

**INVESTIGATION AND OPTIMIZING OF INTERPLANATARY
TRAJECTORIES TO TRANS-NEPTUNIAN OBJECTS**

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Degree
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ABSTRACT

The outer reaches of the Solar System have great potential for scientific discovery. Unfortunately, visiting celestial bodies beyond Neptune, or trans-Neptunian objects, require lengthy mission durations with limited payload capacity. These missions are also costly due to the large infrastructure, manpower, and time required to plan and execute them. This thesis summarizes, analyzes, and discusses 14 years of research on interplanetary trajectories to trans-Neptunian objects completed by The University of Tennessee to discover methods of decreasing transit time and increasing payload mass. Additional research was then completed to investigate and expand upon these discoveries and optimize various aspects of the trajectories. There is a gap in the research done by The University of Tennessee pertaining to delta-VEGA maneuvers for orbiters and other complex inner planet trajectories to increase mass on target which were not investigated in this thesis but are important topics.

The summary revealed areas which could be optimized including powered/unpowered Jupiter gravity assists, two-stage capture burns, and dual-target missions. The pass distance for Jupiter flybys was found to have a substantial impact on transit times. Lower Jupiter periapsis flyby distances decrease transit times, and/or the required delta-v for powered flybys, but increase the amount of radiation experienced by the spacecraft. The propellant mass for each stage of a two-stage burn is dependent on the size of the burn and initial mass before the burn. A technique to optimize a two-stage burn is proposed which would increase the arrival mass for lander and orbiter missions. A dual-target mission architecture with a single launch vehicle is devised and optimized which couples two trajectories by their powered Jupiter flyby delta-v to find the minimum sum transit time to both targets. A mass calculation method deemed “mission mass fraction” was also proposed allowing missions to be evaluated for a range of potential launch vehicles quickly.

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CHAPTER ONE

BACKGROUND

Trans-Neptunian Objects

Trans-Neptunian objects (TNOs) include any object in the Solar System beyond the orbit of Neptune. The first TNO was discovered in 1930 by Clyde Tombaugh and is now known as Pluto [1]. Over 5,000 TNOs have been discovered since although the list continues to grow. The term trans-Neptunian object is a broad description which can be further broken down by the behavior and location of each object. Many TNOs' orbits are in the main part of the Kuiper belt located 30-50 AU from the Sun. The name comes from Gerard Kuiper who discussed the region in a paper in 1951 [2]. The outer region of the Kuiper belt is called the scattered disc which extends from 50-1,000 AU from the Sun. These objects often have highly elliptical orbits which can cause them to pass into the inner Kuiper belt as well. The outermost region of the solar system is the Oort cloud which is thought to begin 2,000-5,000 AU and extend to 10,000-100,000 AU from the Sun [3].

The most well-known TNO is Pluto which was also the primary target of the only probe sent to research these objects. Launching in 2006 the New Horizons probe encountered Pluto in 2015. The probe used the Pluto flyby to alter its trajectory towards Arrokoth which it encountered in 2019 [4]. The purpose of encountering these far-away objects is to gain a deeper understanding of how the Solar System was formed and uncover mysteries of the Universe itself. The major challenge of the mission is how lengthy the transit times were, which is due to the enormous distance between Earth and the TNOs. Since this mission, research at the University of Tennessee and elsewhere has provided hundreds of trajectories with different architectures to dozens of TNOs. With the abundant proof of plausibility, the new question becomes which trajectory architecture provides the optimum results.

My Purpose

Review extensive studies we and others have done on TNO mission design and then extend that work by examining various means of reducing transit time to these distant targets using currently available technology while still providing sufficient mass.

University of Tennessee Research

The University of Tennessee, Knoxville (UTK) has done research into interplanetary mission design to Trans-Neptunian objects for 14 years. During this time 10 papers have been published, one by the Journal of British Interplanetary Science [5] and nine by the American Astronautical Society [6, 7, 8, 9, 10, 11, 12, 13, 14, 15], all of which can be found online [16]. Trajectory architectures in this research include direct, unpowered and powered Jupiter gravity assists, unpowered Saturn gravity assists, Delta Velocity-Earth-Gravity-Assist (delta-VEGA), low thrust, or some combination. **Table 1.1** contains orbital information on the 34 TNOs which have previously been investigated by the University of Tennessee.

The University of Tennessee has not only researched various mission architectures but also different mission objectives. Groups have investigated flyby, orbiter, and lander missions including missions which visit multiple TNOs. There are 269 flyby, 94 orbiter, 12 lander, and 4 multi-target missions for a total of 379 missions from The University of Tennessee included in this paper which have their parameters and results tabulated in the appendix. The length of the mission, or transit time, varies considerably depending on the mission architecture and objective.

Outside Research

Many other researchers have contributed many findings to the same areas of interest. The research conducted outside the University of Tennessee can be categorized into two groups: current technology and theoretical technology.

Current Technology

Current technology focuses on mission design and feasibility which align more with this paper. A large contributor to this area of research is Zubko [18, 19, 20, 21, 22]. He has written multiple papers researching complex trajectories mostly focusing on the TNO Sedna. His trajectories utilize inner planet flybys to decrease the total Δv of the missions and outer planet flybys to decrease transit times. Mainly Venus and Earth in various combinations are used for inner planet flybys.

Table 1.1. List of Trans-Neptunian Objects Previously Considered by UTK.

(a)

TNO	Semi-major Axis	Eccentricity	Source
1996 TS66	44	0.13	[6]
1998 WW31	44.95	0.081	[17]
1999 TC36	39.8	0.232	[17]
2001 UR163	51.634	0.287	[7]
2002 AW197	47	0.13	[8]
2002 KX14	38.64	0.046	[17]
2002 MS4	42	0.14	[7]
2002 UX25	43.02	0.146	[17]
2002 XW93	37.353	0.254	[7]
2003 AZ84	39.476	0.176	[17]
2003 QA91	44.63	0.067	[7]
2007 UK126	74.389	0.494	[17]
Albion	43.925	0.071	[6]
Arrokoth	44.624	0.044	[7]
Borasisi	43.981	0.092	[6]
Chaos	45.48	0.098	[6]
Duecalion	44.287	0.067	[6]

(b)

TNO	Semi-major Axis	Eccentricity	Source
Eris	87.8	0.44	[5]
Haumea	43.2	0.19	[7]
Huya	39.694	0.281	[6]
Ixion	39.802	0.246	[6]
Kagara	43.901	0.089	[7]
Lempo	39.388	0.225	[6]
Makemake	45.43	0.161	[6]
Manwe	43.743	0.11	[6]
Orcus	39.174	0.227	[7]
Pluto	39.45	0.244	[6]
Quaoar	43.1	0.04	[7]
Rhadamanthus	38.773	0.155	[6]
Salacia	42.186	0.106	[7]
Sedna	495	0.85	[7]
Typhon	37.876	0.536	[7]
Varda	46.11	0.143	[7]
Varuna	43	0.05	[7]

A majority of his trajectories also use Jupiter as an outer planet flyby with other planets including Saturn and Neptune receiving consideration as well. He also lists asteroids which could be encountered as secondary targets while en route to the main TNO target. Note that Zubko has other publications not listed which are either in Russian or published on Russian websites which cannot be accessed.

Return to the Kuiper Belt is another notable source by Zangari et. al [23]. This paper conducted a widespread feasibility investigation into trajectories to over 45 TNOs. Many outer planet flybys were considered including Jupiter, Saturn, Neptune, Uranus, or some combination. Over 100 trajectories were presented each with multiple C3 options and corresponding transit times.

Hibberd and Eubanks investigated trajectories to an Oort cloud comet (UN271) which is approaching its perihelion along its about 2-million-year period [24]. With its aphelion at the far reaches of our solar system (about 33,000 AU), it is an interesting target which has been discovered early enough to plan a trajectory for. Hibberd and Eubanks present a wide range of trajectory architectures and launch dates which range from direct missions to complex inner and outer planet flybys including Earth, Venus, and Jupiter. These complex missions provide C3 values reaching as low as $8.4 \text{ km}^2/\text{s}^2$ with a high in-flight Δv tradeoff.

Recent work by Hayes et al has shown that metal oxide coatings can provide radiation shielding with significantly less mass than traditional aluminum enclosures [25, 26, 27, 28, 29]. Application of these materials (which were developed for Earth orbital operations) to Kuiper belt missions could likely allow closer Jupiter flybys while still providing acceptable risk of radiation-induced damage.

Near Future Technology

There are papers which take proposed, theoretical technology and investigate trajectories which could be completed with them. Many trajectories are planned a decade or more in advance, so these near-term technology trajectories may be viable by their launch dates.

Stern et. al. [30] investigated a direct Earth-Jupiter-Pluto orbiter mission with a low thrust electric propulsion powered by a radioisotope thermoelectric generator (RTG)

(present technology). The RTG is no longer in production and the mission requires the development of a more powerful ion thruster. The low thrust propulsion is used to slow the spacecraft down en route to Pluto causing a lengthy transit time (about 31 years). A nuclear fission reactor would be required to decrease the mission time further. There is enough propellant left to depart Pluto and encounter other objects. A list of possible post-Pluto targets and the corresponding mission parameters are also listed in the paper.

An early paper by Poncy et. al. [31] proposed multiple low thrust missions to Haumea. The various transit times and payload masses are found using various thrust (0.1 and 0.5), ISP (3,000 to 30,000 s), and specific power (48, 24, and 16 W/kg) values. While smaller thrust and ISP values are currently available with present low thrust technology, larger values towards 0.5 N thrust and 30,000 s ISP are not yet feasible.

Low Technology Readiness Level

Low technology readiness level (TRL) refers to technology which will not be available for the foreseeable future.

One such technology is a direct fusion drive (DFD). This device is a steady state nuclear fusion reactor which would provide both thrust and power to the spacecraft. Aime et. al. [32] states that it could provide a thrust between 1-10 N and an ISP of 1,000 to 100,000 s. However, a thrust of 8 N and ISP of 10,000 s were chosen for subsequent trajectories. Makemake, Haumea, and Eris were investigated and were found to have trajectories with in-flight Δv 's exceeding 25-30 km/s with transit times of about 6-10 years. Thomas [33] proposed a less than 4-year Pluto orbiter mission with a DFD (5-10 N thrust) inserting 1,000 kg of payload into orbit.

An enveloping aerodynamic decelerator (EAD) is an aerodynamic body which uses a TNO's atmosphere to slow down a landing vehicle. Warnecke, Nock, and Aaron [34] proposed a Pluto lander using such a device. The goal would be to slow down the spacecraft using an EAD and perform the remaining delta-v required for landing by burning propellant. The EAD would provide a delta-v of nearly 14 km/s and the remaining 90 m/s would be completed by the landing vehicle.

Trajectory Architectures

The simplest trajectory architectures are direct missions which launch from Earth and head directly to the TNO of interest. These missions are the starting point for trajectory research and are not commonly the final product; therefore, they are excluded in this paper.

Jupiter Gravity Assist

Adding a Jupiter gravity assist (JGA) to a mission adjusts the trajectory so the respective gas giant is encountered after launching from Earth. The probe does a flyby and “slingshots” around the planet towards the TNO of interest which reduces the amount of energy needed at launch. This required launch energy is called characteristic energy (C3) which is used to determine the allowable outbound mass from Earth. During this maneuver, the spacecraft increases its heliocentric (sun-centered) velocity by utilizing the respective planet’s gravitational energy and angular momentum. The energy and momentum are transferred from the planet to the spacecraft, keeping the total energy and momentum of the system conserved, giving the spacecraft the required energy to complete the mission. The magnitude of the velocity with respect to the flyby planet is not affected, just its direction [35]. **Figure 1.1** compares direct (top) and JGA (bottom) ecliptic projections. The direct mission requires a curved, elliptical trajectory to reach the target while the JGA allows a nearly direct path due to a directional change to the velocity (bend angle). A greater bend angle results in a more effective gravity assist.

Powered Jupiter Gravity Assist

Some missions utilize a powered Jupiter gravity assist (PJGA) which includes a propulsive maneuver at the minimum pass distance, or periapsis, around Jupiter. The PJGA accomplishes the same feats as the JGA while also increasing the magnitude of the velocity. The reason for this approach, instead of completing a burn somewhere else along the flight path, is to maximize the energy acquired from Jupiter’s gravity. Approaching a planet is similar to falling into a hole. The planet’s gravity drags the spacecraft in towards itself while the spacecraft’s angular momentum pulls it back out.

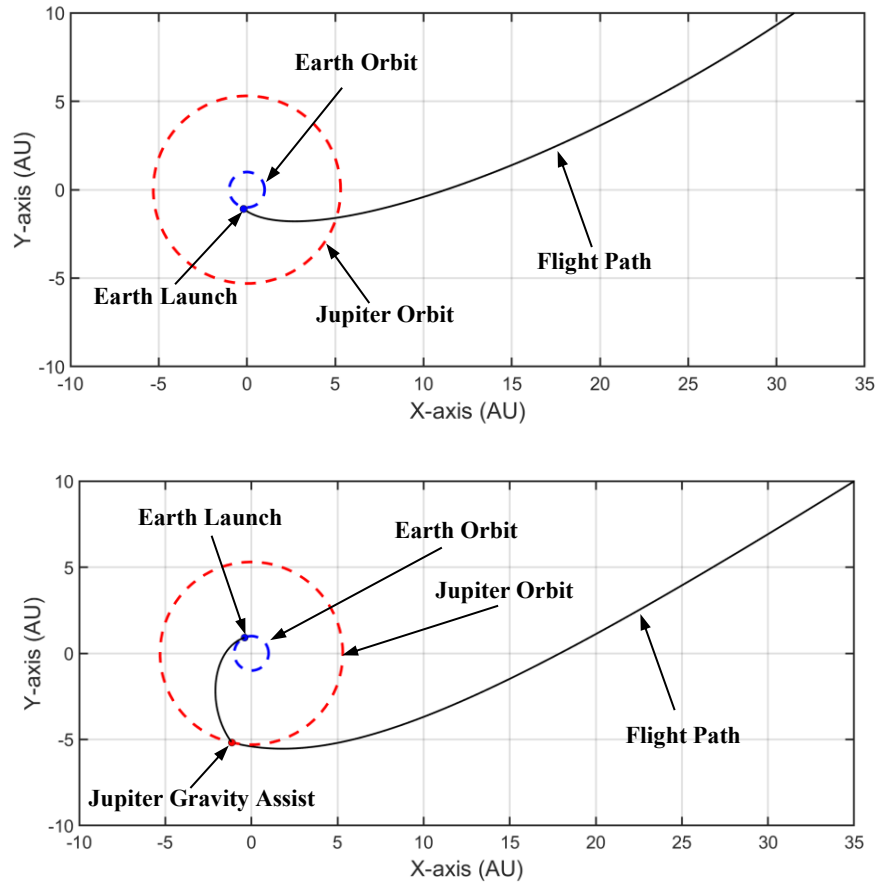


Figure 1.1. Two-dimensional ecliptic projections of a direct Earth to target mission (top) and an Earth to Jupiter gravity assist to target mission (bottom).

It's analogous to a ball rolling down a hill and up another identical hill. Assuming no losses, which is nearly the case in space, the ball will roll up the same distance it rolled down. A spacecraft falling into and out of a planet's gravity acts the same way during a gravity assist. However, a powered gravity assist (PGA) dumps mass via a burn at the "lowest point in the hole" (i.e., the periapsis) meaning it will require less energy to escape the "hole" (i.e., the planet's gravitational pull) therefore maximizing the energy acquired from the maneuver by capturing the potential energy from the propellant. An example of a gravity assist and where the burn would occur for a powered gravity assist is shown in **Figure 1.2**.

Delta-V Earth Gravity Assist

A delta-VEGA launches from Earth, performs a space burn, reencounters Earth and performs a gravity assist, either powered or unpowered, before proceeding on with the remainder of its mission. The original Earth departure puts the spacecraft into a heliocentric orbit with a period of about two years before it reencounters Earth. The space burn is performed near the apoapsis of the initial heliocentric orbit to target the Earth reencounter. A visual is provided in **Figure 1.3**. This reduces the C3 more than a JGA which allows for an even larger outbound mass from Earth. However, the delta-VEGA maneuver adds about two years to the mission's transit time. **Table 1.2** lists the TNO targets and architectures previously examined by the UTK team and indicates the paper in which each of those results are detailed.

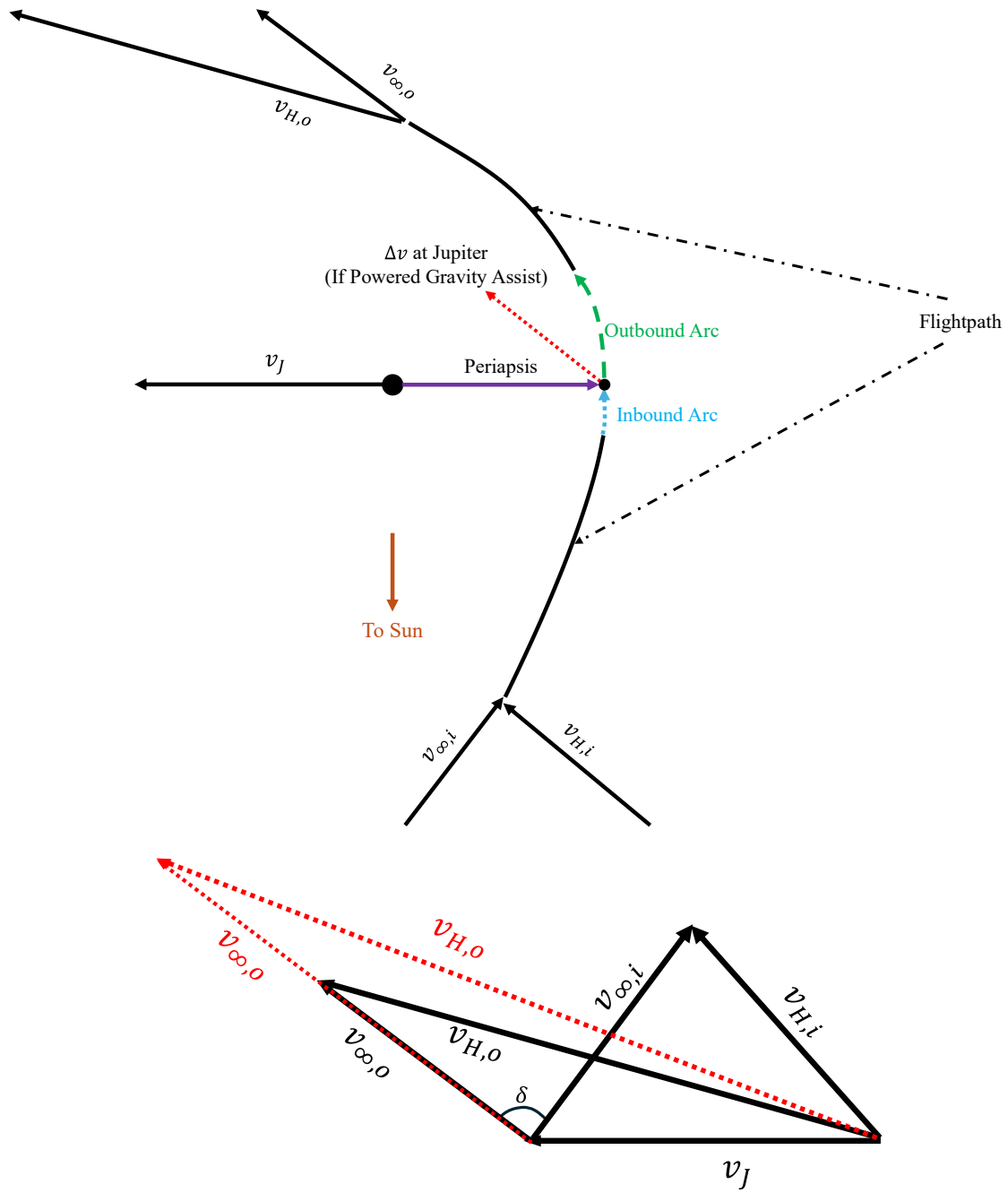


Figure 1.2. Example of a gravity assist (and powered gravity assist in red) around Jupiter with velocity vectors.

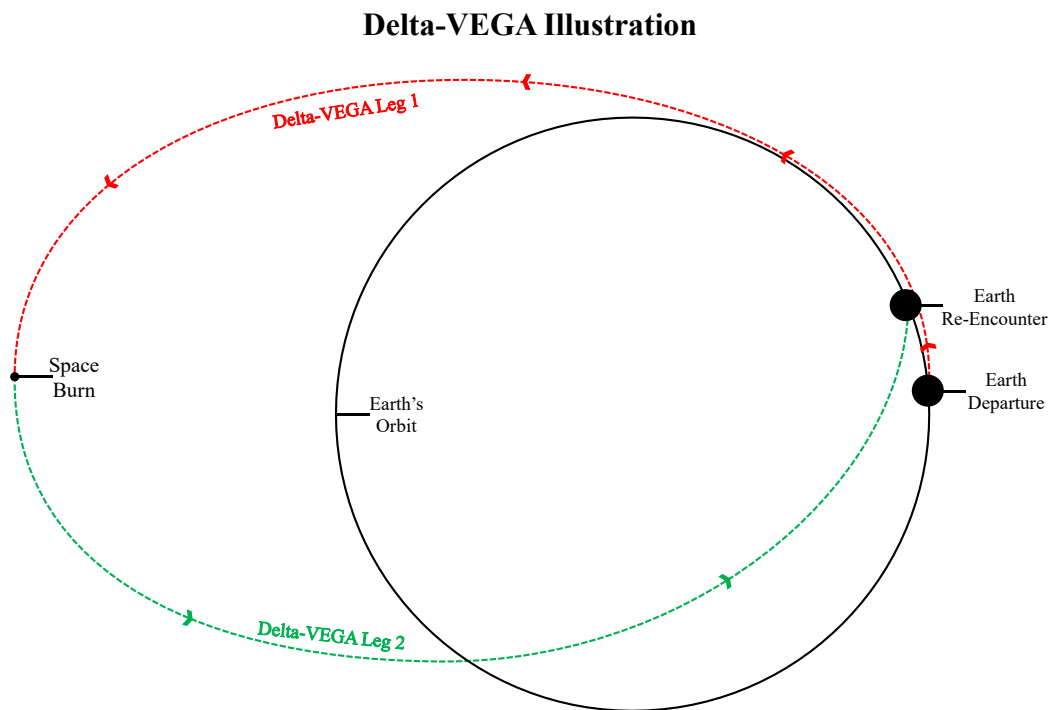


Figure 1.3. Delta-VEGA illustration.

