius of 71,492 km.

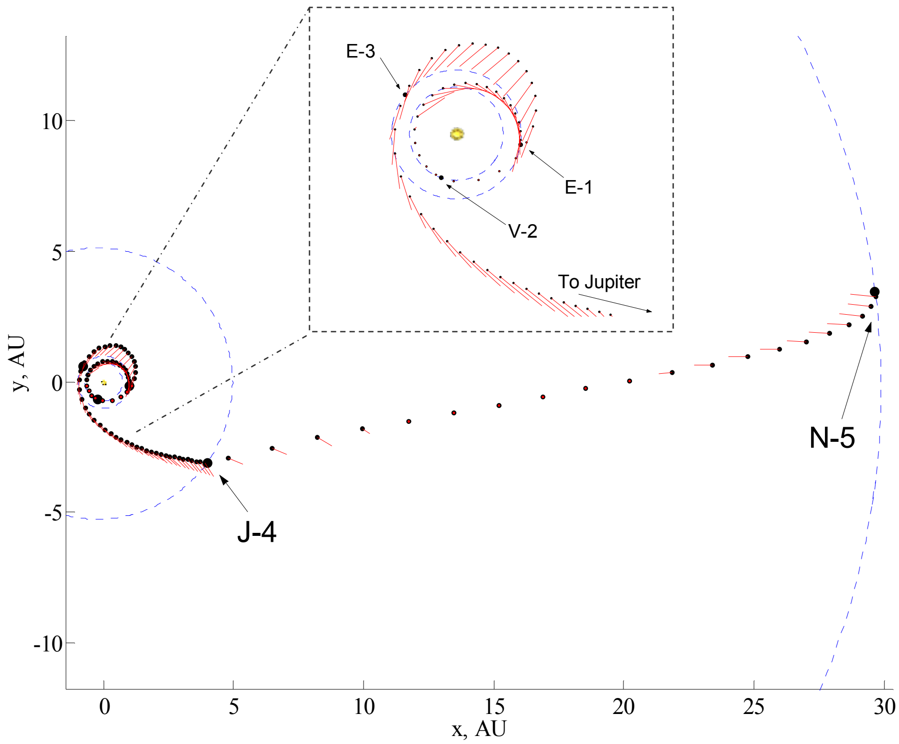

Figure 15: Trajectory plot of an Earth-Venus-Earth-Jupiter-Neptune rendezvous mission.

Table 10: E-V-E-J-N rendezvous trajectory

Characteristics	Value
E-1 Launch date	Sep 13, 2018
E-1 Launch V_∞, km/s	2.48
V-2 Flyby date	Mar 11, 2019
V-2 Flyby V_∞, km/s	6.37
V-2 Flyby altitude, km	500[b]
V-2 Flyby B-plane angle[a]	15°
E-3 Flyby date	Feb 12, 2020
E-3 Flyby V_∞, km/s	10.4
E-3 Flyby altitude, km	500[b]
E-3 Flyby B-plane angle[a]	-180°
J-4 Flyby date	June 26, 2021
J-4 Flyby V_∞, km/s	16.1
J-4 Flyby altitude, R_J [c]	14.4
J-4 Flyby B-plane angle[a]	0°
N-5 Arrival date	July 22, 2028
Total TOF, years	9.86
Initial mass, kg	17,882
Final mass, kg	9,264
Propellant mass fraction	48%

[a] Fundamental plane taken as ecliptic of J2000.
[b] On lower bound.
[c] Assuming a Jupiter radius of 71,492 km.

VI. Summary of Direct and LTGA Trajectories to the Outer Planets

The goal of these missions is to deliver spacecraft to the outer planets in a reasonable flight time and with an acceptable final mass (9,000 kg or more). In the launch years from 2014 to 2025, we study rendezvous missions for cases with and without gravity assist. Table 11 summaries the mission performance of the trajectories we have studied. All trajectories in Table 11 have TOF constraints with the encounter dates being free.