Table 1.2. Trajectory architectures considered for each potential target.

(a)

Flyby					
	High Thrust				Low Thrust
TNO	JGA	PJGA	Delta-VEGA JGA	Delta-VEGA PJGA	JGA
2001 UR163	-	-	[7]	-	-
2002 AW197	-	-	[8]	-	-
2002 KX14	-	-	[7]	-	-
2002 MS4	-	-	[8]	-	-
2002 XW93	-	-	[7]	-	-
2003 QA91	-	-	[7]	-	-
Arrokoth	-	-	[7]	-	-
Borasisi	-	-	[7]	-	-
Chaos	-	-	[7]	-	-
Duecalion	-	-	[7]	-	-
Eris	[5]	-	-	-	-
Haumea	[5, 9, 11]	-	-	-	[9, 11]
Huya	[11]	-	-	-	[11]
Ixion	[11]	-	-	-	[11]
Kagara	-	-	[7]	-	-
Lempo	-	-	[7]	-	-
Makemake	[5, 9]	[14]	[8]	-	-
Orcus	[14]	[14]	[7]	[14]	-
Quaoar	[5, 9, 11]	-	-	-	[9, 11]
Salacia	-	-	[7, 8]	-	-
Sedna	[15]	[14]	[15]	[14]	-
Typhon	-	-	[7]	-	-
Varda	-	-	[7]	-	-
Varuna	-	-	[7]	-	-

(b)

Orbiter					
	High Thrust				Low Thrust
TNO	JGA	PJGA	Delta-VEGA JGA	Delta-VEGA PJGA	JGA
2002 AW197	[9]	-	-	-	-
2002 TC302	[9]				
2003 AZ84	[9]	-	-	-	-
2007 UK126	[9]	-	-	-	-
Haumea	[9]	-	-	-	-
Huya	[10, 9]	-	-	-	[10]
Ixion	[10, 9]	-	-	-	-
Makemake	[9]	-	-	-	-
Orcus	[10]	-	[7]	-	-
Quaoar	[9]	-	-	-	-
Rhadamanthus	[10]	-	-	-	-
Salacia	[10]	-	-	-	-
Varuna	[10]	-	-	-	-

Table 1.2 Continued

(c)

Lander					
	High Thrust				Low Thrust
TNO	JGA	PJGA	Delta-VEGA JGA	Delta-VEGA PJGA	JGA
1996 TS66	-	-	[6]	-	-
Albion	-	-	[6]	-	-
Borasisi	-	-	[6]	-	-
Chaos	-	-	[6]	-	-
Deucalion	-	-	[6]	-	-
Huya	-	-	[6]	-	-
Ixion	-	-	[6]	-	-
Lempo	-	-	[6]	-	-
Makemake	-	-	[6]	-	-
Manwe	-	-	[6]	-	-
Pluto	-	-	[6]	-	-
Rhadmanthus	-	-	[6]	-	-

(d)

Joint Missions					
	High Thrust				Low Thrust
TNO	JGA	PJGA	Delta-VEGA JGA	Delta-VEGA PJGA	JGA
1998 WW31	-	-	[12]	-	-
1999 TC36					
2002 UX25					
Huya	-	-	[8]	-	-
Quaoar					
Salacia	-	-	[7]	-	-
Neptune					
Makemake	-	-	-	[14]	-
Orcus					

CHAPTER TWO

METHODOLOGY

Trajectory Calculations

The methodologies described only apply to research conducted by The University of Tennessee. It is conducted to convert all trajectories into a standardized methodology which provides a direct comparison of results. This is also referred to as normalizing the results. Outside sources were not normalized and utilize other tools and mathematical methods which are not discussed here.

Mission Analysis Environment

All the trajectories were calculated using the Mission Analysis Environment (MANE) software [36]. This commercial software allows the user to input mission parameters and constraints and outputs a trajectory. The trajectory is either converged (i.e., plausible) or non-converged (i.e., not plausible). This paper only uses converged, plausible trajectories for the data.

There are three constraints in place for all missions to ensure a plausible trajectory. The first is a declination of the launch asymptote (DLA) between -28.5 and +28.5 degrees. Due to the latitude of Kennedy Space Center, this is the DLA range for launch on a direct injection without the need for a declination change maneuver. The second is a minimum Jupiter pass distance of 1.7 Jupiter radii. Performed by Pioneer 11, this is the closest successful flyby, other than the Juno spacecraft, which was heavily shielded, a probe has completed while remaining functional [37]. Due to the amount of radiation trapped by Jupiter's magnetosphere, closer approaches can damage sensitive onboard instruments. The final constraint is an Earth pass distance of 1.04 Earth radii to avoid entering the atmosphere. Galileo completed an Earth flyby with a pass distance of 1.045 Earth radii [38].

MANE allows the user to pick the mission architecture (e.g., Earth-JGA-TNO) and mission objectives (e.g., flyby). It also allows the user to constrain leg, nodal, and mission variables as shown in **Table 2.1** Leg variables control what the spacecraft does along its flight path.

Table 2.1. Examples of Mission Analysis Environment variables.

(a)

Mission Variables
Launch Declination/Perigee Offset
Thrust/Weight Ratio
Mission Duration/Flight Time
Initial Mass/Net Mass
Heliocentric Final Orbit

(b)

Node Variables	2	3	...
Stay Time			
Match Excess Speeds			
Passage Distance			

(c)

Leg Variables	1	2	...
Departure Time			
Arrival Time			
Inclination			
Flight Path Angle			
Departure v_{∞}			
Arrival v_{∞}			
Departure delta-v			
Arrival delta-v			

Nodal variables control the trajectories' characteristics (e.g. pass distance or delta-v) at each node (e.g., a planet or space burn). Finally, the mission variables affect the entire mission, such as the mission duration.

Payload Mass

The three most important attributes to a mission, assuming a converged and plausible trajectory, are C3, payload mass, and transit time. Transit time and C3 are calculated directly in MAnE, however the payload mass is not. While the payload mass for each trajectory is recorded in their respective papers, the assumptions and launch vehicle vary between them. To have direct comparisons for analysis, the payload masses were normalized to the same launch vehicle and with the same assumptions for equations. To normalize the payload mass, a mix of curve fits and analytical equations were coded inside Matrix Laboratory (MATLAB) coding software.

Launch Vehicle

The first step to calculating payload mass is picking a launch vehicle as it directly impacts how much outbound mass can be launched from Earth. The more powerful the vehicle is, the more it can launch. However, usually as the vehicle performance increases, so does the launch cost. This trend has changed recently due to recoverable rocket boosters. A list of popular, high-performance rockets and their cost and status are found in **Table 2.2**.

The choice of launch vehicle ultimately comes down to what is needed for a specific mission. In the case of this paper, over 300 trajectories with varying mission needs are being normalized to compare with each other in the most objective way possible. This will lead to inefficiencies in some missions, especially with the orbiter and lander capture burns.

The launch vehicle chosen for all the missions is the Delta IV Heavy with Star-48 due to its high performance and wide range of available allowable launch mass data. While this rocket has been retired, it provides the needed C3 data for high energy launches. The performance curves are displayed in **Figure 2.1**, and the curve fit information for five launch vehicles are shown in **Table 2.3**.

It is important to note that different launch vehicles perform better at specific C3 values. For example, The Falcon Heavy Expendable would be ideal for missions using a delta-VEGA maneuver since the C3 is usually between $20 \text{ km}^2/\text{s}^2$ and $40 \text{ km}^2/\text{s}^2$. However, the Delta IV Heavy with Star-48 surpasses the Falcon Heavy Expendable for C3s greater than $60 \text{ km}^2/\text{s}^2$. This will also lead to inefficiencies for certain trajectories.

Normalization is done by finding the outbound mass from Earth using the C3 in a curve fit (**Figure 2.1**) created from NASA’s high performance launch data [39]. Inserting the missions C3, which is taken from MANE, into the curve fit provides the outbound mass from Earth.

If the mission architecture contains no space, flyby, or capture burns, this is the final payload mass. Any burn during the mission affects the payload mass due to the propellant, tanks, and propulsion system mass. It is important to note there are no assumptions for radiation shielding, propellant mass for minor corrective maneuvers, or hardware for onboard flight controls. These are all assumed to be a part of the payload mass.

Rocket Equation and Assumptions

The mass after a burn (m_f) is found using the rearranged rocket equation

$$m_f = \frac{m_i}{\exp\left(\frac{\Delta v}{g_0 I_{SP}}\right)} \quad (1)$$

Where Δv is the space burn and/or PGA velocity change and m_i is the initial mass before the burn. Earth’s gravitational constant (g_0) is 9.81 m/s^2 and specific impulse (I_{SP}) is 317 seconds. The I_{SP} was chosen due to NASA’s Juno mission.

Table 2.2. Launch vehicle costs.

Launch Vehicle	Cost	Status	Source
Delta IV Heavy	\$350M	Retired (2024)	[40]
Atlas V 551	\$150M	Retired (2025)	[41]
Falcon Heavy Expendable	\$150M	Active	[40]
Vulcan VC6	\$110M	Active	[42]
Falcon Heavy Recoverable	\$97M	Active	[40]

Allowable Outbound Mass From Earth Per C3for Various Launch Vehicles

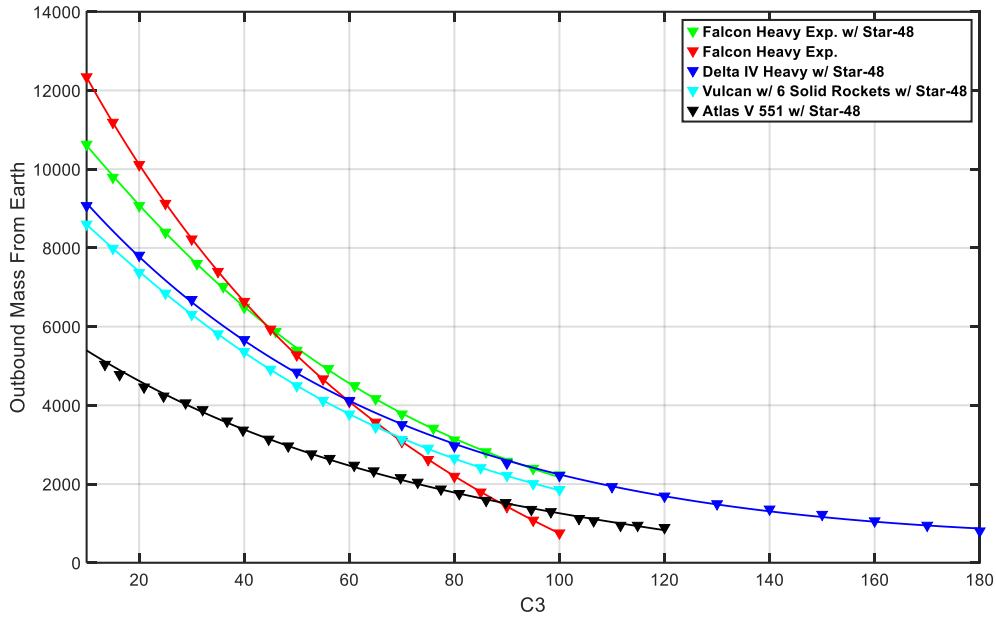


Figure 2.1. Curve fits to determine allowable outbound mass from Earth for various launch vehicles.

Table 2.3. Curve fit equations to determine allowable outbound mass from Earth for various launch vehicles.

Launch Vehicle	Curve Fit	R^2	Mass for C3 = 30	Mass for C3 = 100
Falcon Heavy Expendable with Star-48	$m = 14889e^{-0.02x} - 2694e^{-0.05x}$ [kg]	1.00	7570 [kg]	1996 [kg]
Falcon Heavy Expendable	$m = -874e^{-0.01x} + 15835e^{-0.02x}$ [kg]	1.00	8043 [kg]	1821 [kg]
Delta IV Heavy with Star-48	$m = 10661e^{-0.02x} + 81e^{0.01x}$ [kg]	1.00	5960 [kg]	1663 [kg]
Vulcan with 6 Solid Rockets with Star-48	$m = 11644e^{-0.02x} - 1812e^{-0.05x}$ [kg]	1.00	5986 [kg]	1563 [kg]
Atlas V 551 with Star-48	$m = -4.15e^{0.03x} + 6303e^{-0.02x}$ [kg]	1.00	3168 [kg]	781 [kg]

The Juno mission completed a thirty-five minute, $542 \text{ m/s } \Delta v$ orbital capture burn using Leros-1b hydrazine engine with an I_{sp} of 317 seconds [43, 44]. In the case of multiple burns, each mass calculation is done in chronological order with the m_f from the first burn being the m_i of the second burn and so on. To account for the mass of the tanks carrying the propellant for the burns, the tankage fraction mass is assumed to be 10% of the propellant mass (m_p). Propellant mass is found using

$$m_p = m_i - m_f \quad (2)$$

The tankage mass can be accounted for to find the usable payload mass by combining

$$m_f = m^* + 0.1m_p \quad (3)$$

$$m_i = m^* + 1.1m_p \quad (4)$$

Giving

$$m^* = 1.1m_f - 0.1m_i \quad (5)$$

If there is no capture burn (i.e., a flyby mission), this result is the payload mass (m^*). All constants are shown in **Table 2.4**.

The orbiter and lander missions follow the same process but also include a two-stage capture burn at the target. These capture burns, which are also their excess speed at target, are large ranging from 3.99 km/s to 10.72 km/s for orbiters and 2.87 km/s to 5.92 km/s for landers. The lander missions long transit times result in lower excess speeds than the orbiters. The lander vehicle is assumed to be in the payload mass as the specific lander needed will vary depending on the mission objective and the target.

Table 2.4. Values of constants used in payload mass calculations.

Constant	Variable	Assumed Value	Source
Earth's Gravitational Constant	g_0	9.81 m/s^2	-
Specific Impulse (1)	I_{SP} (First Stage Capture Burn)	290 s	[45]
Specific Impulse (2)	I_{SP} (Space Burn, PGA Burn, Second Stage Capture Burn)	317 s	[44]
First Stage Propellant Mass	m_p (First Stage Capture Burn)	630 kg	[45]

To calculate a two-stage burn, a propellant mass for the first stage is chosen based on the maximum storage of the first stage rocket. The first stage rocket chosen is the Star-30E. It has an I_{SP} of 290 s and a propellant storage of 630 kg [45]. Again, the rearranged rocket equation is used, but now it is solving for the Δv instead of the final mass (eq. 4).

The final mass is simply the initial mass subtracted by the mass of the stage-one propellant mass. The mass of the propellant and the tankage mass (i.e., an additional 10% of the propellant mass) is discarded from the stage-one final mass as the entire first stage is jettisoned. This is used for the initial mass of the second burn. The Δv for the second stage is the required burn for capture subtracted by the Δv of the first stage. Now eq. 1 can be used to solve for the mass after the second stage and eq. 2 for the final payload mass. If previous research papers did not provide the required data for all calculations, the trajectories were not included.

CHAPTER THREE

ANALYSIS OF PREVIOUS WORK

The University of Tennessee has been researching trajectories to trans-Neptunian objects for over a decade. An in-depth accumulation and analysis of this work has been completed in order to summarize their results and determine areas of future research. Outside sources have also been considered and reported when applicable but without any recalculations.

Flyby Missions

Payload mass and transit time are the most important trajectory attributes. **Figure 3.1** displays these two attributes for TNO flyby missions with only one trajectory. For **Chapter 3**, filled in markers represent low thrust trajectories and hollow markers are high thrust trajectories. Low thrust trajectories will also be noted in the legend, and any makers which cannot be hollow (e.g., x and *) always represent high thrust missions. The payload masses have been normalized by calculating all payload masses using the same rockets (**Table 2.3**) and constants (**Table 2.4**), however the transit times have not. The time of flight to TNOs depends heavily on their distance from the sun. Therefore, it is difficult to compare the time of flights to different TNOs.

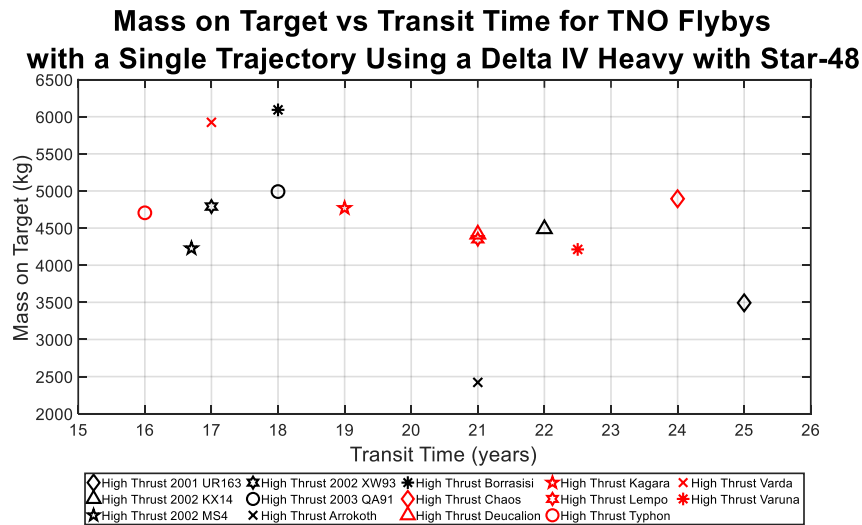


Figure 3.1. Payload mass vs transit time for targets with single flyby missions.

Fortunately, due to the extensive research done, a few select TNOs have enough data points to directly compare different mission architectures. The flyby missions contain the most extensive research of the three mission types. Flyby missions denote the spacecraft approaches its target, performs a swingby at a specific excess speed, and then continues its flight path. Eight TNO's have multiple flyby trajectories which are displayed in **Figure 3.2**. The most obvious trend is the superior results when a low thrust architecture is included which can be seen in the Haumea, Huya, Ixion, and Quaoar plots (**Figure 3.2**).

Low Thrust and High Thrust Comparison

Low thrust flyby missions were investigated by Baskaran et. al. [11]. The low thrust missions use low thrust NASA Evolutionary Xenon Thrusters (NEXT) powered by solar arrays for their Earth-Jupiter leg of the mission [11]. After the JGA, the sun does not provide enough energy for NEXT to be useable. By using NEXT, the required C3 is lowered to by $\sim 50 \text{ km}^2/\text{s}^2$ (**Figure 3.3**). The NEXT dry mass is 50 kg which is assumed to be a part of the payload mass.

Two other important parameters are excess speed at target and Jupiter pass distance. The excess speed determines the data collection period for a flyby and the required burn for any orbital insertion or lander and is preferred to be low. The Jupiter pass distance is prevalent when determining the radiation dosage the spacecraft experiences and is preferred to be high. **Figure 3.4** displays the excess speed at the target and Jupiter pass distance versus transit time of low thrust and high thrust missions. There is no clear trend comparing the two mission architectures since the Excess speeds and Jupiter pass distances are affected by multiple mission variables. For each individual mission, however, both the excess speed and pass distance have recurring trends. The shorter the transit times, the higher the excess speed since the spacecraft must move faster to get to the target faster. The pass distance at Jupiter lowers for shorter transit times as closer passes provide more efficient gravity assists. There is also the correlation of shorter pass distances at Jupiter result in higher excess speeds at the target due to the same principles.

Mass on Target vs Transit Time for TNO Flybys with Multiple Trajectories Using a Delta IV Heavy with Star-48

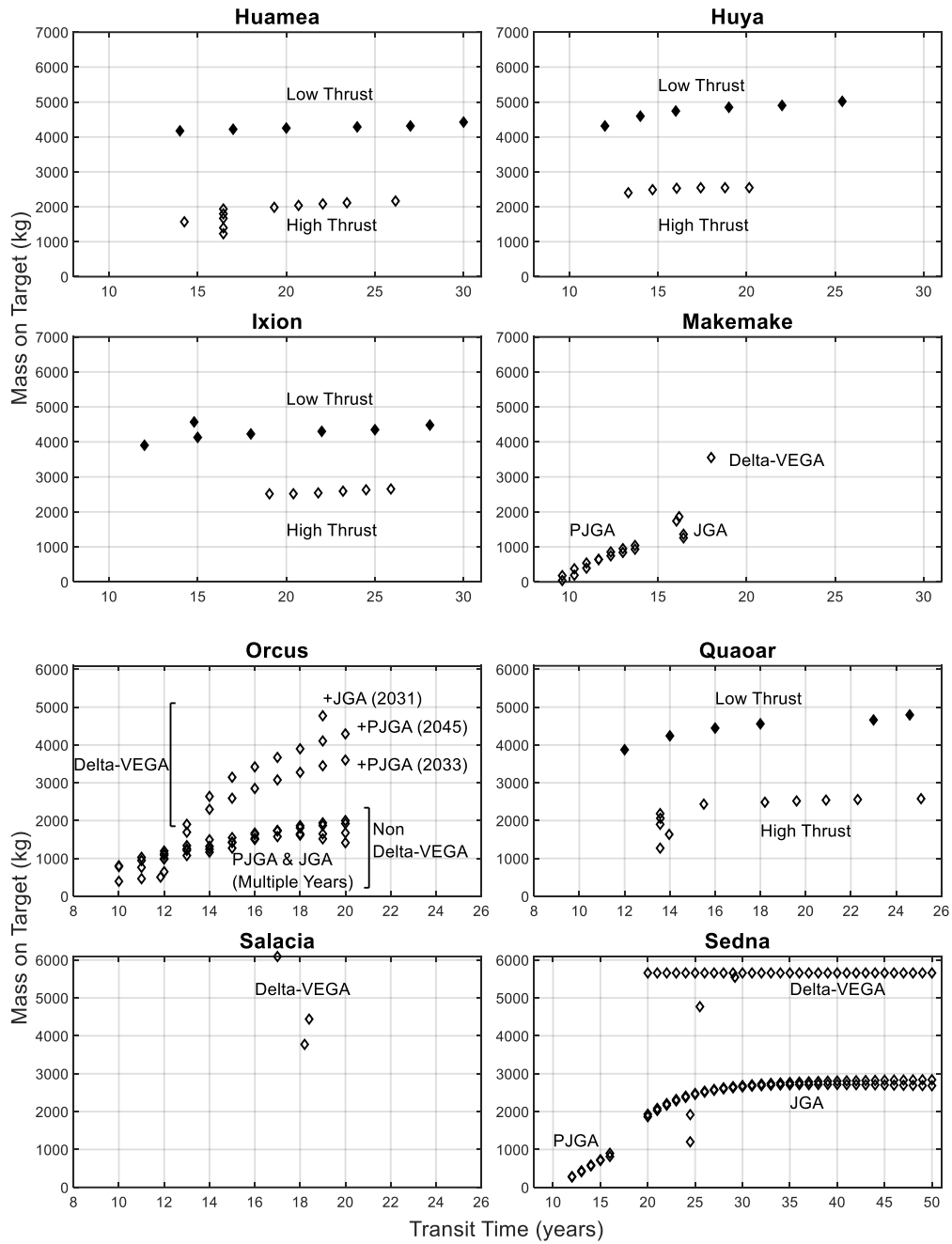


Figure 3.2. Flyby data for objects with more than one calculated trajectory. (x-axis scale varies to allow for adequate marker spacing for legibility)

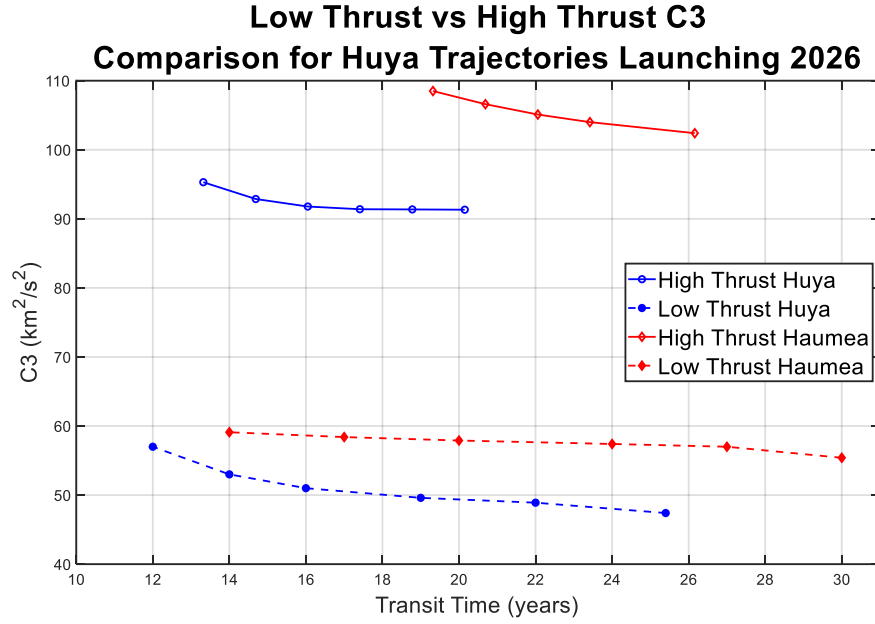


Figure 3.3. Comparison of low thrust and high thrust C3 requirements for the same launch year of 2026.

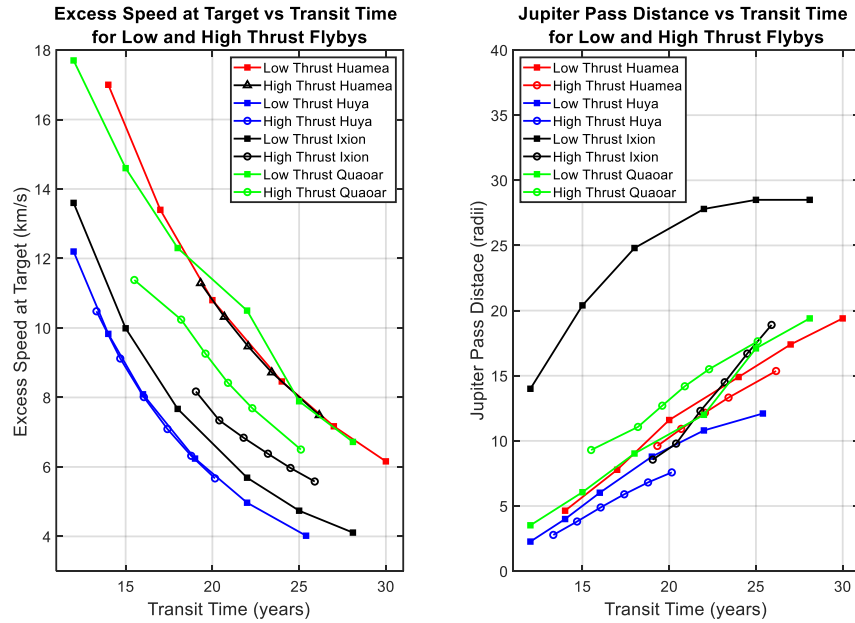


Figure 3.4. Excess speed at target and Jupiter pass distance vs transit times for low and high thrust missions to Haumea, Huya, and Ixion.

Architecture Comparison

Orcus has the largest data set for mission architectures. It includes powered Jupiter flybys with and without a delta-VEGA and unpowered Jupiter flybys without a delta-VEGA. **Figure 3.5** displays the different mission architectures from Allison et. al. and Anderson et. al. [7, 14].

For Orcus, there is no architecture which is always better than the others; it depends on the mission requirements. The Earth-delta-VEGA-Jupiter missions are superior for transit times exceeding 12-13 years proven by two separate launch years (2033 and 2045) in **Figure 3.5** However, Earth-Jupiter mission durations below 12.5 years transport more payload mass. The unpowered Jupiter gravity assist missions seem to provide the best results in this range; however, they have different launch years than the powered missions, so other variables may cause this discrepancy.

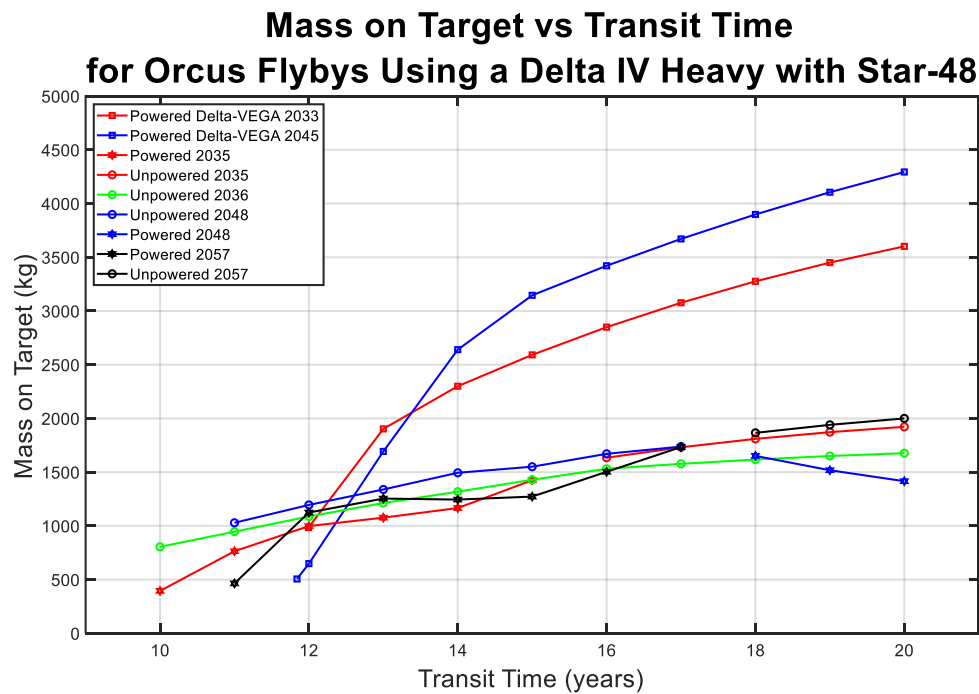


Figure 3.5. Comparing mass on target vs transit times for various mission architectures to Orcus.

Orbiter and Lander Missions

Orbiter and lander missions act the same as flybys until the target encounter where a capture burn is required.

Capture Burns

A capture burn is done to slow a spacecraft down and capture into orbit around or land on a target. For orbiters, the required velocity for a circular orbit can be calculated based on the radius of the target assuming it is a rocky body like Earth. The standard gravitational constant for the target (μ_{target}) can be calculated using

$$\mu_{target} = \left(\frac{r_{target}}{r_{Earth}}\right)^3 \mu_{Earth} \quad (6)$$

Where r_{target} is the radius of the target, r_{Earth} is the radius of Earth, and μ_{Earth} is Earth's standard gravitational constant. As the spacecraft approaches the target, it will speed up due to the target's gravitational pull. The new velocity at the periapsis radius is

$$v_{p,target} = \sqrt{2 \left(\frac{v_{\infty}^2}{2} + \frac{\mu_{target}}{r_{orbit}} \right)} \quad (7)$$

Where $v_{p,target}$ is the periapsis velocity of the spacecraft and v_{∞} is the excess speed at the target and r_{orbit} is the radius from the center of the target to the orbit ($r_{target} + 100 \text{ km}$ for this paper). For a circular orbit, the required orbital velocity is

$$v_{c,target} = \sqrt{\frac{\mu_{target}}{r_{orbit}}} \quad (8)$$

Where $v_{c,target}$ is the circular velocity required for orbit. The required capture burn to slow down the spacecraft to this speed is

$$\Delta v_{capture\ burn} = v_{p,target} - v_{c,target} \quad (9)$$

Where $\Delta v_{capture\ burn}$ is the required change in velocity to capture into orbit.

For landers, $v_{p,target}$ is calculated to account for the velocity gained as the spacecraft approaches a target. Since they will be landing, this will be the required burn to slow the spacecraft to a stop at the target's surface. Therefore, orbiter missions would benefit from larger targets while landers would benefit from smaller targets. The targets and TNOs and respective radii for orbiters and landers are listed in **Table 3.1**.

Note that flyby missions can be converted into orbiter or lander missions by taking their excess speed and applying a capture burn to recalculate their payload mass. Note the lander missions will require a landing vehicle which would further subtract from the payload mass. However, this was not included in the mass calculations and so the payload mass is assumed to include the mass of a landing vehicle.

The capture burn is assumed to have two stages with the first stage containing 630 kg of propellant and an ISP of 290 s (**Table 2.4**). The ISP of 290 assumes a solid rocket booster for the first stage. All the capture burns required a larger burn than could be provided by this single booster which sets the requirement for a two-stage rocket. The second stage assumes an ISP of 317 (liquid engine) and the propellant mass required to complete the remaining delta-v. Liquid fuel tanks can be made to size depending on the mission. Since all the missions have different excess speeds and mass, the method of solving the required propellant mass can be further optimized which is done in **Chapter 5: Two-Stage Burn Optimization**.

Orbiter Payload Mass

Unlike flybys, orbiters must complete a capture burn which requires a great deal of propellant along with rocket engines to conduct. Due to this, the mission's payload masses are drastically reduced; some to a point where the trajectory could not deliver instruments while carrying radiation shielding and extra propellant for corrective maneuvers. The payload mass for all orbiters is shown in **Figure 3.6**.

Table 3.1. Orbiter (left) and Lander (right) TNOs and their respective radii.

Orbiters		Landers	
<i>TNO</i>	<i>Radius (km)</i>	<i>TNO</i>	<i>Radius(km)</i>
2002 AW197	384	1996 TS66	80
2002 TC302	250	Albion	63*
2003 AZ84	362	Borasisi	63
2007 UK126	307	Chaos	300
Haumea	797	Deucalion	78*
Huya	206	Huya	206
Ixion	355	Ixion	355
Makemake	715	Lempo	136
Orcus	455	Makemake	715
Quaoar	556	Manwe	715
Rhadamanthus	73*	Pluto	1188
Salacia	433	Rhadmanthus	73*
Varuna	334		

* Estimated based on orbital characteristics. See [55] for details.

**Mass on Target vs Transit Time for TNO Orbiters
with Multiple Trajectories Using a Delta IV Heavy with Star-48**

(a)

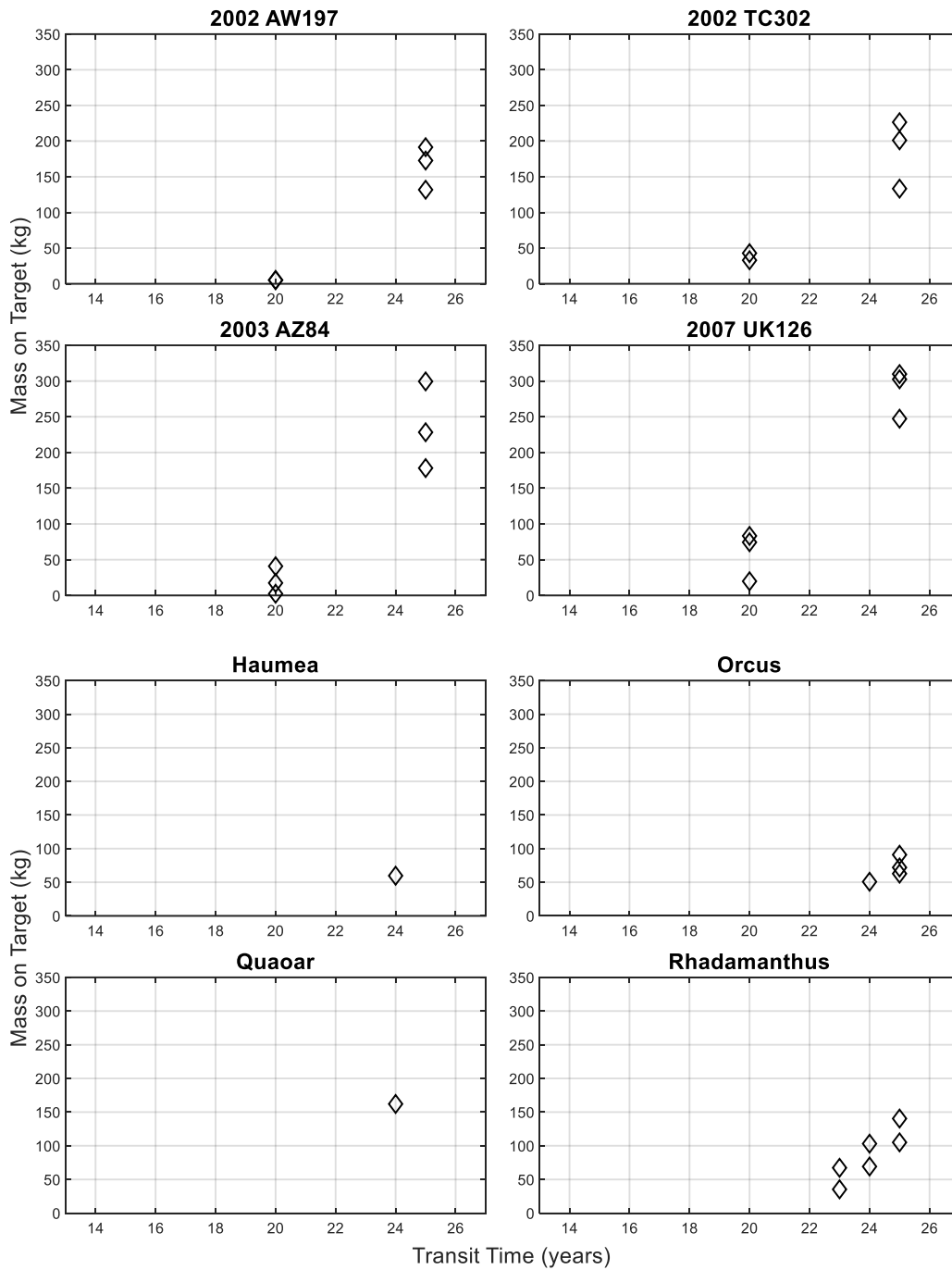


Figure 3.6. Payload mass for every orbiter trajectory vs transit time. (y-axis scale varies for final Ixion and Huya plot to display all data)

(b)

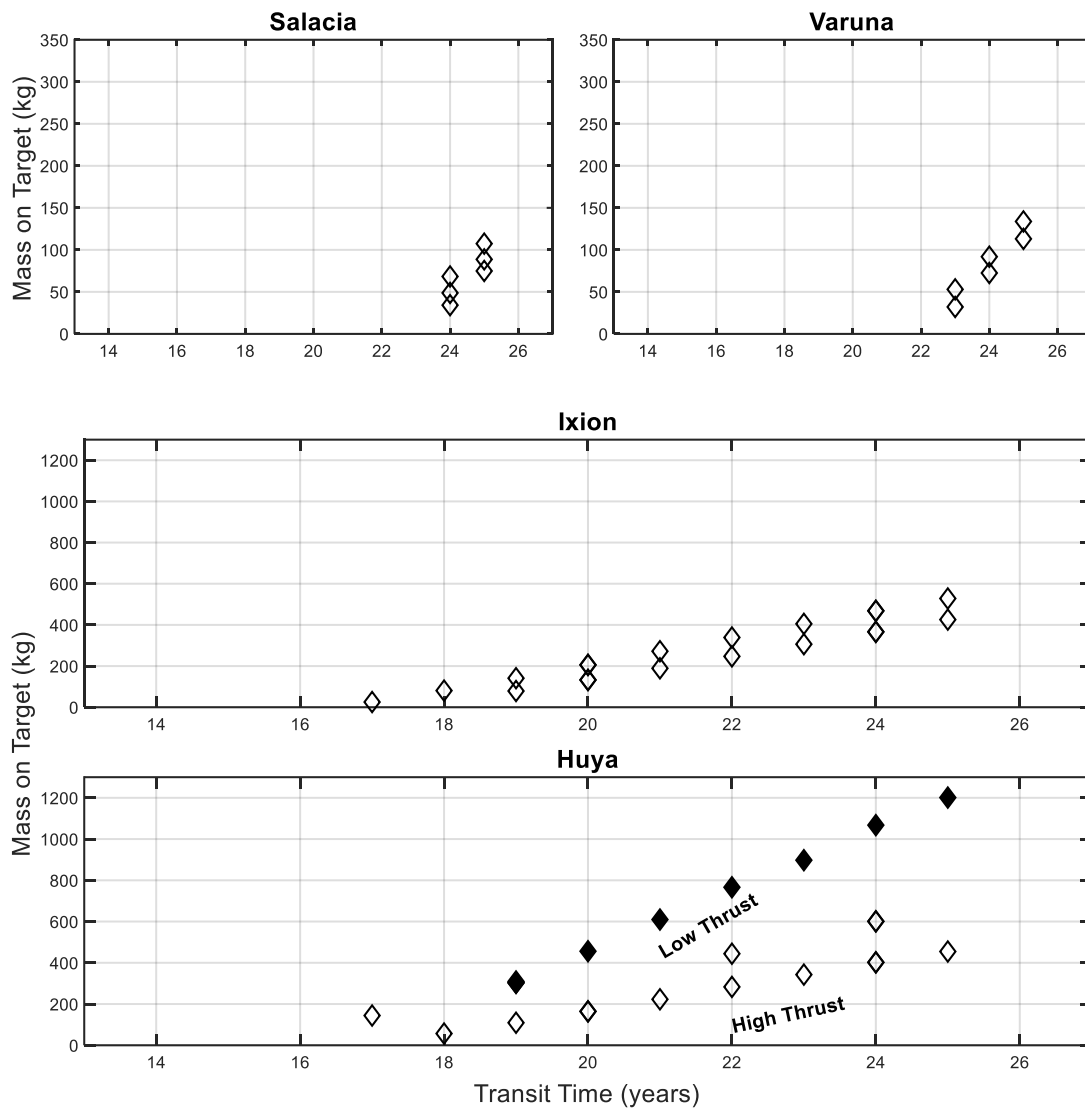


Figure 3.6 continued.

New Horizons, the only mission to visit trans-Neptunian objects, had a launch mass (i.e., total mass of the payload, propellant for corrective maneuvers, and other devices required for overall function) of 476 kg [46]. This launch mass would be impossible for almost all the trajectories under the current assumptions (**Table 2.4**). Only the slower transit times of the low thrust missions would be able to accomplish the feat. To allow for more payload mass, the missions need to increase transit times, lower C3 (i.e., use Delta-VEGA or low thrust architectures), or optimize capture burns.

Low Thrust Missions

It is again noted that the low thrust missions provide larger payload mass capabilities than high thrust missions (data from Gleaves et. al. [10]). It can also be seen that the low thrust missions receive more payload benefits when increasing transit times than high thrust missions (**Figure 3.7**). Implementing low thrust engines also decreases the C3 by $\sim 50 \text{ km}^2/\text{s}^2$ compared to an identical mission without low thrust. This is the same benefit as the flyby missions. The identical results are expected since the mission architecture of orbiters does not change until the capture burns.

Lander Payload Mass

The lander missions utilize longer transit times than orbiters to increase their even greater payload requirements due to the need for a landing vehicle. **Figure 3.8** displays the payload mass for the 11 lander trajectories (data from Ball et. al. [6]).

The lander missions generally have a much larger payload mass than the orbiter missions. The lander missions utilize a delta-VEGA maneuver to increase their allowable launch mass from Earth by decreasing the C3 (**Figure 2.1**). The lander missions also have extended mission durations which decreases the excess speed at the target. A longer transit time signifies the spacecraft travels slower resulting in a lower excess speed at the target. A lower excess speed allows for a smaller capture burn which increases the payload mass.

Comparison of Low Thrust and High Thrust Payload Mass And C3

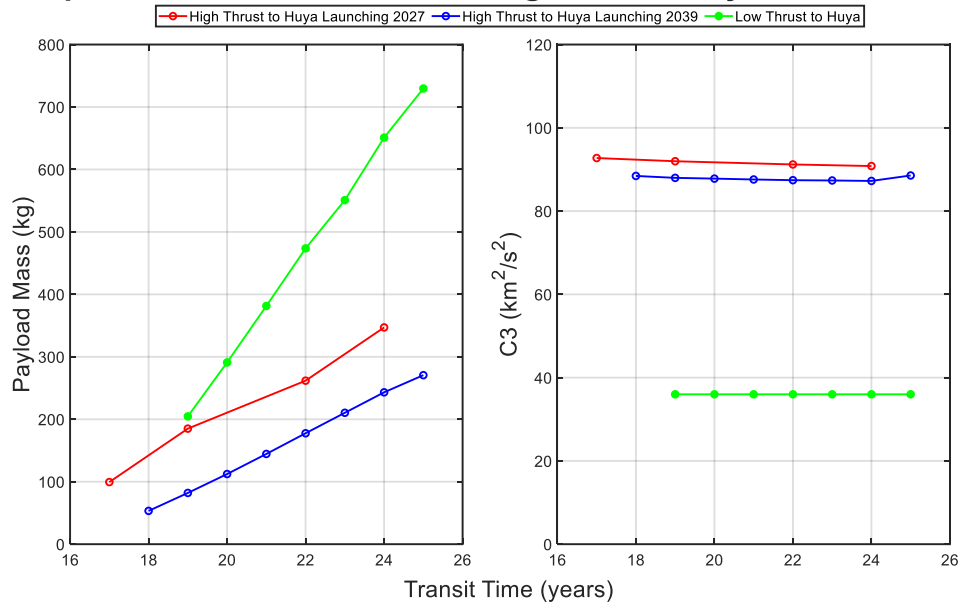


Figure 3.7. Comparison of low thrust and high thrust mission parameters payload mass (left) and C3 (right) for trajectories to Huya for orbital capture. Two launch years are represented for high thrust missions while the low thrust mission did not report the launch year [10].