Table 11: Rendezvous missions to the outer planets

Encounter Sequence	Launch Date	Launch V_∞, km/s	Flight Times for each leg, days	Total Time of Flight, years	Final Mass, kg
Jupiter					
E-J[a]	Apr 13, 2024	0	2019	5.53	13,301
E-J	Apr 24, 2024	1.23	2019	5.53	13,709
E-M-J[a]	Jan 12, 2022	0	771, 1038	4.95	13,996
E-M-J	Feb 17, 2022	1.04	759, 1050	4.95	14,879
E-E-J	Sep 6, 2015	0.59	460, 1340	4.93	16,260
E-E-J	Sep 29, 2015	0.71	441, 1209	4.52	16,181
E-V-E-J	Aug 22, 2018	2.19	188, 347, 1115	4.52	15,606
Saturn					
E-S[a]	Oct 9, 2022	0	2631	7.20	11,285
E-S	Jan 24, 2023	1.30	2631	7.20	12,653
E-M-S	Feb 16, 2022	1.20	587, 2043	7.20	13,943
E-M-S	Apr 14, 2022	2.17	541, 1759	6.30	12,444
E-V-E-S	Oct 21, 2021	1.95	172, 324, 1804	6.30	14,306
E-V-E-J-S	June 4, 2015	2.24	179, 340, 648, 1283	6.71	12,872
Uranus					
E-U[a]	Oct 20, 2020	0	3537	9.68	8,268
E-U	June 5, 2021	2.32	3537	9.68	10,402
E-E-J-U	Jan 28, 2019	1.02	423, 599, 2378	9.31	12,912
E-E-J-U	Feb 11, 2020	0.94	445, 582, 2374	9.31	13,029
E-M-E-J-U	Dec 28, 2017	1.16	1039, 195, 515, 2450	11.5	14,097
E-V-E-J-U	Aug 25, 2018	2.21	184, 345, 587, 2284	9.31	12,172
E-V-E-J-U	Sep 27, 2018	2.81	172, 327, 526, 1976	8.21	9,225
Neptune					
E-N[a]	Aug 7, 2019	0	4114	11.3	6,347
E-N	Apr 21, 2020	2.63	4114	11.3	8,543
E-V-E-J-N	Aug 26, 2018	2.22	187, 348, 518, 3247	11.8	11,783
E-V-E-J-N	Sep 13, 2018	2.48	179, 338, 501, 2583	9.86	9,264
Pluto					
E-P[a]	May 28, 2015	0	4320	11.8	5,661
E-P	Aug 20, 2015	2.75	4320	11.8	6,890
E-J-P	May 29, 2014	1.11	1675, 3125	13.1	9,109
E-J-P	Feb 10, 2015	2.03	1706, 3214	13.5	8,666
E-E-J-P	Sep 2, 2015	2.44	424, 640, 2879	10.8	8,867
E-M-E-J-P	Mar 19, 2014	2.06	737, 285, 470, 2799	11.7	9,162
E-V-E-J-P	May 13, 2015	2.19	155, 448, 479, 2768	10.5	9,196

[a] The minimum TOF trajectory with launch V_∞ constrained to be zero.

For direct missions, we consider both zero and non-zero launch V_∞ cases as baselines for comparison with the gravity-assist trajectories. The final mass is acceptable only for the cases to Jupiter, Saturn and Uranus. For Neptune and Pluto, not only is the final mass too small, the TOF is too long (over eleven years). We therefore conclude that missions to Neptune or Pluto without gravity assist are infeasible.

We are interested to know what gravity-assist sequence would give us the best results (in terms of mission duration and propellant cost) to each outer planet. For Jupiter and Saturn, the case with a single Mars gravity assist allows us to have a shorter TOF and a higher final mass than the direct mission.

We also explore the potential of Earth (E-E) and Venus-Earth (E-V-E) gravity-assist sequences. From Table 11, we notice that both of the E-E and E-V-E trajectories are more effective than a single Mars gravity assist. In particular, for the mission to Jupiter with a flight-time of 4.5 years, the E-E-J final mass is 500 kg greater than the final mass of the E-V-E-J case. Therefore, the E-E-J trajectory is a better option than the E-V-E-J (when the TOF is 4.5 years). For a target beyond Saturn, we employ Jupiter as a gravity-assist body in addition to the E-E and E-V-E sequences. In a mission to Uranus with a TOF of 9.3 years, the E-E-J-U trajectory is found to be more effective than the E-V-E-J-U. The efficacy of the E-E sequence, however, does not extend to the Pluto mission. The best case to Pluto is found to be the E-V-E-J-P case, which has a slightly higher final mass and a shorter TOF than the E-E-J-P trajectory.

For missions to Saturn and Neptune, the best trajectories we have found are the E-V-E-S and the E-V-E-J-N respectively. However, this conclusion is made based upon the limited trajectories we have investigated.

VII. Future Work

1. Cleary, we have not exhausted all the gravity-assist sequences (such as EVEEJ and EVVVJ) that could improve mission performance (TOF and final mass).
2. Because of the efficiency of the Earth-Earth gravity-assist trajectories in the case of Jupiter and Uranus, we expect that missions to Saturn and possibly to Neptune could benefit from the sequence.
3. There may be launch windows where Jupiter is not well positioned for gravity assist to the far outer planets, which suggests we should find trajectories that do not use Jupiter gravity assist.
4. We also have to consider what gravity-assist sequences may be used if Earth is not allowed as a gravity-assist body.
5. There are other scientific targets of interest in the solar system, which include missions to Mercury, asteroids, comets and other solar system bodies. There is also interest in missions to escape the solar system such as an interstellar probe[47] and the 1000 AU mission.[48]

VIII. Conclusions

1. The nuclear electric propulsion system being designed for the JIMO spacecraft will enable direct missions to Jupiter, Saturn and perhaps Uranus, but not to Neptune and Pluto.
2. Gravity assists via Mars, Earth and Venus-Earth significantly reduce the TOF and increase the final mass to Jupiter and Saturn.
3. The inclusion of gravity assist with Venus, Earth or Mars in conjunction with a Jupiter gravity-assist further improves the mission performance to Uranus, Neptune and Pluto.
4. The JIMO mission will perform the most ambitious mission at Jupiter to-date. Its success will also open the door to solar system exploration.

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