Mass on Target vs Transit Time for Landers Using a Delta IV Heavy with Star-48

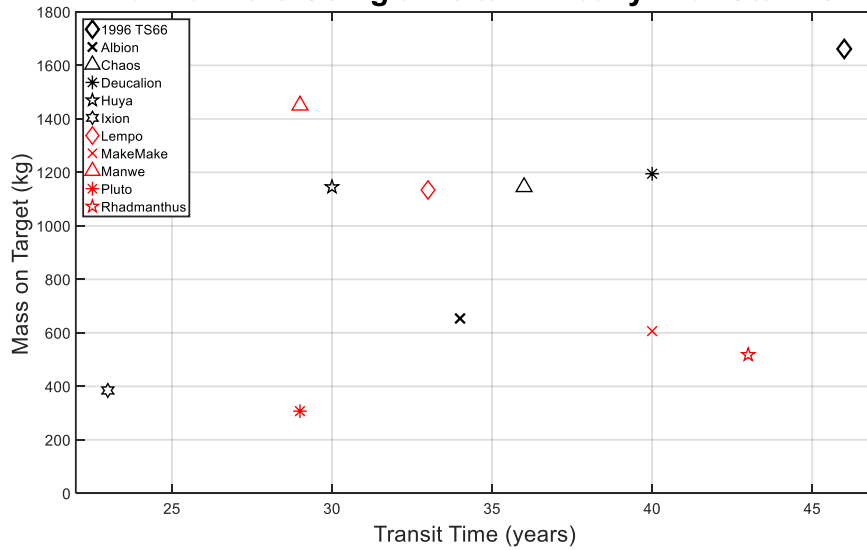


Figure 3.8. Payload mass for every lander trajectory vs transit time.

Jupiter Flybys

Jupiter flybys are often used to increase sun-relative velocity of interplanetary missions. The gas giant's gravity is used to change the direction of the trajectory and velocity relative to the sun. A spacecraft performs either an unpowered gravity assist, or powered gravity assist by introducing a burn ideally at the periapsis of the flyby (**Figure 1.2**).

Jupiter Pass Distance

Sedna is another TNO with extensive research. Sedna is reaching its perihelion (closest approach to the sun) of 310 AU in the year 2072. It is critical to intercept the dwarf planet during this approach because it has a period of 10,500 years with a highly elliptical orbit causing an aphelion (furthest approach from the sun) of 1020 AU [47, 48]. So, once the window passes by Sedna will move farther away from Earth increasing transit times.

Due to the unique circumstances of its orbit and scientific significance, Sedna is a popular target of research. The University has investigated the dwarf planet in multiple papers with one paper focusing solely on Sedna written by Brickley et. al. [15]². In this paper, an Earth-delta-VEGA-JGA-Sedna mission launching 2046 for a transit time range of 20 to 50 years was investigated. The C3 was kept nearly constant ($26 \text{ km}^2/\text{s}^2$) as was the space burn (620 m/s). This allowed the payload mass to be held constant as well. The driving factor for shortening the transit time was the Jupiter pass distance (**Figure 3.9**).

This shows how effective decreasing the Jupiter pass distance can be for decreasing the mission's transit time. A decrease of almost 7 Jupiter radii provides a transit time that is 30 years faster. The limit of this approach is twofold. The obvious is that a spacecraft could only approach up to 1 Jupiter radius before entering the planet's atmosphere. This would most likely cause the destruction of the spacecraft. The other is a problem of radiation produced by Jupiter.

² The reported payload mass numbers for the lander section of this paper did not include tankage fraction and therefore are inaccurate. This mistake is rectified in **Chapter 5: Sedna Lander Mass Correction**.

Transit Time for vs Jupiter Pass Distance Earth-Delta-VEGA-JGA-Sedna Trajectory Launching 2046

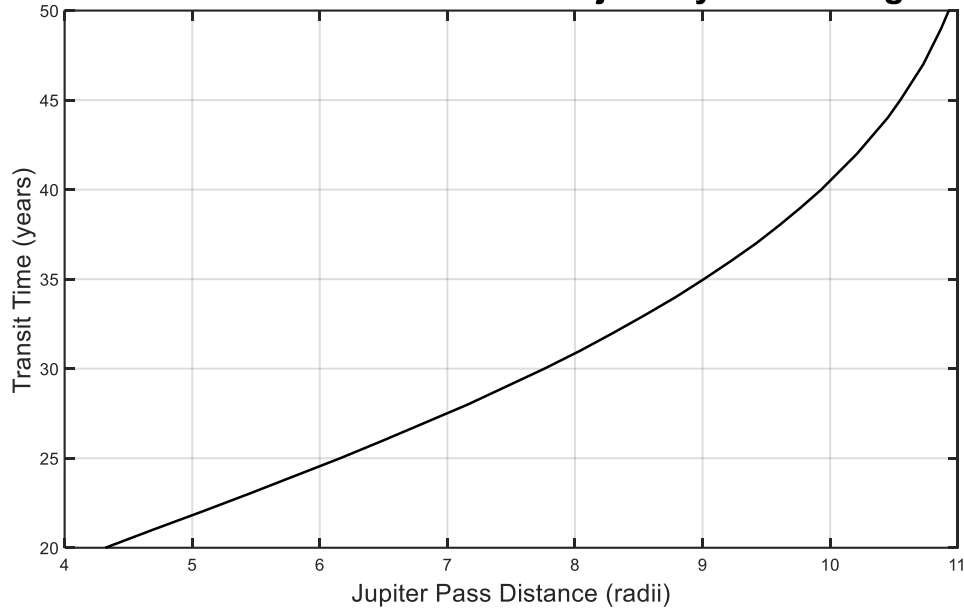


Figure 3.9. Jupiter pass distance vs transit time for an Earth-delta-VEGA-JGA-Sedna trajectory launching in 2046 with a constant payload mass.

Powered Jupiter Gravity Assist

While 1 Jupiter radii is the theoretical minimum, a pass distance of 1.7 Jupiter radii is the assumed minimum due to Pioneer 11's successful mission at that flyby distance [37]. For final mission design, radiation dosage the spacecraft can handle based on its radiation shielding is the true determiner of allowable pass distance. If a shorter transit time is still desired once the minimum pass distance is reached, a powered gravity assist could be of use granted the mission has excess mass capabilities.

Unlike an unpowered gravity assist, a powered gravity assist transfers new energy to the spacecraft by doing a space burn, ideally at the periapsis of the flyby (**Figure 1.2**). This maneuver requires propellant mass, tankage mass, and engine mass to complete. Essentially, payload mass is sacrificed to shorten the transit time. **Figure 3.10** displays this via payload mass curves for unpowered and powered gravity assist trajectories to Sedna. The curves, powered and unpowered gravity assists, are produced from two

different papers using data from Anderson et. al. [14] and Brickley et. al. [15] respectively.

They also have different launch years which are 2045 for powered and 2044 for unpowered. The different launch years simply move Earth and Jupiter along their orbital paths around the sun. However, this greatly affects the trajectories considering how long Sedna period around the sun is. It barely moves over the span of a year so aligning Earth, Jupiter, and Sedna two years in a row is feasible.

Another benefit of a powered gravity assist is the ability to complete a mission quicker than an unpowered gravity assist with the same pass distance. **Figure 3.11** displays the pass distances (left y-axis) for the two missions to Sedna along with the required delta-v (right y-axis) for the powered mission. By introducing a burn, the powered mission can complete the gravity assist at the same pass distance as the unpowered mission and arrive several years sooner.

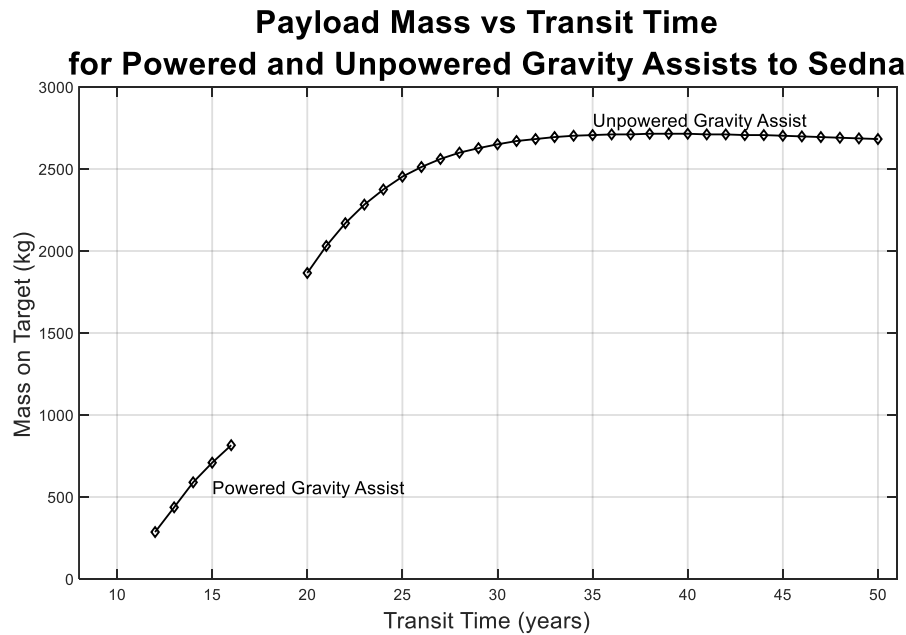


Figure 3.10. Payload mass vs transit time for an unpowered gravity assist mission and powered gravity assist mission to Sedna launching in 2044 and 2045 respectively.

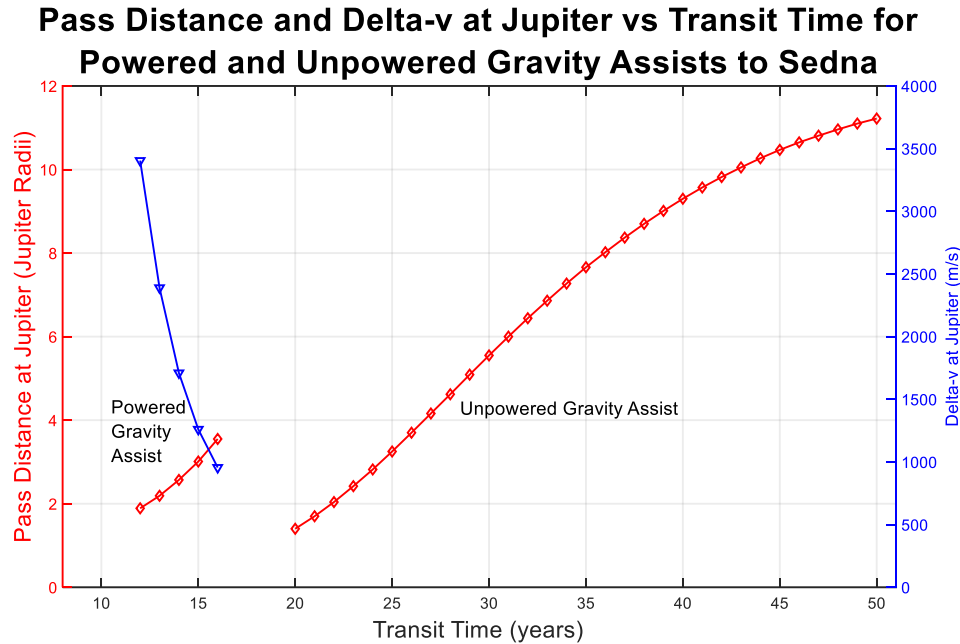


Figure 3.11. Pass distance and delta-v at Jupiter vs transit time for an unpowered gravity assist mission and powered gravity assist mission to Sedna launching in 2044 and 2045 respectively.

Another option is instead of driving down transit times with the burn, the pass distance can be increased. This would allow for an arrival at the same transit time, but with a further pass distance and decreased payload. This could be beneficial for missions with sensitive instruments which cannot be exposed to a large amount of radiation incurred by a close flyby. Or the payload fairing of the launch vehicle may not be large enough to include thick enough radiation shielding for close Jovian encounters.

Jupiter Radiation

Any flyby of Jupiter will experience some level of radiation. Jupiter has a strong radiation zone due to its magnetosphere created by its high rotational speed [49]. The radiation zone is aptly called a radiation belt as it is a “donut-shaped” zone with Jupiter at its center. It also has a “tail” extruding from one side which extends far enough to affect missions (such as Pioneer 10) far after a flyby occurs (**Figure 3.12**). Due to its irregular shape, the amount of radiation experienced during a flyby is dependent on the periapsis

radius and excess speed as well as other factors. See Stewart's Thesis for more details [49].

However, for sets of trajectories with similar mission parameters (e.g., launch year, target, C3), there is a correlation between the radiation dosage and pass distance. **Figure 3.13** displays radiation dosage curves as a function of pass distance for various high and low thrust missions (data from Baskaran et. al. [11]). Similar to excess speed at the target and pass distance at Jupiter, there appears to be no benefit to either high or low thrust architectures with regards to radiation. However, a trend between all trajectories is the further the pass distance, the lower the radiation [11].

The radiation is the most intense closest to Jupiter and dissipates further out causing this trend. The curves begin to flatten at a pass distance of 15 Jupiter radii, so there is little to no benefit beyond that distance. This sets the theoretical maximum pass distance to avoid radiation while maximizing the benefits of a Jupiter flyby.

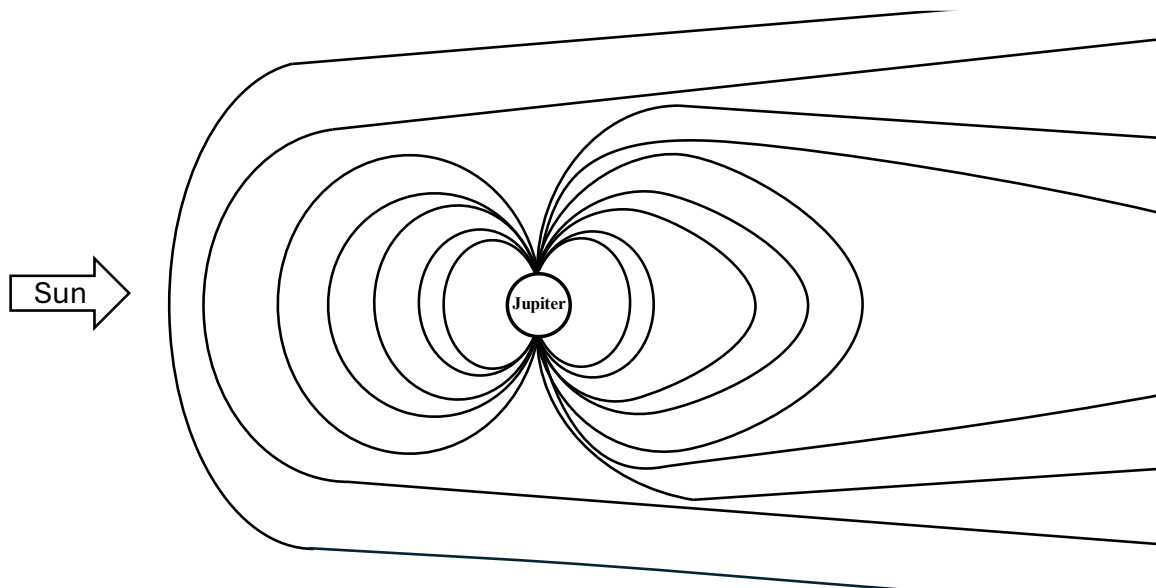


Figure 3.12. 2-D visual representation of Jupiter's magnetosphere.

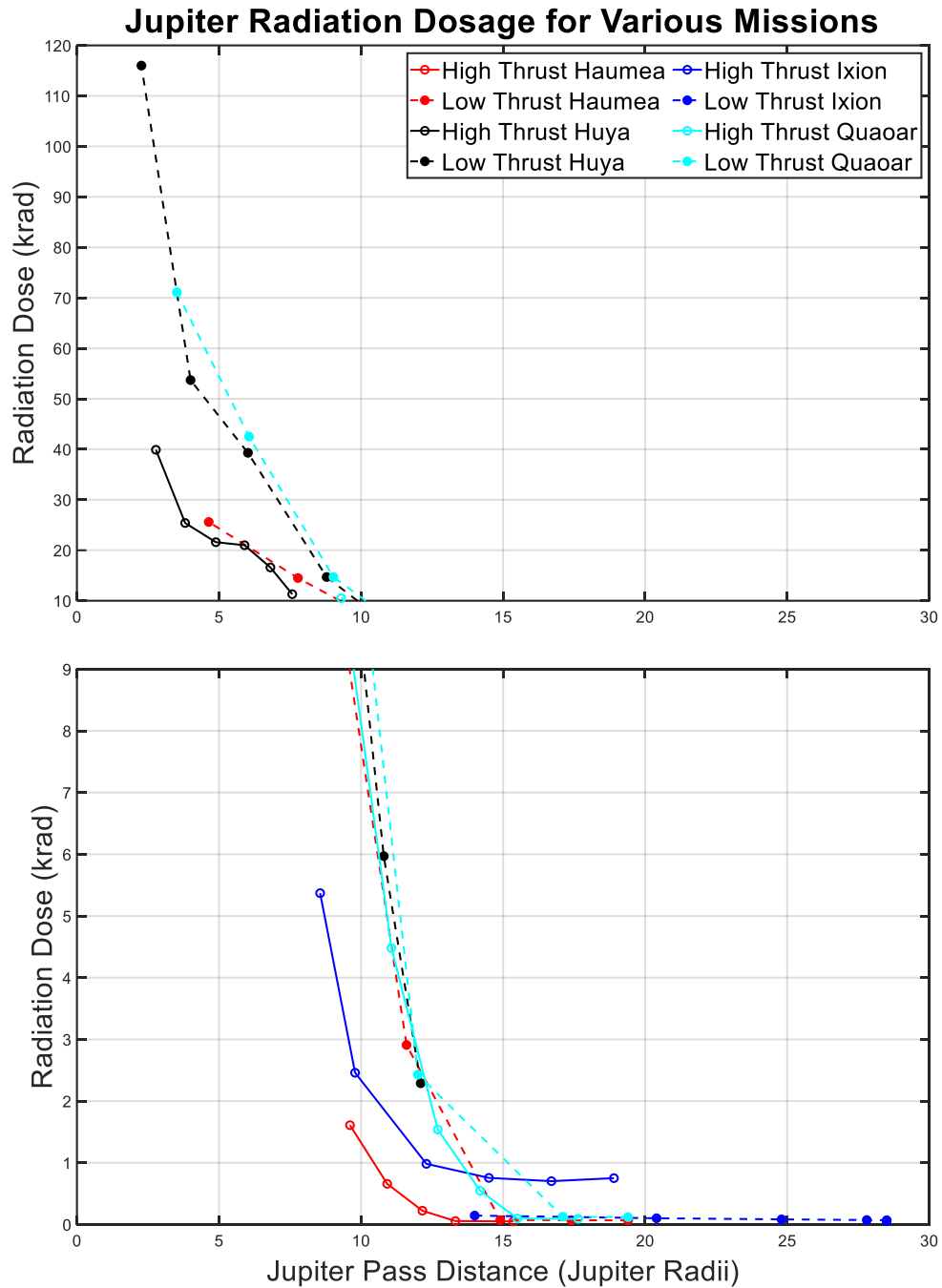


Figure 3.13. Radiation dosage behind 5mm aluminum shielding vs pass distance at Jupiter for unpowered gravity assist missions to various targets. (y-axis scale varies to allow for adequate marker spacing for legibility)

Multi-Target Missions

Buying launch vehicles is an essential but expensive part of an interplanetary mission (**Table 2.2**). To reduce the cost of missions and increase the amount of scientific data collected, multi-target missions have been investigated. A multi-target mission launches either a single probe which intercepts multiple targets or two or more probes each intercepting a different target.

Mission Architectures

There have been 4 multi-target missions investigated by The University of Tennessee [8, 12, 13, 14]. One paper investigates a single probe which encounters multiple targets [13]. While this is the least complicated mission and allows for the most payload mass (or shortest transit time if the excess mass is used for propellant), it is extremely rare to find two targets close enough to one another to send one probe. The mission consists of two flybys, Huya and Quaoar (in that order), with a propulsive maneuver at Huya to course correct to Quaoar. Unfortunately, the launch window to encounter utilizing one probe has expired, and no other missions have been discovered where one probe can encounter two targets.

Another option is to send multiple probes on one launch vehicle which separate to create multiple flight paths to separate targets. The remaining three papers investigated this possibility [8, 12, 14]. These missions have the same trajectory architecture (Earth-Delta-VEGA-Jupiter-Targets). MAnE is used to optimize the trajectories like the previous single target missions. The launch date and subsequent dates until the probes separate (e.g., space burn, Earth re-encounter, and Jupiter encounter) are constrained. By locking these dates in MAnE, the leg parameters between the nodes converge to the same solution (e.g., C3, space burn, pass distances). After the probes separate, they can be treated as two separate trajectories with their own parameters.

Two of the papers split two probes after the Earth re-encounter and before the Jupiter encounter [8, 14]. This allows them to utilize a Jupiter gravity assist to course correct to their respective targets; however the probes encounter Jupiter the same day which limits the turn angle to a specific range (dependent on the pass distance and burn at Jupiter). The final paper allows three probes to split after the deep space burn but before

the Earth re-encounter [12]. The Earth re-encounter date is the same for all three probes but the pass distance differs to allow for different Jupiter encounter dates. This expands the range of available targets (**Figure 3.14**).

Mission Results

Three papers each reported a single mission with set trajectory parameters [8, 12, 13]. The summary of the trajectory parameters is in **Table 3.2**. These papers have allowed MAnE to optimize trajectories using input mission constraints. The fourth paper focused on utilizing a range of powered gravity assists at Jupiter to have a transit time range similar to many of the single target missions.

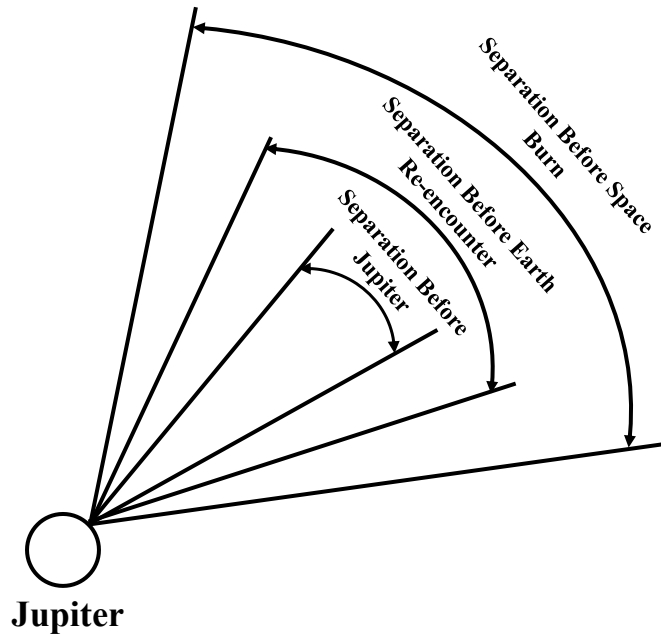


Figure 3.14. Visual representation of effective target range for different probe separation locations for multi-probe missions. (not to scale)

Table 3.2. Set trajectory parameters of three multi-target missions.
(a – Unspecified launch vehicle and payload mass. See source [11] Table 4 for multiple options)

Joint Mission: Huya and Quaoar (One Probe)										
Mission Type	TNO	Launch Date	Mission Objective	Transit Time (years)	Space Burn (m/s)	C3 ($\frac{kn^2}{s^2}$)	Jupiter Pass Distance (radii)	Excess Speed at TNO (km/s)	Required Delta-V at TNO (km/s)	Source
E-Delta-VEGA-JGA-TNO-TNO	Huya	10/15/2025	Flyby	20.7	888.2	33.5	16.59	25.3	1.677	[13]
	Quaoar			27				17.3	-	

(b – Delta IV Heavy launch vehicle and assumed dry probe masses of 380 kg each)

Joint Mission: WW31, UX25, and TC36 (3 Probes)							
Mission Type	TNO	Launch Date	Mission Objective	Transit Time (years)	C3 (km^2/s^2)	Excess Speed at TNO (km/s)	Source
E-Delta-VEGA-JGA-TNOs	1998 WW31	2/22/2030	Flyby	25.3	27.47	25.3	[12]
	2002 UX25		Flyby	17.3		17.3	
	1999 TC36		Orbiter	26.3		26.3	

(c –Atlas V 551 launch vehicle with and a Salacia probe mass of 800 kg and Neptune probe mass of 883 kg.)

Joint Mission: Neptune and Salacia (2 Probes)										
Mission Type	TNO	Launch Date	Mission Objective	Transit Time (years)	Space Burn (m/s)	C3 ($\frac{kn^2}{s^2}$)	Jupiter Pass Distance (radii)	Excess Speed at TNO (km/s)	Powered Gravity Assists Delta-v (m/s)	Source
E-Delta-VEGA-JGA-TNOs	Salacia	2/28/2042	Flyby	17.6	774	28.23	10.29	13.44	Earth: 866 Jupiter: 0	[8]
	Neptune			11.1			5.26	16.05	Earth: 866 Jupiter: 880	

CHAPTER FOUR

RESULTS FROM SENIOR DESIGN RESEARCH

The research in this chapter was completed during myself and my team's senior design project. I researched missions to the TNO Makemake and developed a dual payload optimization technique which was further developed as part of my thesis research.

Dual-Mission to Makemake and Orcus

Mass Correction

In [14] Anderson et al describe a two-probe mission to Makemake and Orcus using a delta VEGA and a powered Jupiter flyby. Unfortunately, an error was made in mass calculations in that the tankage mass for the space burn was neglected. This was remedied and new values are presented below. While the mass calculations were incorrect, the trajectories themselves were all converged and viable on their own.

The corrected initial mass (including the previously neglected tankage mass) was calculated for a range of transit times to Makemake (Mission 1). This was then used in conjunction with performance data for a Falcon Heavy in Expendable Mode to determine the corrected total initial mass available for the Orcus probe (Mission 2). The initial mass of the Orcus probe was then used to determine the allowable delta-v at Jupiter for the Orcus mission. This new, plausible delta-v was used with a previously-developed curve fit of transit time vs Jupiter flyby delta-v to determine the corrected transit time to Orcus. Corrected mission details are given in **Table 4.1**, and plots of the original (incorrect) and corrected transit times are shown in **Figure 4.3**. The corrections resulted in small increases in the combined transit time, on the order of a few months.

Transit Time Calculation

For this dual mission, the mass to arrive at each target is set for 500 kg for each probe which was selected to match the New Horizons probe mass. This eliminates an unknown from mass calculations and allows for a more iterative approach. The process is broken down into three sections. First is the launch and proceeding delta-VEGA maneuvers until Earth re-encounter. The C3 of the mission is taken from MAnE which will be used to calculate the allowable launch mass from the provided launch

performance vehicle curve fits (**Table 2.3**) much like for the single target missions. The space burn is also taken from MAnE to calculate mass after the space burn using

$$m_f = \frac{m_i}{\exp\left(\frac{\Delta v}{g_0 I_{SP}}\right)} \quad (1)$$

Where Δv is the space burn burn, g_0 is Earth's gravitational constant (9.81 m/s^2), I_{SP} is the specific impulse of the rocket used for the space burn (317 s) and m_i is the mass before the burn. The tanks are assumed to be jettisons giving the final usable mission mass as

$$m_{usable} = 1.1m_f - 0.1m_i \quad (5)$$

The two remaining sections are the two separate missions which begin when the probes separate from the main launch capsule after the Earth re-encounter. **Figure 4.1** displays and example ecliptic projection of a dual mission. In this example, the targets are Eris and Borasisi, however the architecture is the same as the Makemake-Orcus mission. The spacecraft splits into two probes after the space burn. Their re-encounter with Earth is identical leading to the same encounter date and pass distance of $1.7 \text{ Jupiter radii}$ at Jupiter. The probes perform powered Jupiter gravity assists which allow them to have different turn angles creating different outbound trajectories to their respective targets.

First Mission

With a set payload mass, only the delta-v at Jupiter is required to calculate the initial mass of the first mission. The Jupiter delta-v can be iterated and input into MAnE which will output the time of flight. Alternatively, a transit time can be set in MAnE which will output the required delta-v at Jupiter. To solve for the mass required to complete the first trajectory (i.e., tankage, propellant, and payload masses), the delta-v at the powered Jupiter gravity assist is iterated from zero until the mission is no longer feasible. To be feasible, there must be at least 500 kg leftover from the first trajectory for the second probe which assumes it completes an unpowered gravity assist.

With the payload mass and delta-v known, equations 1, 2, and 3 can be combined to create

$$m_i = \frac{\exp\left(\frac{\Delta v}{g_0 I_{SP}}\right) m^*}{\left[1.1 - 0.1 \exp\left(\frac{\Delta v}{g_0 I_{SP}}\right)\right]} \quad (10)$$

This provides the initial mass of the first trajectory where Δv is the Jupiter burn, g_0 is Earth's gravitational constant (9.81 m/s^2), I_{SP} is the specific impulse of the rocket used for the Jupiter burn (317 s), m^* is payload mass (500 kg), and m_i is the initial mass of the first trajectory (the fractions are a product of the tankage mass assumed to be 10% of the propellant mass).

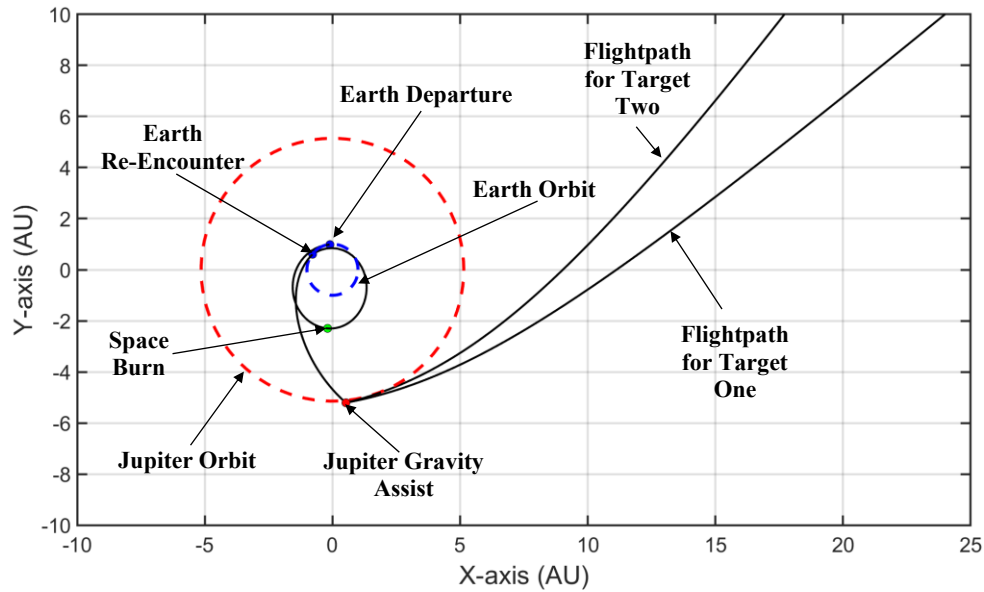


Figure 4.1. Ecliptic projection of a dual mission which the probes separate after the space burn (targets are Eris and Borasisi).

The mass after the Jupiter burn can be calculated by combining equations 2 and 3 to create

$$m_f = \frac{m^* + 0.1m_i}{1.1} \quad (11)$$

where m_f is the mass after the Jupiter burn (and final mass of the mission).

Second Mission

The initial mass of the second probe's trajectory is simply the allowable launch mass minus the initial mass of the first probe's trajectory. The final mass can be calculated using eq. 11. With the initial and final mass known, the delta-v of the burn can be found using the simplified rocket equation

$$\Delta v = g_0 I_{SP} \ln \left(\frac{m_i}{m_f} \right) \quad (12)$$

MANE is used to find the transit time of the second mission by locking the encounter dates before probe separation and inputting the calculated maximum allowable delta-v at Jupiter. This process is repeated for each iteration of the first mission (i.e., all Jupiter delta-v iterations). The mass calculation process is described in a flowchart in **Figure 4.2**.

Transit Time Results

Table 4.1 displays the parameters for both trajectories. Each row is one mission for a total of 12 missions. Each row is linked together as a combination of delta-v's at Jupiter. As the delta-v for the Makemake trajectory increases, the required initial mass increases due to the propellant requirement. This decreases the available mass for the second mission which, in turn, decreases the delta-v. There is essentially a trade-off. As the delta-v for one mission increases, the delta-v for the other decreases.

Dual-Mission Mass Calculation Flowchart

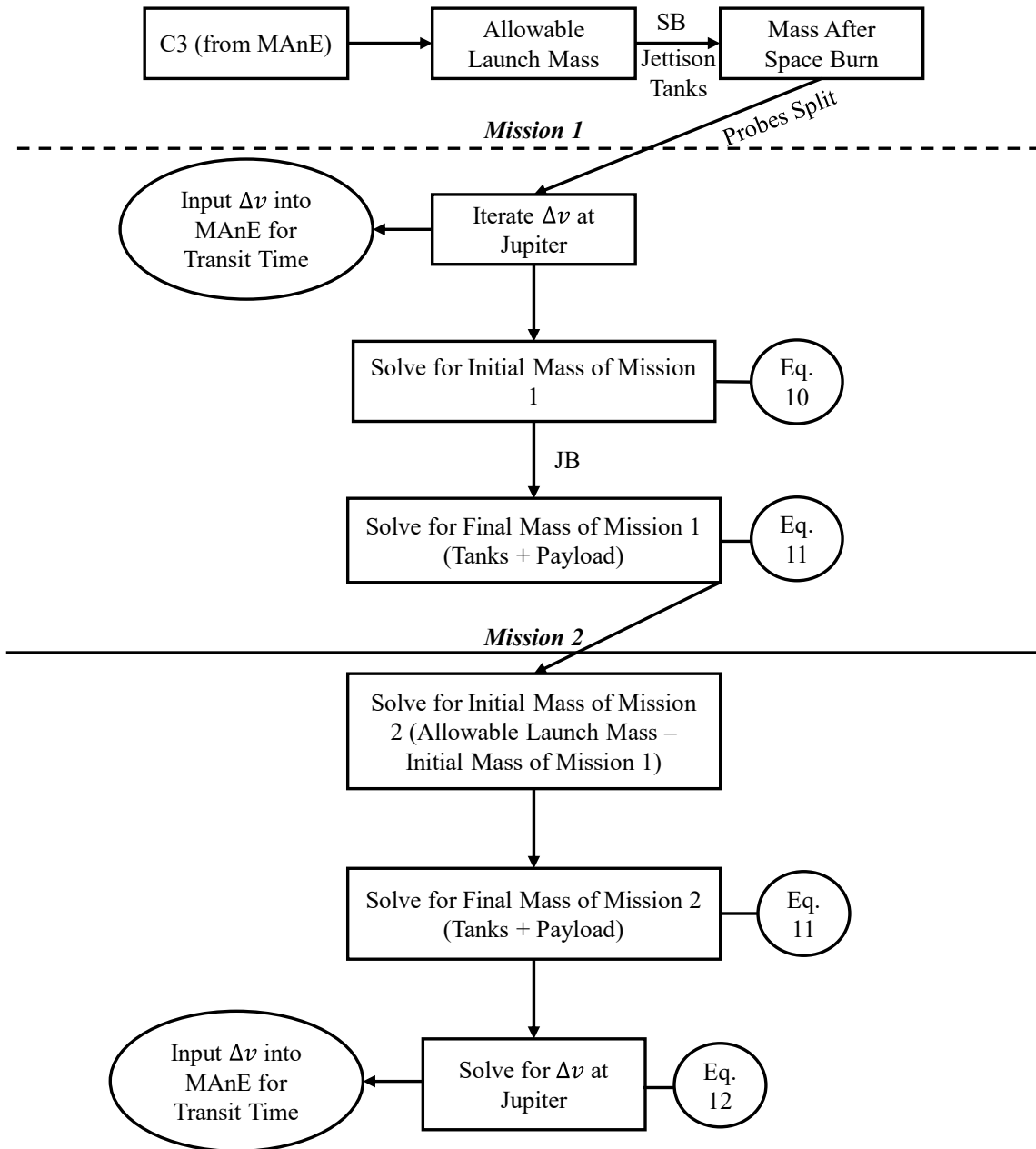


Figure 4.2. Flowchart describing the mass calculation process for a dual mission with powered Jupiter gravity assists.

With the payload mass set, the important parameter is transit time. The transit time for trajectory one (Makemake) is plotted as a function of the second trajectory's (Orcus) transit time (**Figure 4.3**). The tradeoff between the two trajectories can easily be seen. Further work on finding the minimum sum transit time by utilizing curve fits and iterative processes in **Chapter 5: Dual Mission Optimization**.

Trajectories to Makemake

A specific target, Makemake, was chosen to investigate the effects of powered Jupiter gravity assists. Transit time and mass on target were prioritized for optimization. Makemake had optimal launch dates on August 22, 2036, and September 27, 2048, due to the alignment of Earth, Jupiter, and Makemake. A 30-day launch window was created to allow for a month-long range of dates to account for any launch delays. Trajectories are calculated ± 15 days from the optimal launch date in 5-day increments. From these 7 trajectories, the worst attributes are taken from each (i.e., highest C3, excess speed at target, Jupiter delta-v and lowest pass distance at Jupiter). These numbers were used for all payload calculations and figure creation. The C3 and mass on target ranges are shown across the 30-day launch window for each chosen launch date in **Figure 4.4**.

Table 4.1. Dual-mission trajectory parameters for Makemake (left) and Orcus (right).

Makemake			Orcus		
Jupiter Pass Distance (radii)	Delta-v at Jupiter (m/s)	Transit Time (years)	Jupiter Pass Distance (radii)	Delta-v at Jupiter (m/s)	Transit Time (years)
3.51	500	18.62	3.51	5242	10.57
2.56	1000	16.67	2.56	5203	10.60
1.94	1500	15.15	1.94	5153	10.64
1.7	2000	14.17	1.7	5088	10.68
1.7	2500	13.71	1.7	5000	10.75
1.7	3000	13.35	1.7	4876	10.84
1.7	3500	13.04	1.7	4694	10.99
1.7	4000	12.76	1.7	4401	11.23
1.7	4500	12.51	1.7	3859	11.74
1.7	5000	12.27	1.7	2501	13.51

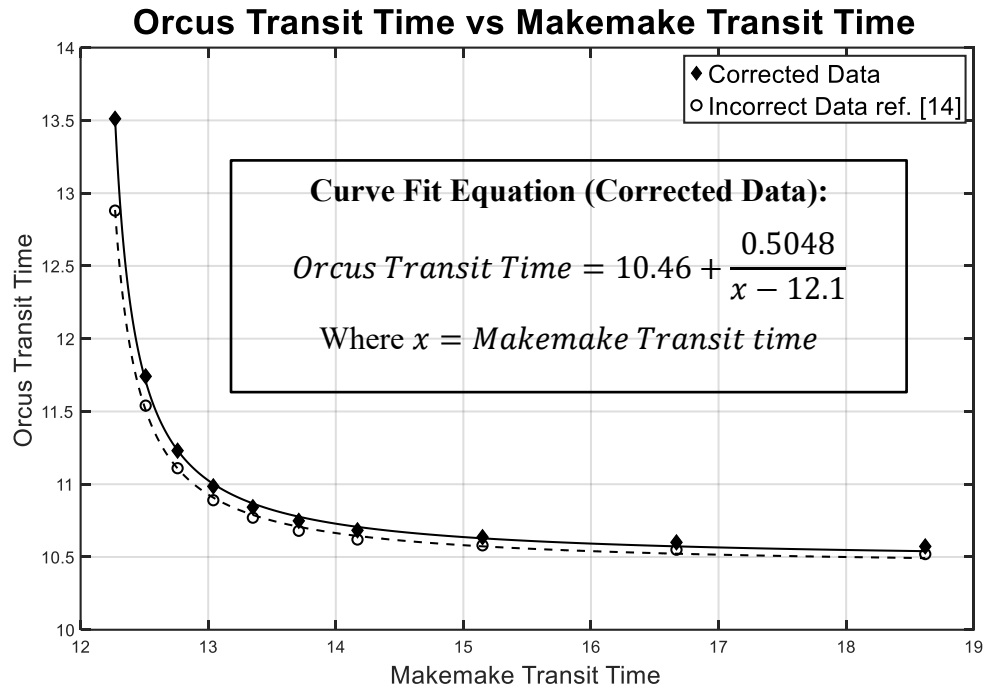


Figure 4.3. Makemake transit time as a function of Orcus transit time for dual missions for two 500 kg probes.

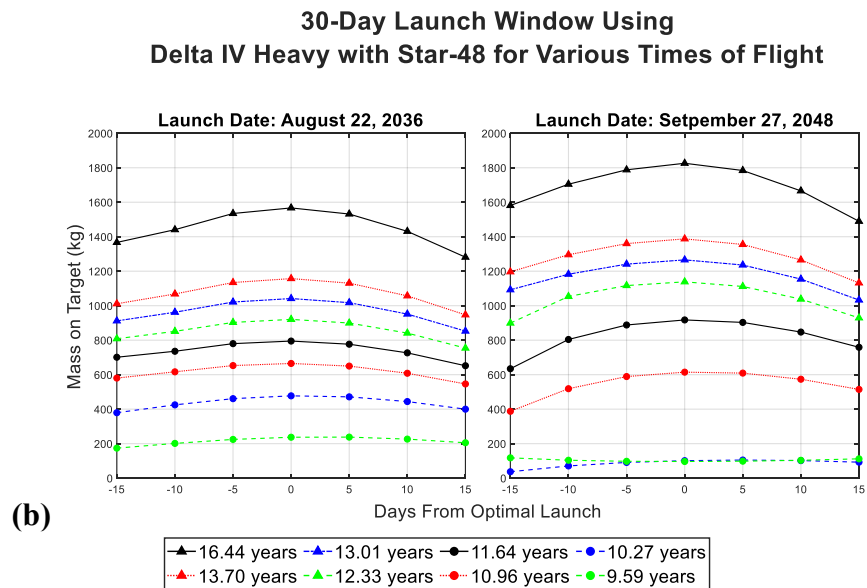
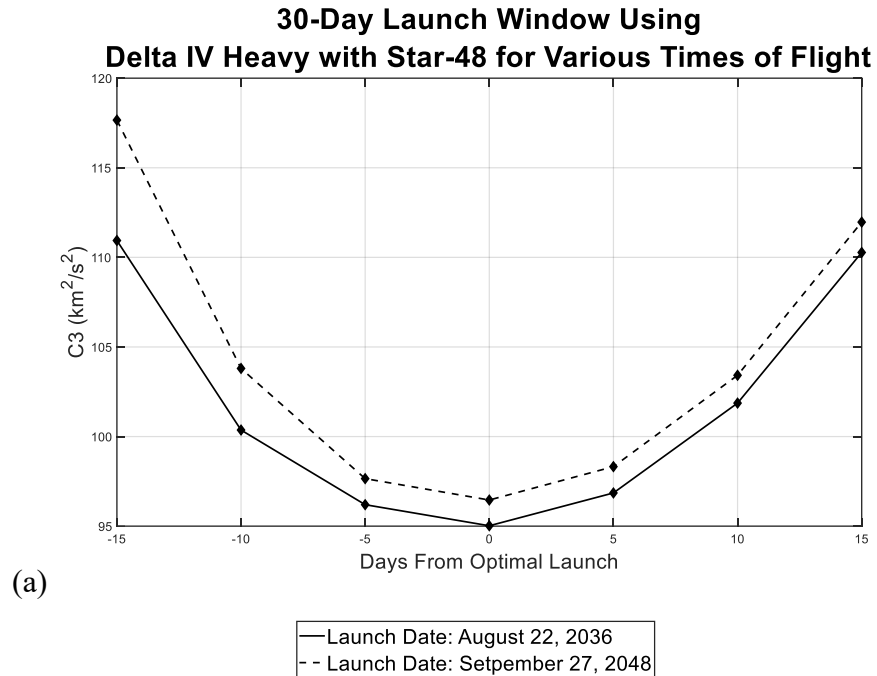


Figure 4.4. Launch windows for 2036 and 2048 of Earth-Powered Jupiter-Makemake missions with ideal launch date from Earth on August 22, 2036 (left) and September 27, 2048 (right) using the Delta IV Heavy rocket with additional Star 48 third stage. The figure displays the C3 and mass on target (kg) as a function of plus or minus 15 days from the optimal departure date for various transit times (years).

The C3 shown in **Figure 4.4** does not vary as a function of transit time, since the launch date and Jupiter encounter date, once optimized, were set constant. The transit time was determined by the Jupiter passage distance and flyby delta-v.

Normally a C3 launch window graph is shown to display the highest C3 as it is the biggest indicator of mass on target for a mission (**Figure 4.4**). However, due to the missions having a powered flyby at Jupiter, the required burn also needs to be considered for mass on target analysis. So, the launch window graph of mass on target vs launch date is shown as well. The mission is run seven times over five-day intervals to find the most undesirable characteristics. This process was done for every transit time. This worst-case data is displayed on the remaining figures in this section.

On these dates, there are feasible mission durations ranging from just under 11 years to over 16 years. As mission duration increases, parameters such as mass on target and Jupiter passage distance will both increase to over 1200 kg and 4.5 Jupiter radii respectively. The opposite is shown to be true for decreasing transit times. As the mission time decreases, the mass on target approaches 0. Analysis ends around 200 kg (2036) and 50 kg (2048). The Jupiter passage distance decreases to 1.7 which is the mission minimum.

Graphing the data allows for a quick comparison of launch years and how changing a trajectory variable affects the rest of the mission. In **Figure 4.5** the mass on target and Jupiter pass distance are plotted against transit time. This mass represents the lowest possible mass on target within the 30-day launch window. As transit time decreases, mass on target and Jupiter pass distance also decrease. The pass distance at Jupiter decreases to increase how effective the flyby is. The closer the pass distance at Jupiter, the more energy Jupiter adds to the flyby maneuver. The mass on target decreases due to the burn at Jupiter increasing (**Figure 4.6**). More of the allowable launch mass from Earth, calculated with the curve fit is used for fuel to burn at Jupiter, decreasing the mass able to be used for the payload.

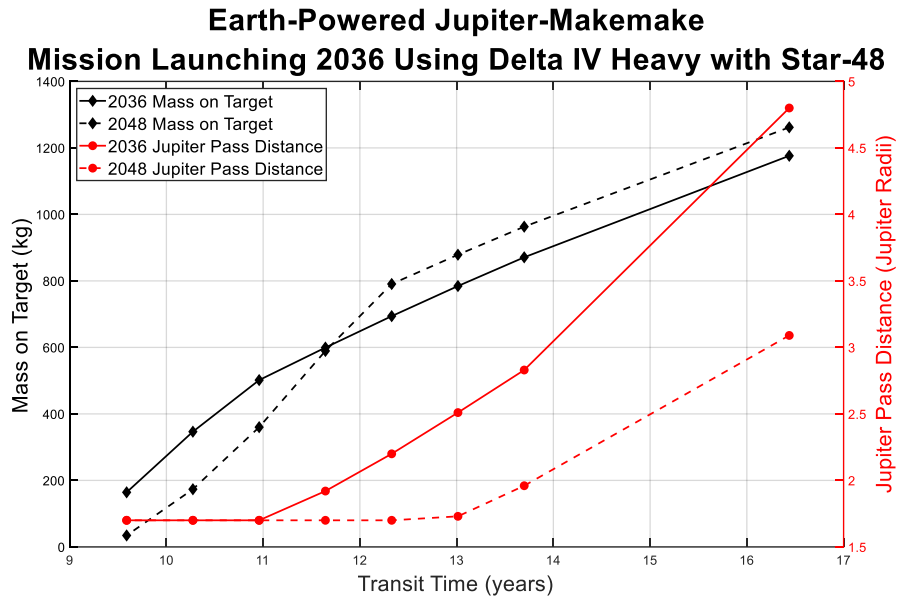


Figure 4.5. Mass on target and Jupiter pass distance as a function of the transit time to reach Makemake.

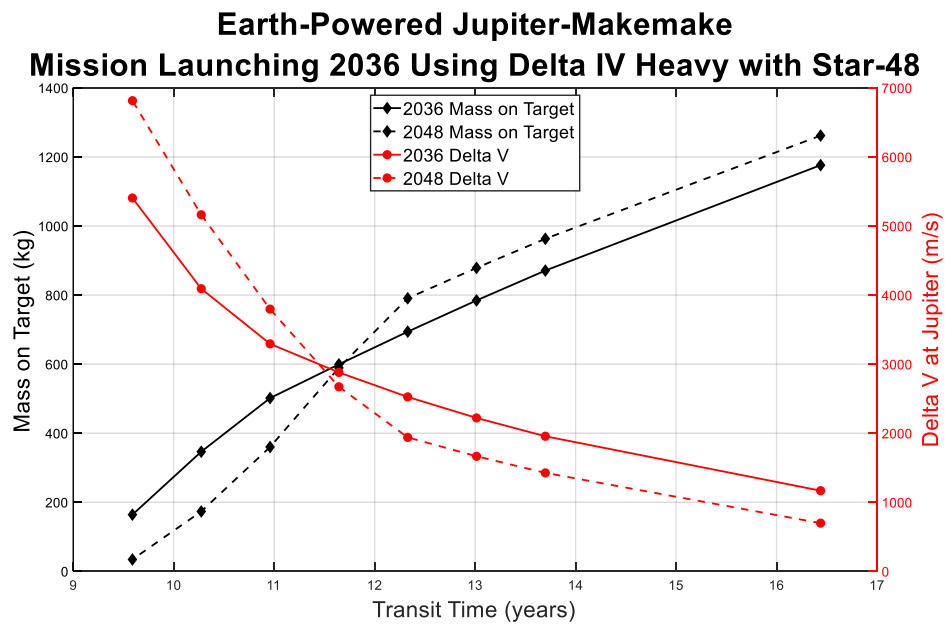


Figure 4.6. Mass on target and Delta V at Jupiter as a function of the transit time to Makemake.

As the transit time of the mission decreases, the excess speed at the target increases (**Figure 4.7**). This is due to the powered flyby of Jupiter adding more and more energy to the trajectory causing the spacecraft to speed up. The added energy is due to the burn increasing and pass distance decreasing at Jupiter when transit times decrease. While this allows the spacecraft to reach the target quicker, it also causes high speeds once the spacecraft reaches its destination. High arrival excess speeds decrease the available observation time and will impact requirements on the remote sensing instrument suite. New Horizons had a suitable excess speed at Pluto of almost 14 km/s [50].

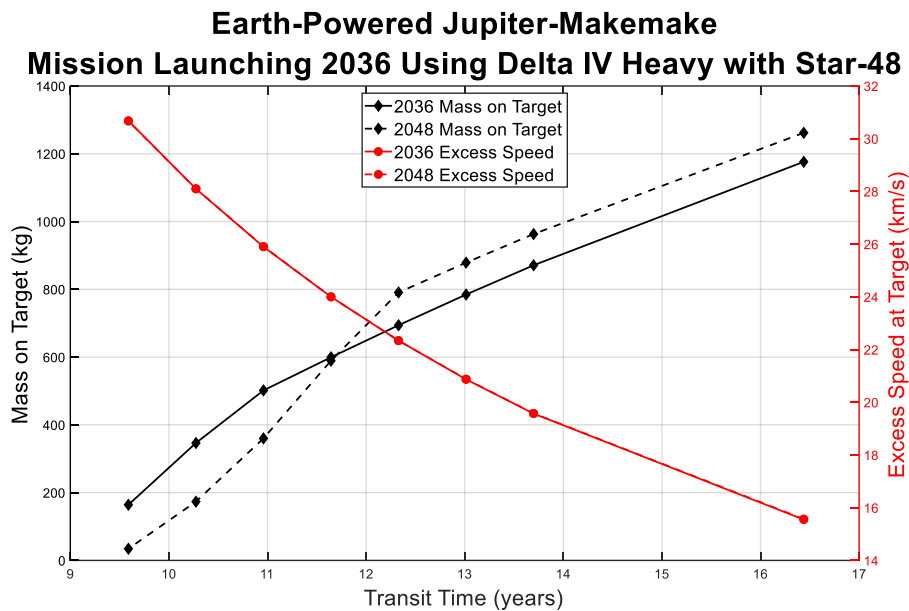


Figure 4.7. Mass on target and Excess Speed at target as a function of transit time to Makemake.

CHAPTER FIVE

RESULTS OF TRAJECTORY OPTIMIZATION

Multiple topics for further research were discovered during review and summation of previous work. These topics along with further development of senior design research were researched in this Chapter.

Dual-Mission Optimization

A dual mission can be further analyzed to determine the minimum sum transit time. That is, the minimum total transit time when adding the mission length of both targets. This mission length encompasses the entire mission of each trajectory including before they separate (i.e., from Earth launch to target encounter). This optimization is possible because the missions are linked together by their delta-v at Jupiter. As one mission's Jupiter delta-v increases (meaning propellant mass increases), the other's decreases due to shared allowable launch mass. Therefore, as one mission's transit time decreases (i.e., the Jupiter delta-v increases), the other's transit time decreases.

Makemake and Orcus

To find the minimum sum transit time, a curve fit is used. The transit time of one target is plotted against the transit time of the other target. A curve fit is created which provides a function to solve for the transit time of the y-axis target using transit time values from the x-axis target. The target on the x-axis has an evenly spaced transit time range created from its minimum to maximum values. While these transit times were not calculated with MAnE, all are possible as they fall within the calculated range. These values are run through the curve fit to provide corresponding transit times for the y-axis target. In this case, Orcus is the y-axis target and Makemake is the x-axis target. Each point can be summed together to provide values for the sum transit time for each linked delta-v. The minimum value of this array is the minimum sum transit time.

The individual trajectory parameters for each mission can also be found. Plotting the sum transit time against the transit time range of the x-axis target creates a "check" curve. The check curve and transit time curve can be seen in **Figure 5.1** and **Figure 5.2** respectively. The x-location of the minimum sum transit time is the idea transit time for Makemake (x-axis target).

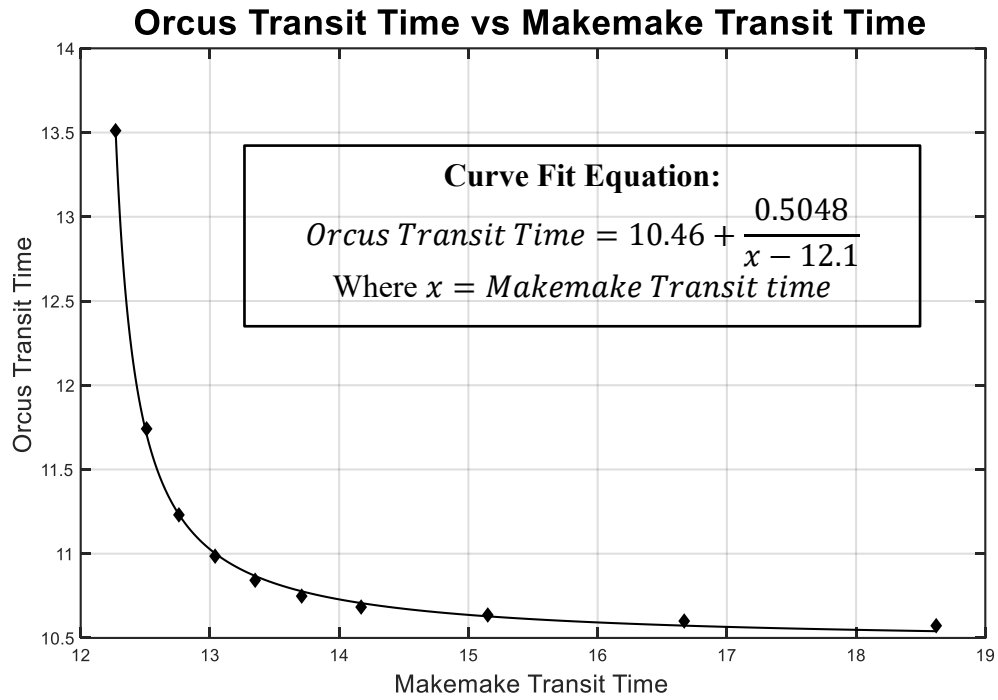


Figure 5.1. Orcus vs Makemake transit time data and curve fit equation used for optimizing sum transit times.

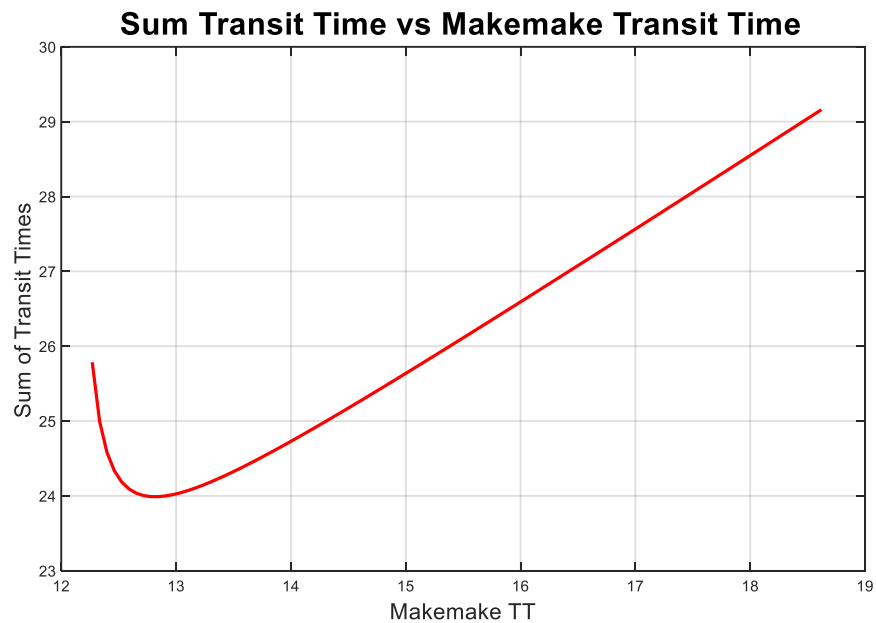


Figure 5.2. Sum transit time vs Makemake transit time.

Inserting this value into the first curve fit provides the ideal transit time for Orcus (y-axis target). Plotting the other important trajectory parameters (i.e., delta-v at Jupiter and pass distance) against their respective targets' transit times, curve fitting them, and inserting the ideal transit times will provide the ideal trajectory parameters for both missions. An example of this process is displayed in **Figure 5.3**. The values are tabulated in **Table 5.1**. This mission uses the Falcon Heavy Expendable launch vehicle, an ISP of 315, and a tankage fraction of 10%.

Eris and Borasisi

The process is repeated for a dual mission to Eris and Borasisi. Eris is a scientifically interesting target. However, it has a highly elliptical orbit and is currently located near its apoapsis of 90 AU. To make a lengthy mission more appealing, a secondary target, Borasisi, is encountered by a separate probe around 35 AU.

Trajectory Architecture

The spacecraft splits into two probes after the launch from Earth and before the space burn. They then each have different space burn magnitudes allowing different Earth re-encounters. The probes perform powered separate Jupiter gravity assists on different dates and with different magnitudes. This allows them to have different turn angles creating different outbound trajectories to their respective targets. An example ecliptic projection of the mission can be seen in **Figure 5.4**.

While the previous Makemake-Orcus mission's probes split after the space burn, the Eris-Borasisi probes split before the space burn. This causes a different mass calculation process. The allowable launch mass is still calculated using the C3 and high energy performance curve fit, however the mass after the space burn must be calculated separately for each trajectory. For the first mission (Eris), the mass before the Jupiter burn is calculated using eq. 10 as the delta-v and payload mass are both known. This will be the required mass to complete the mission excluding the space burn. Using the mass before Jupiter burn (eq. 10) as m_f and Δv as the space burn, the mass before the space burn (initial Eris mission mass), can be found with

$$m_i = \exp\left(\frac{\Delta v}{g_0 ISP}\right) m_f \quad (13)$$

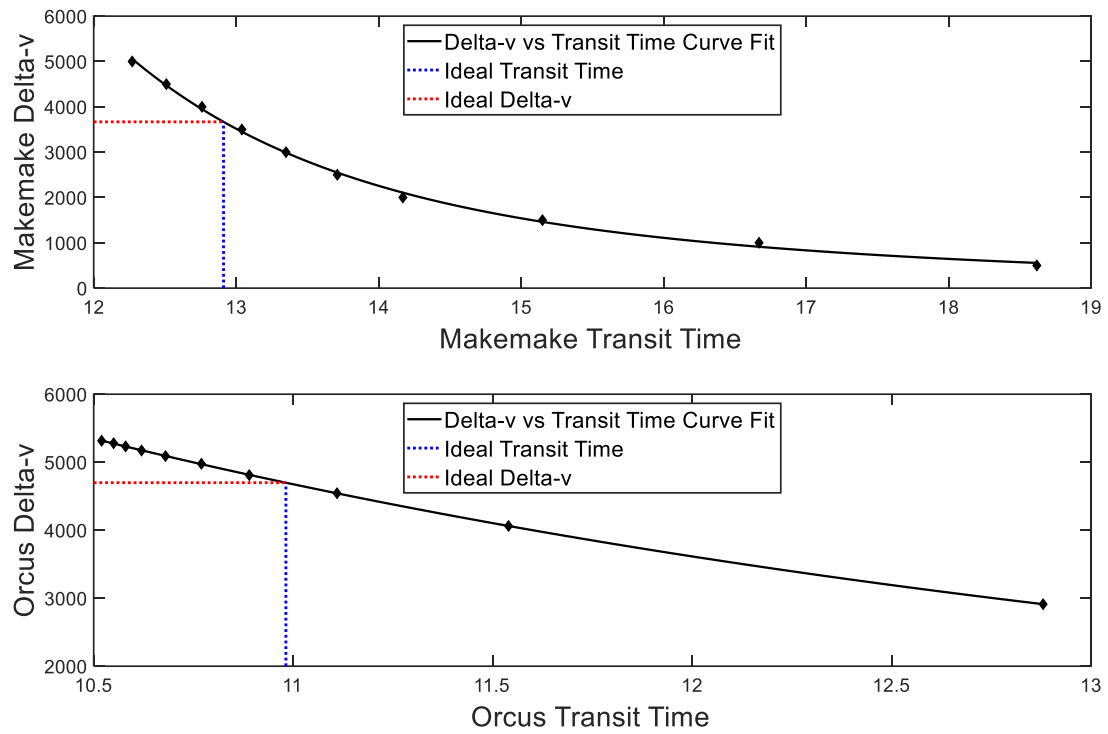


Figure 5.3. Makemake (top) and Orcus (bottom) delta-v vs transit time.

Table 5.1. Optimized mission parameters for dual mission to Makemake and Orcus.

Assumptions		
ISP	317 s	
Tankage Fraction	10 %	
Launch Vehicle	Falcon Heavy Expendable	
Payload Mass	500 kg (Each Probe)	
Initial Mission (Before Probes Split)		
Mission Parameters	Values	
Space Burn Delta-v	1055 m/s	
C3	30.9 kg ² /s ²	
Earth Departure Mass	8090 kg	
Mass After Space Burn	5530 kg	
Ideal Sum Transit Time	24.0 years	
Individual Missions (After Probes Split)		
Ideal Parameters	Ideal Values: Makemake	Ideal Values: Orcus
Transit Time	12.8 years	11.1 years
Delta-v at Jupiter	3784 m/s	4597 m/s
Pass Distance at Jupiter	1.7 Jupiter Radii	1.7 Jupiter Radii
Initial Mass	2213 kg	3316 kg
Mass After Jupiter Burn	656 kg	756 kg
Tankage Mass	156 kg	256 kg
Propellant Mass	1558 kg	2560 kg

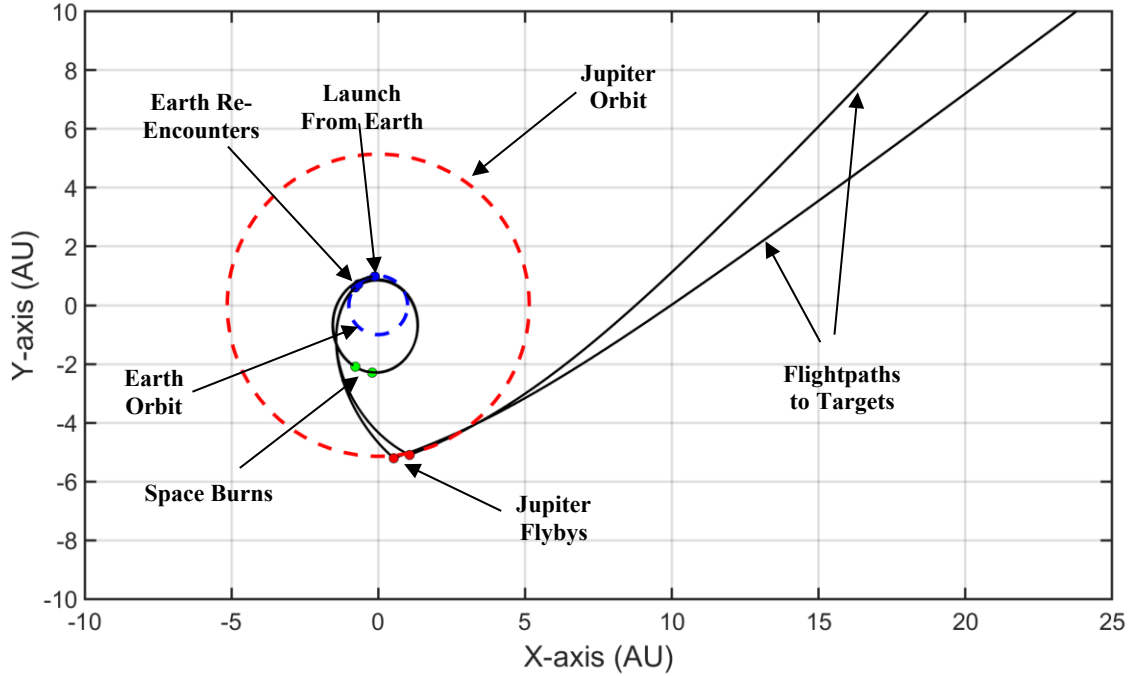


Figure 5.4. Ecliptic projection of a dual mission which the probes separate Before the space burn.

The second mission (Borasisi) calculates the initial mass as previously described. The mass after the space burn is calculated with eq. 1 where Δv is the space burn and m_i is the initial mass of the mission. From there, the process is the same as described for Makemake and Orcus. The mission parameters are shown in **Table 5.2**.

Mission Results

The transit time curve fit is shown in **Figure 5.5**, and the sum transit time is shown in **Figure 5.6**. This mission uses the Falcon Heavy Expendable launch vehicle, an ISP of 317, and a tankage fraction of 10%. The space burn and Jupiter burn dates were allowed to vary throughout the MAnE optimization and therefore do not have specified dates. The probe mass is set to 500 kg for both probes.

Table 5.2. Optimized mission parameters for dual-target mission to Eris and Borasisi.

Assumptions		
ISP	317 <i>s</i>	
Tankage Fraction	10 %	
Launch Vehicle	Falcon Heavy Expendable	
Payload Mass	500 <i>kg</i> (Each Probe)	
Initial Mission (Before Probes Split)		
<i>Mission Parameters</i>	<i>Values</i>	
Launch Date	December 17, 2039	
C3	28 <i>kg</i> ² / <i>s</i> ²	
Earth Departure Mass	8610 <i>kg</i>	
Ideal Sum Transit Time	38.6 <i>years</i>	
Individual Missions (After Probes Split)		
<i>Ideal Parameters</i>	<i>Ideal Values: Eris</i>	<i>Ideal Values: Borasisi</i>
Transit Time	23.7 <i>years</i>	14.9 <i>years</i>
Space Burn Delta-v	800 m/s	625 m/s
Delta-v at Jupiter	4852 m/s	4140 m/s
Pass Distance at Jupiter	1.7 <i>Jupiter radii</i>	1.7 <i>Jupiter radii</i>
Space Burn Date	Allowed to Vary	Allowed to Vary
Jupiter Flyby Date	Allowed to Vary	Allowed to Vary
Initial Mass	4934 kg	3676 kg
Mass After Space Burn	3814 kg	3007 kg
Mass After Jupiter Burn	903 kg	789 kg
Tankage Mass	403 kg	289 kg
Propellant Mass	4852 kg	2887 kg

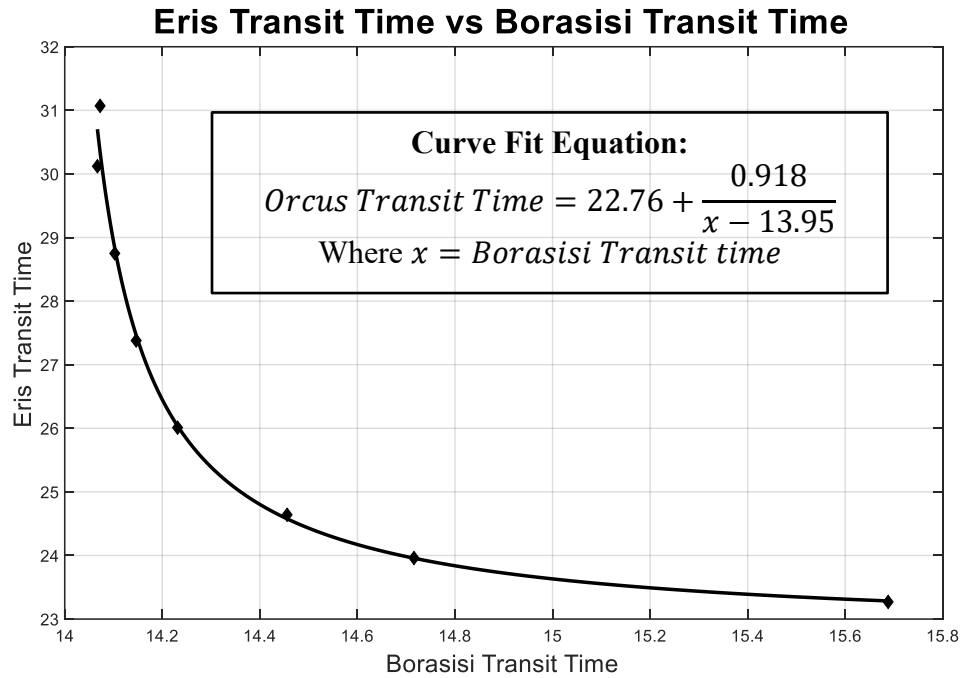


Figure 5.5. Eris transit time vs Borasisi transit time curve fit.

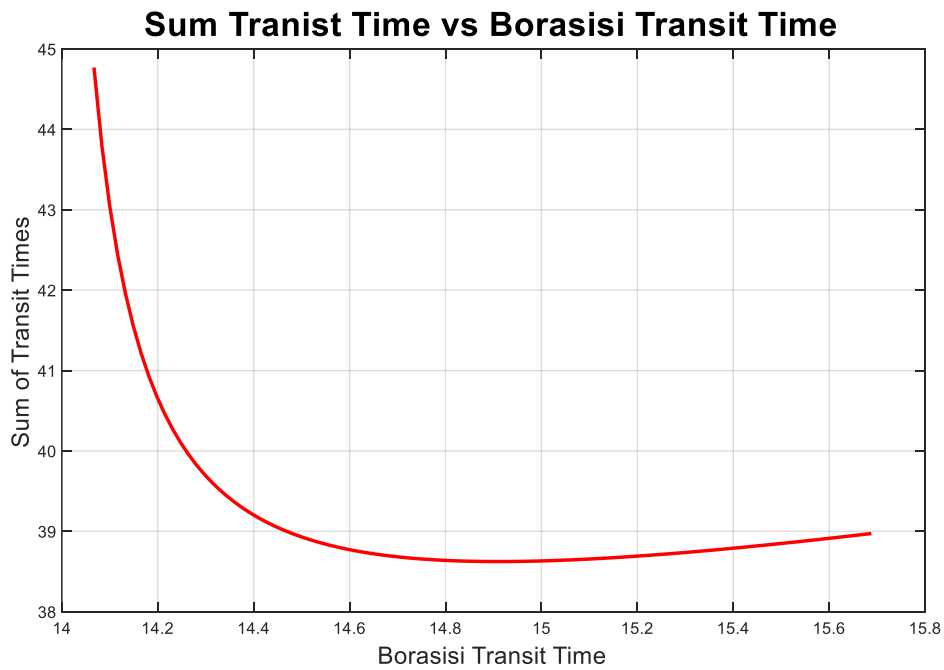


Figure 5.6. Sum transit time vs Borasisi Transit Time

Two-Stage Capture Burn Optimization

Orbiters and landers require large burns to capture into orbit/land at targets. This is due to the desire to shorten transit times which increase excess speeds at the target. And in the case of orbiters, the relatively small mass of TNOs requires a burn essentially equal to the excess speed to orbit. To increase the usable payload mass at a target, a two-stage burn can be implemented. The first stage engine conducts a burn then jettisons the leftover engine and tankage mass. The second burn is then conducted to complete the remaining velocity change to capture into orbit or land. The second stage must slow down less mass which requires less propellant.

Optimization

The amount of propellant needed to complete a burn depends on the required delta-v and initial mass before the burn. These variables also affect the ratio of the first stage to second stage propellant masses. The assumptions used for the optimization are Earth's gravitational constant is $g_0 = 9.18 \text{ m/s}^2$, first stage specific impulse is $ISP_1 = 290$ (solid stage), second stage specific impulse is $ISP_2 = 317$ (liquid stage), and the tankage fraction for both stages is 10%.

The optimization code iterates through a process to solve for the mass after the two-stage burn for a given initial mass and required delta-v. The iterations begin with the first stage propellant mass ($m_{p,1}$) equal to the initial mass

$$m_{p,1} = m_{i,1} \quad (14)$$

and ends with propellant mass equal to zero

$$m_{p,1} = 0 \quad (15)$$

The delta-v of the first stage (Δv_1) is calculated using the ideal rocket equation (eq. 7) where

$$m_{f,1} = m_{i,1} - m_{p,1}. \quad (16)$$

The mass of the first stage propellant mass and 10% tankage fraction (i.e., $1.1 \cdot m_{p,1}$) is subtracted from the initial mass before the burn to find the initial mass of the second burn ($m_{i,2}$).

$$m_{i,2} = m_{i,1} - 1.1 \cdot m_{p,1} \quad (17)$$

The mass after the burn ($m_{f,2}$) is calculated using the rearranged ideal rocket equation (eq. 1)

$$m_{f,2} = \frac{m_{i,2}}{\exp\left(\frac{\Delta v_2}{g_0 ISP_2}\right)} \quad (18)$$

where

$$\Delta v^{(2)} = \Delta v_{total} - \Delta v^1 \quad (19)$$

The payload mass is

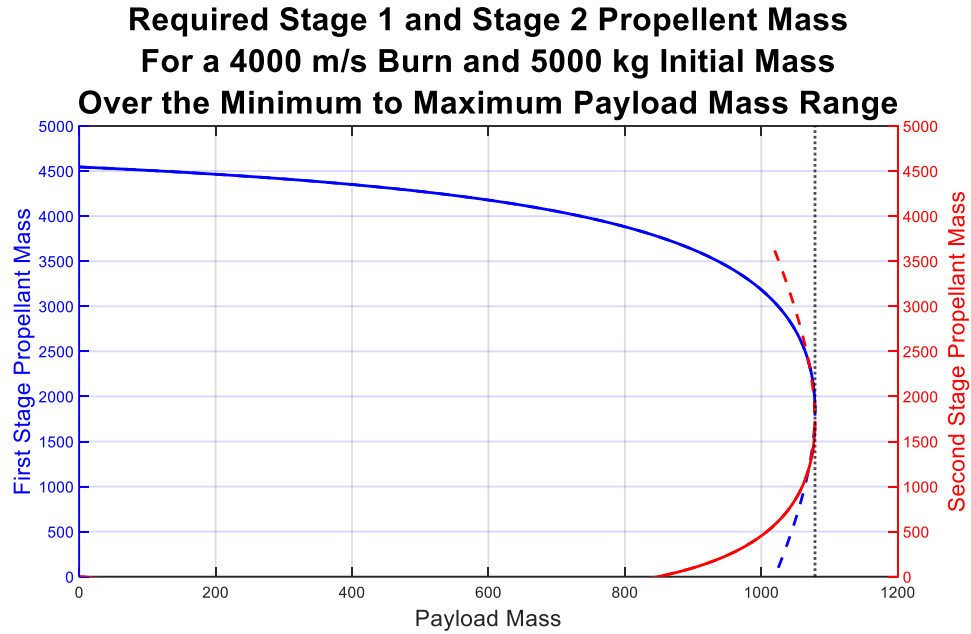
$$m^* = m_f^{(2)} - 0.1m_p^{(2)} \quad (20)$$

Where

$$m_p^{(2)} = m_f^1 - m_f^{(2)} \quad (21)$$

This process is repeated over the desired first stage propellant mass range, and the iteration with the maximum payload mass contains the optimum first and second stage propellant mass values for these specific values of ISP and tankage fraction. **Figure 5.7** displays three examples of a range of capture mass values with the maximum payload mass at the peak of the curves. The blue curve represents the first stage propellant mass and the red curve the second stage propellant mass. The curves have two possible combinations for each payload mass which meet at the peak (black dotted line) where the maximum payload mass and optimum propellant masses lie. The solid lines correspond to one set of combinations and the dotted lines correspond to the other set of combinations.

(a)



(b)

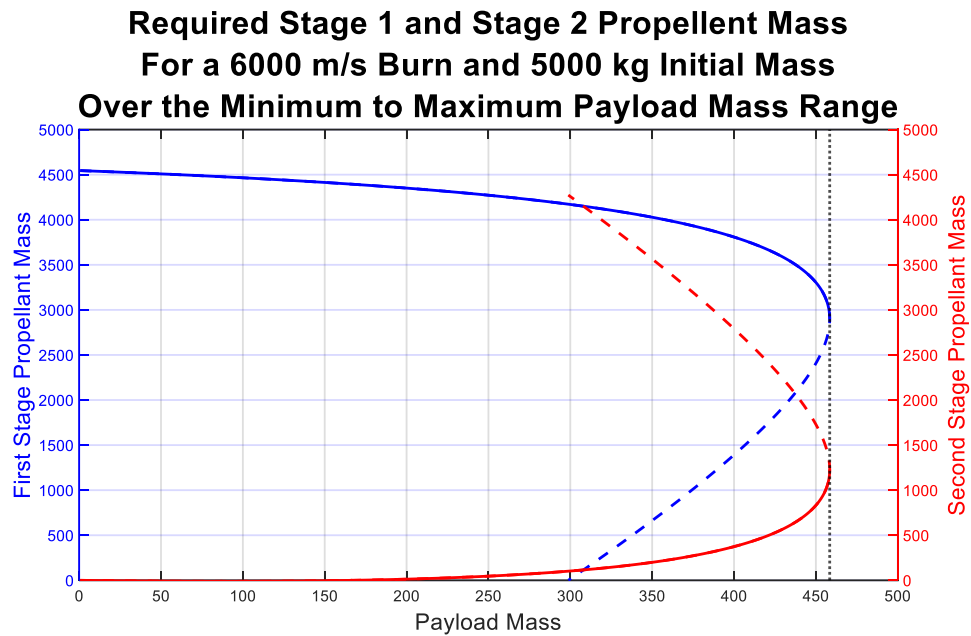


Figure 5.7. Payload mass for a range of first and second stage propellant mass curves for a 4000 (a), 6000 (b), and 8000 (c) m/s burn with an initial mass of 5000 kg. Assumptions: $ISP_1 = 290$ $ISP_2 = 317$ Tankage Fraction = 10%

(c)

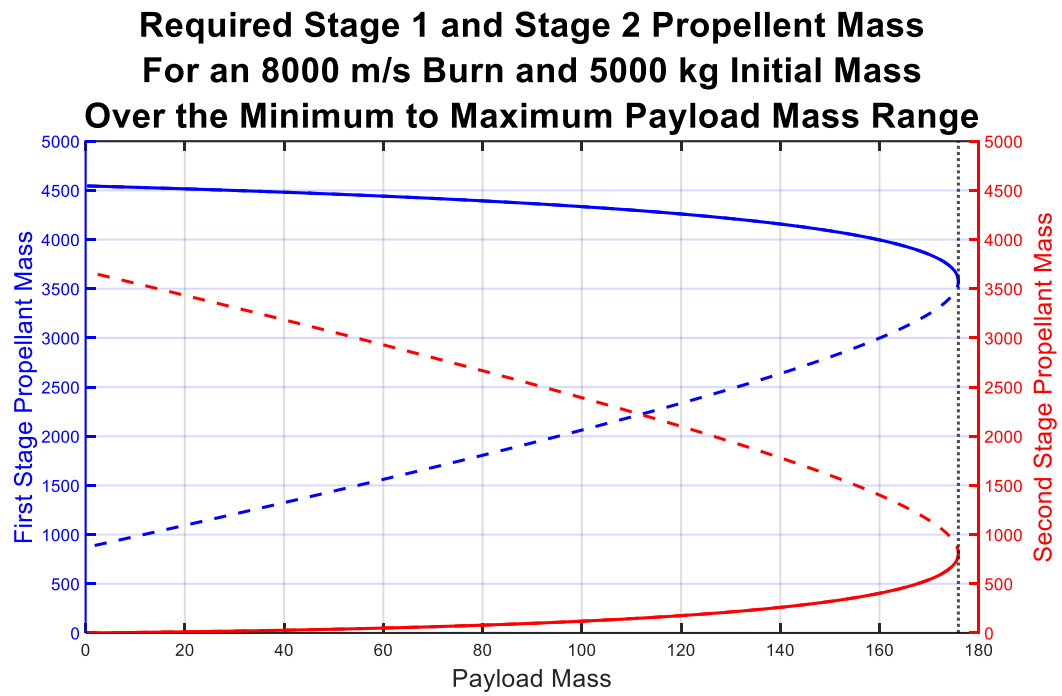


Figure 5.7 conintued.

The effects of initial mass and required delta-v can be seen in **Figure 5.8**. As the delta-v required increases, the mass on target decreases. An increase in initial mass before the burn will shift the payload curves upward. However, the benefit of a larger initial mass diminishes as the delta-v requirement grows larger.

The total required propellant is also affected by both the required delta-v and initial mass. A larger delta-v or larger initial mass will require more propellant mass. However, the ratio of *Stage 1/Stage 2* propellant mass vs required delta-v curve will remain constant if the initial mass before the burn changes (assuming the same ISP and tankage fraction values are used). This is displayed in **Figure 5.9**. The ratio curve remains 0 until the required delta-v exceeds 2000 m/s. This indicates that a two-stage burn is not required, and a single burn can be used for optimal payload mass. The delta-v threshold will vary based on the ISPs and tankage fraction values used.

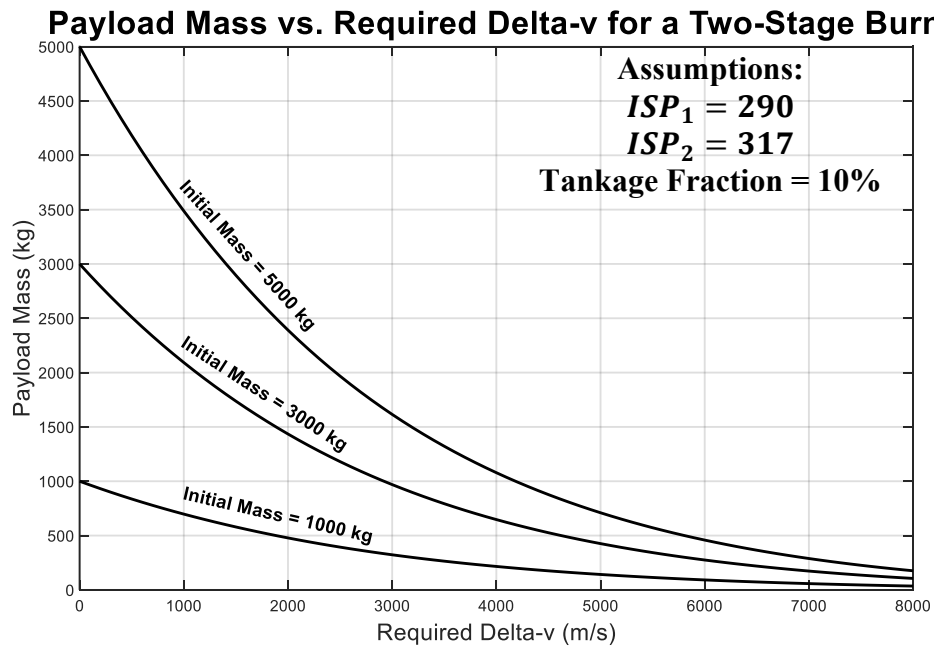


Figure 5.8. Ideal payload mass as a function of the required delta-v where each curve is a different initial mass before the burn.

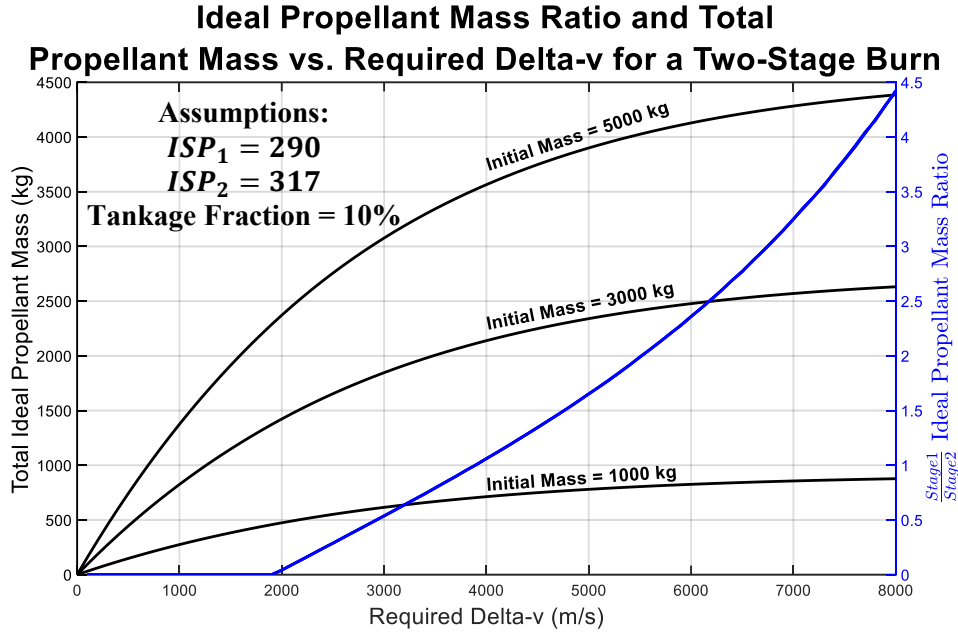


Figure 5.9. Total ideal propellant mass and ideal propellant mass ratio vs required delta-v for multiple initial mass curves.

Mass Fraction

A non-dimensional form of mass can be used to evaluate missions in a more general way. This will be defined as the “Mass Fraction” which takes the general form of equation 16. Mass fraction can be used to generalize entire missions or focus on the effects of certain propulsive maneuvers.

$$m_{Fraction} = \frac{Final\ Mass}{Initial\ Mass} \quad (22)$$

Payload Mass

To determine an arrival payload mass at a target, the allowable launch mass from Earth is needed. This requires a launch vehicle and characteristic energy (C3) as outlined in **Chapter 2**. Alternatively, the non-dimensional form can be applied. The calculation is identical to that described in **Chapter 2** with one exception; the allowable outbound mass from Earth is set to 1 instead of being a mass derived from the launch vehicle performance curve fits (**Figure 2.1**). Therefore, the final mass of the mission will be a percentage of remaining allowable launch mass in decimal form ranging from 0 and 1

(i.e., mass fraction). In this case, initial mass is the allowable launch mass and final mass is the payload mass.

It is important to note that this approach eliminates the dependence of the analysis of any specific launch vehicle and allows a rapid determination of the final payload mass based on the mission C3 and any selected launch vehicle performance curve. This approach allows for previous missions to be reevaluated for new launch vehicles without the need to recalculate every step to derive the payload mass. The mass fraction for each mission from **Chapter 3** has also been tabulated in the appendix.

Maneuver Mass Fraction

Mass fraction can also be used to evaluate different propulsive maneuvers in a more general way. Single stage burns (used for deep space burns and powered gravity assists) and two stage burns (used for capture burns for orbiters and landers) can utilize mass fraction where initial mass is mass before the burn(s) and final mass is mass after the burn(s). To convert to the non-dimensional form, take the mass before the burn to be 1 for calculations. The final Mass fraction will be the percentage of mass before the burn leftover. Multiplying this Mass fraction (or entire curve fit) by a known initial mass before a burn will provide insight into the mass after the burn. This tool provides knowledge on feasibility and limitations to aid in mission design. The mass fraction for single stage burns and optimized two-stage (**Chapter 5: Two Stage Capture Burn Optimization**) burns for certain assumptions (listed in figure legend) are displayed in **Figure 5.10**.

Pass Distance vs Delta-v

The effectiveness of powered gravity assists is determined by how much the maneuver decreases the transit time and/or overall mission propellant requirements. There is a tradeoff between transit time, Earth departure C3, flyby and deep space delta-v maneuvers. The effectiveness of a powered gravity assist is heavily influenced by the pass distance and delta-v. The pass distance is the closest approach made by the spacecraft during the flyby of a planet (known as the periapsis). The delta-v is the change in velocity produced by burning propellant. A more in-depth explanation is detailed in **Chapter 1: Trajectory Architecture**.

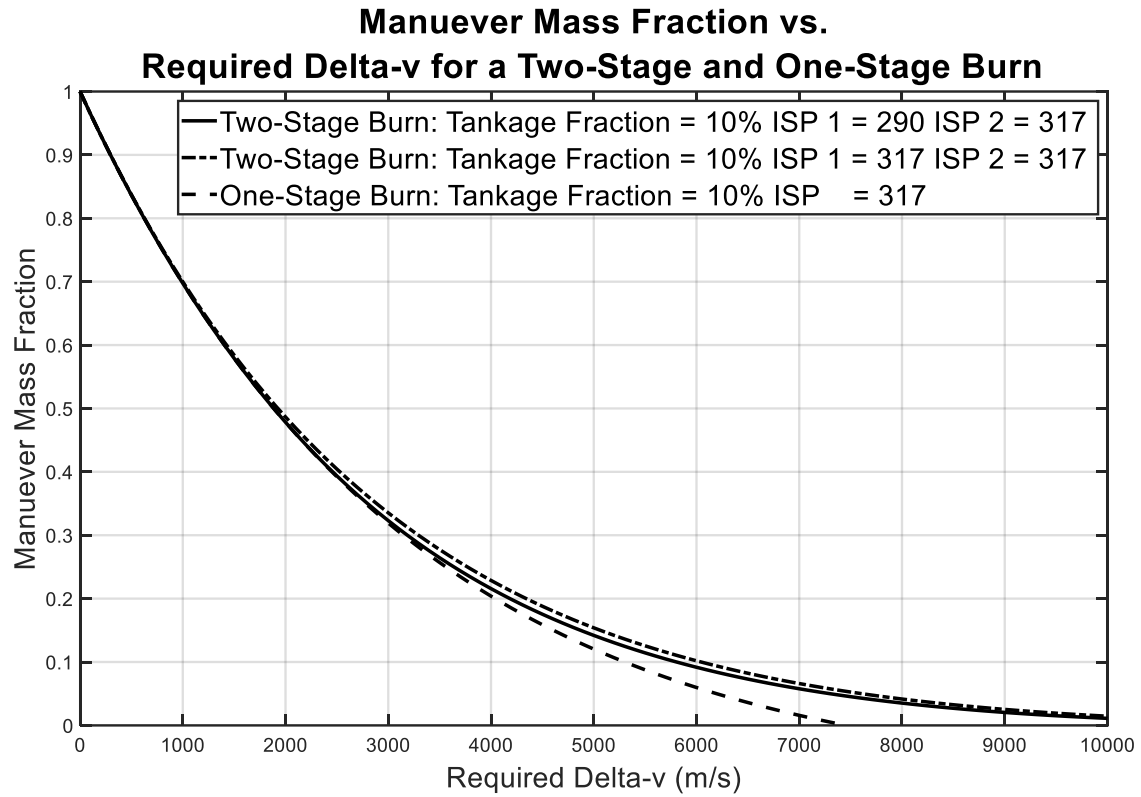


Figure 5.10. Mass fraction for two-stage and one-stage burns over a required delta-v range.

There is a tradeoff for increasing the effectiveness of a PGA. Decreasing the pass distance increases the amount of radiation experienced by the spacecraft (**Figure 3.13**). Therefore, more radiation shielding is required which takes usable mass away from the payload. A larger delta-v requires more propellant and larger tanks which also takes away usable payload mass. All else being equal, a powered Jupiter gravity assist can decrease transit time to the target at the expense of a reduced payload.

Tradeoffs

To view this tradeoff, **Figure 5.10** displays contour lines of constant transit times with combinations of pass distance vs delta-v to reach a target 30 AU from the Sun. The transit time curves begin at 12 years and decrease to 6 years in 1-year increments. The trajectories were calculated using MAnE with the Earth launch date and Jupiter encounter date set to be constant for all missions. This allows the analysis of pass distance vs delta-v as objective as possible by keeping other trajectory parameters constant. The trajectory parameters which were held constant are displayed below **Figure 5.11**.

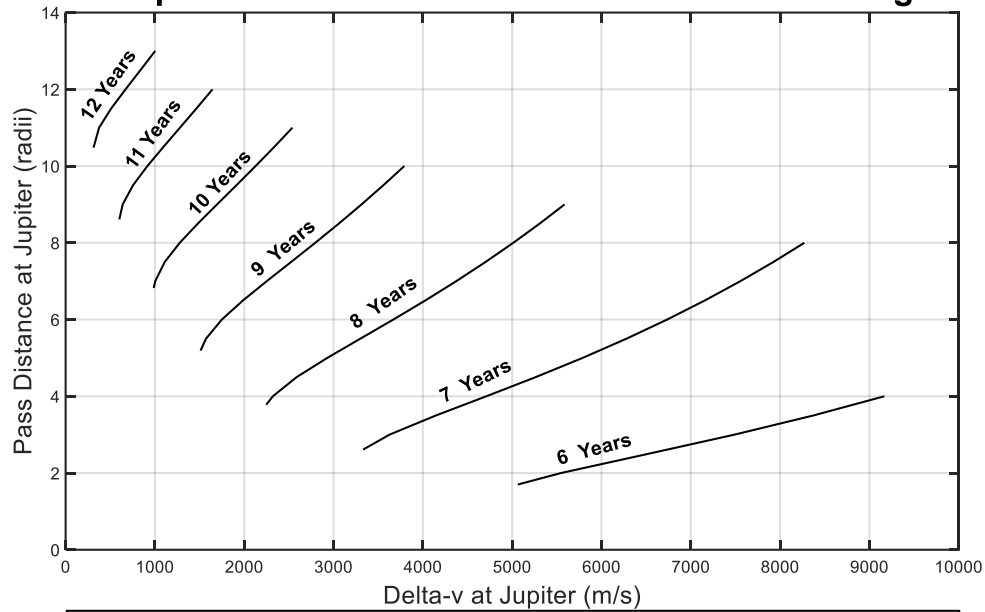
The pass distance vs delta-v tradeoff for constant transit times can be seen by following each curve. For example, a transit time of 9 *years* can be completed with a pass distance of 10 *radii* and delta-v of about 3800 *m/s* or a pass distance of about 6.5 *radii* and delta-v of 2000 *m/s*.

The tradeoff to decrease transit times can also be seen by moving horizontally (delta-v) or vertically (pass distance). For example, a mission can decrease its transit time from 11 years to 10 years by increasing its delta-v by about 1200 *m/s*, decreasing its pass distance by about 3.5 *radii*, or some combination of both.

Effectiveness of Powered Gravity Assists

Using a powered gravity assist will accelerate a spacecraft reducing its transit time to a target. Instead of comparing the transit time changes to specific targets, a more general approach was used to investigate how effective powered gravity assists are and if that effectiveness varies with the delta-v size.

Required Pass Distance and Delta-v at Jupiter for Different Transit Times to 30 AU Target



Mission Parameters
Launch Date: Jan 3, 2077
C3: $85.047 \text{ km}^2/\text{s}^2$
Jupiter Encounter Date: Nov 14, 2078
Excess Speed at Jupiter: 8.58 km/s
Target Distance From Sun: 30 AU

Figure 5.11. Contour lines of constant transit times with varying combinations of pass distances and delta-vs to reach a target 30AU from the Sun.

Flightpaths from MAnE

An optimized Earth-Jupiter-Target trajectory was first established. The target is an arbitrary body set 300 AU from the sun with an eccentricity of 1 and inclination of 0. This circular orbit is essentially in plane with Earth and far enough away to allow for an objective look at how changing the delta-v at Jupiter effects transit time. A visual example of the mission is shown in **Figure 5.12** and **Figure 5.13**.

The pass distance at Jupiter was driven to a minimum of 1.7 using an unpowered trajectory to ensure a constant and efficient flyby radius was achieved. Launch and encounter dates were allowed to vary so MAnE could properly optimize the mission. Once done, the launch date and Jupiter encounter date were locked to be constant for the remainder of the trajectory calculations.

Flightpaths were recorded for different levels of Jupiter delta-v. The purpose is to discover the effect a powered burn at Jupiter will have on transit time. To do this, heliocentric distance is plotted against transit time for various delta-v at Jupiter including an unpowered flyby for reference (**Figure 5.14, 5.15, and 5.16**). Once the Jupiter flyby occurs, the heliocentric distance increases linearly with transit time. This allows for a direct analysis of the effectiveness of a powered Jupiter gravity assist by analyzing the slopes of each line.

Each slope is fit to a linear equation (all with an R^2 value over 0.99). The slope of each line is the rate at which the spacecraft travels (heliocentric distance [AU] traveled per transit time [years]). These slopes are useful in determining the effectiveness of the entire powered gravity assist (i.e., the energy from Jupiter and the burn). Multiplying a slope by a transit time will provide the distance traveled in AU over that period of time. To isolate the effects of only the burn, the slope of the unpowered flightpath can be subtracted from the powered flightpaths. Now the slope only represents the powered portion of the flyby and can be analyzed further.

Trajectory Architecture Used for Powered Gravity Assist Analysis

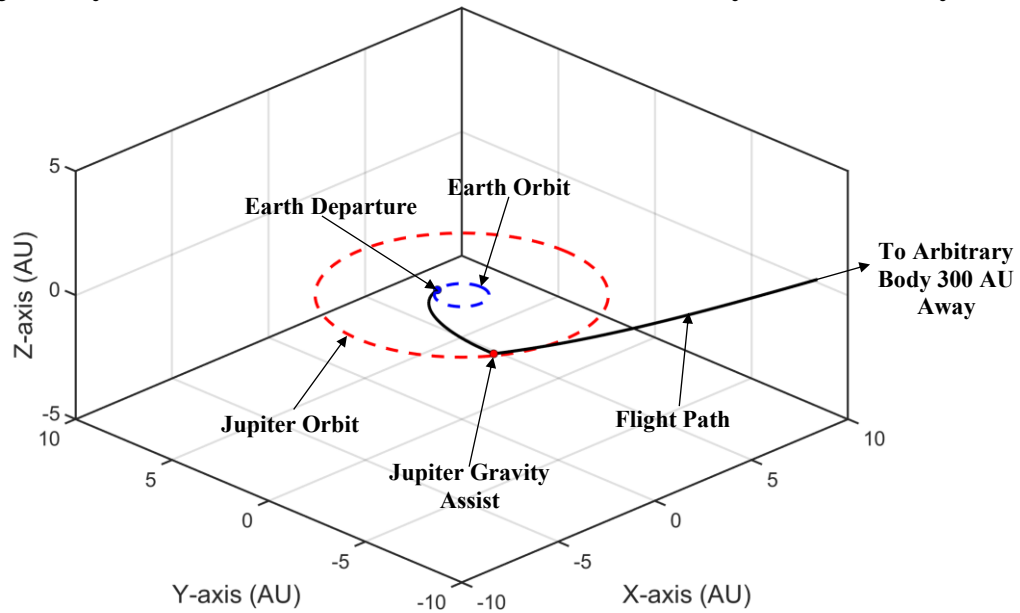


Figure 5.12. General trajectory used for flight path and powered gravity assist analysis centered around the Sun (heliocentric distance on all axes).

Ecliptic Projection of Trajectory Used for Powered Gravity Assist Analysis

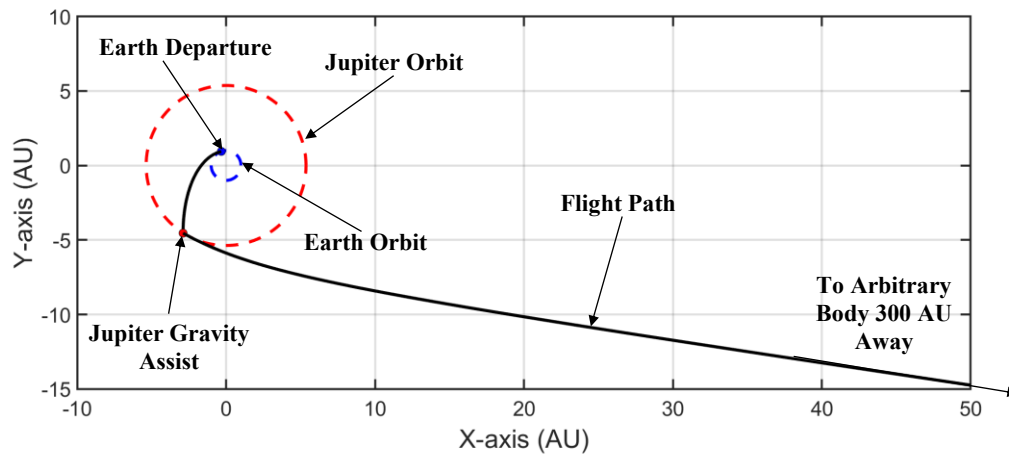


Figure 5.13. 2-D ecliptic projection of general flightpath displaying the spacecraft's radial transit from the Sun.

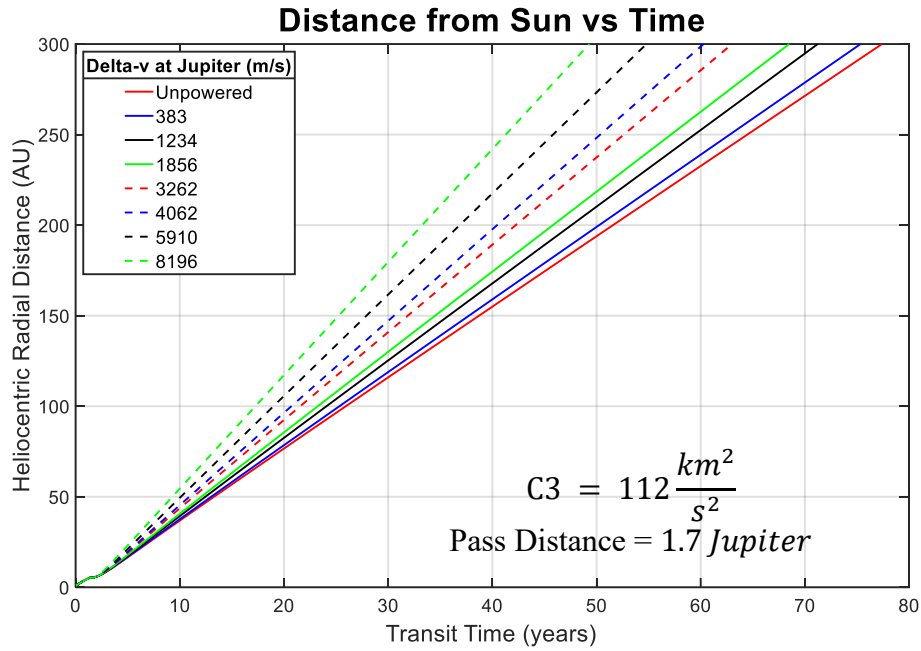


Figure 5.14. Heliocentric distance vs transit time for various delta-v at Jupiter (full).

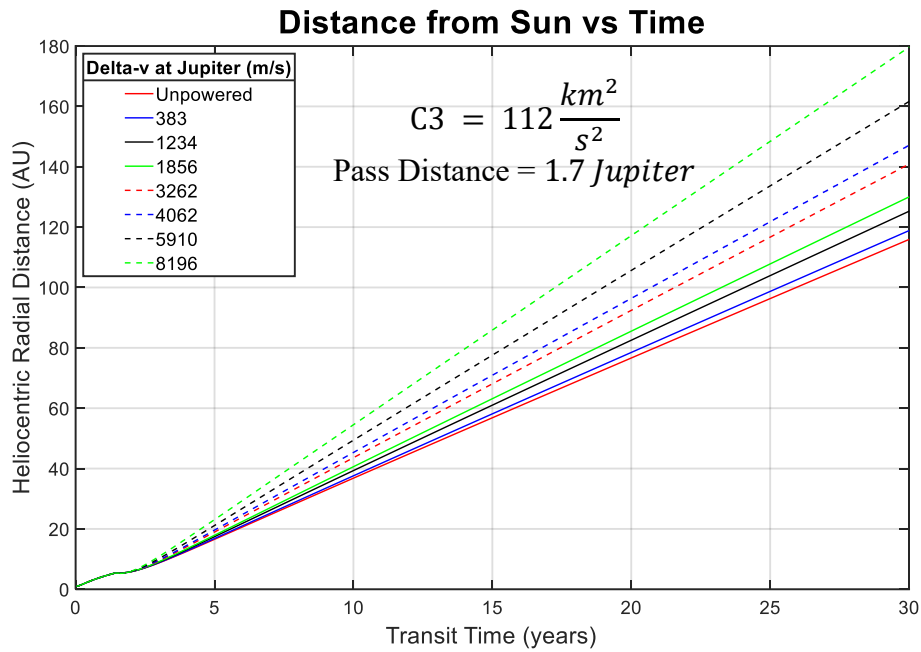


Figure 5.15. Heliocentric distance vs transit time for various delta-v at Jupiter (0-30 years).

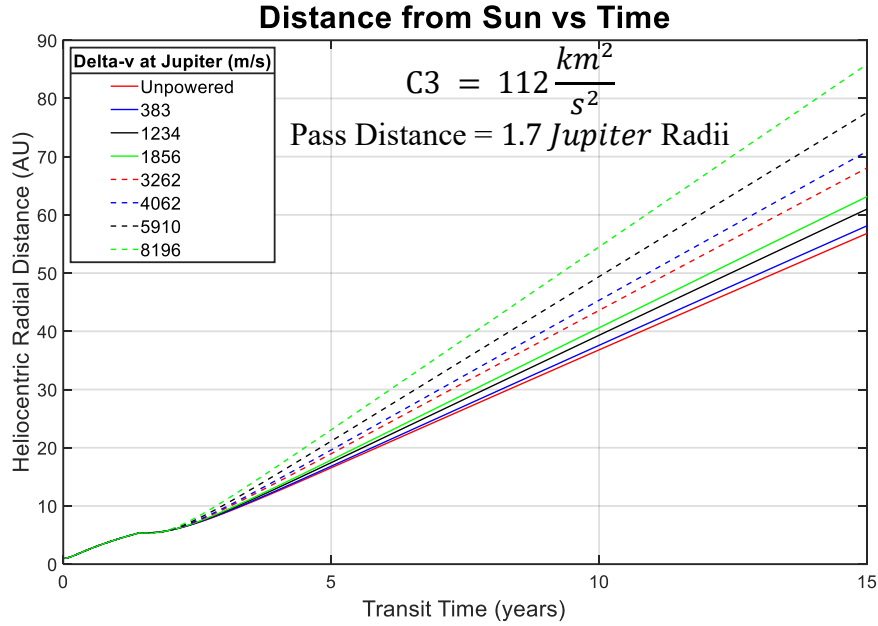


Figure 5.16. Heliocentric distance vs transit time for various delta-v at Jupiter (0-15 years).

First to note is the slopes, once plotted against their respective delta-v at Jupiter, create an almost linear line indicating the benefits of a burn is linear with its size (**Figure 5.17**). From this, it can be determined that for every 3500 m/s delta-v, the slope for this trajectory increases by about 1. In other words, a burn 3500 m/s burn will allow the spacecraft to travel 1 AU further radially per year of transit.

Two additional trajectories to Sedna were calculated to investigate the repeatability of this conclusion. One with a constant $C3$ of $100 \text{ km}^2/s^2$ and the other with a constant excess speed at Jupiter of 12.5 km/s . While the slopes vs delta-v of these two trajectories have different y-intercepts than the arbitrary body missions, the isolated effectiveness of just the burn seems to match for all cases. While three missions do not provide conclusive evidence, further investigation could lead to more convincing evidence.

Sedna Lander Mass Correction

A paper investigating trajectories to Sedna was written in 2023 by Brickley et. al. [15]. Various trajectories architectures were found to be viable for flyby missions.

However, tankage fraction for the capture burn propellant was not accounted for during the payload mass calculations for the lander missions.

The results from Beckley et. al. [15] were reproduced and recalculated using various methods described in previous sections. The reproduction used a single stage capture burn with a tankage fraction of 0% and an ISP of 320. A corrected calculation was done using a single stage capture burn with a tankage fraction of 10% and an ISP of 320.

Two other methods using two-stage capture burns were calculated based on assumptions from previous sections: an unoptimized calculation described in the **Chapter 2: Payload Mass** section, and an optimized calculation described in the **Chapter 5: Two-Stage Capture Burn Optimization** section. The mass curves are displayed in **Figure 5.18** with assumptions in the legend and mass values are tabulated in **Table 5.3**. Accounting for this mass greatly increases the missions' required time of flight to reduce the excess speed (i.e., the capture burn) at the target.

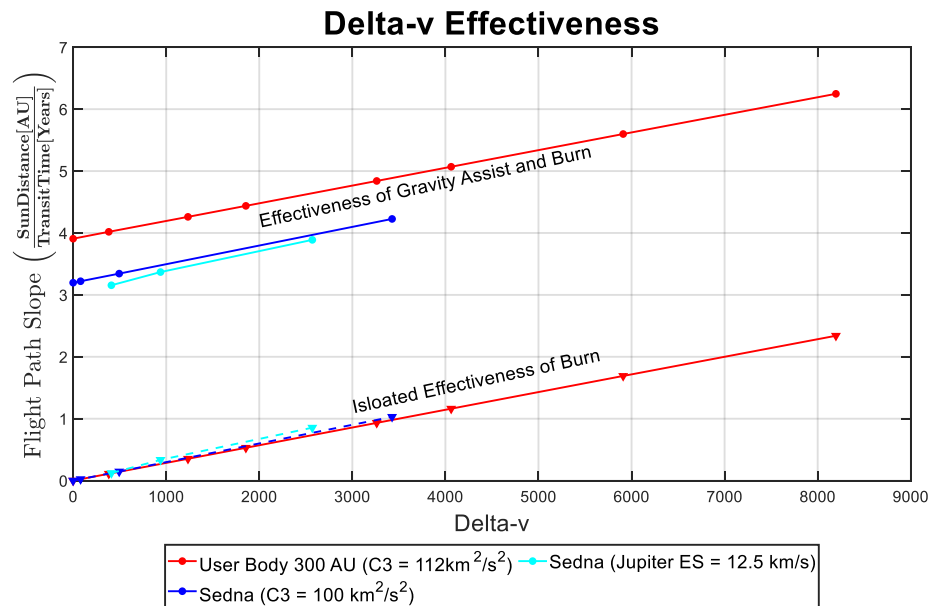


Figure 5.17. The effectiveness of powered Jupiter gravity assists and isolated effects of the delta-v.

Corrected Mass on Target vs Transit Time for Lander Missions to Sedna Using Falcon Heavy Expendable

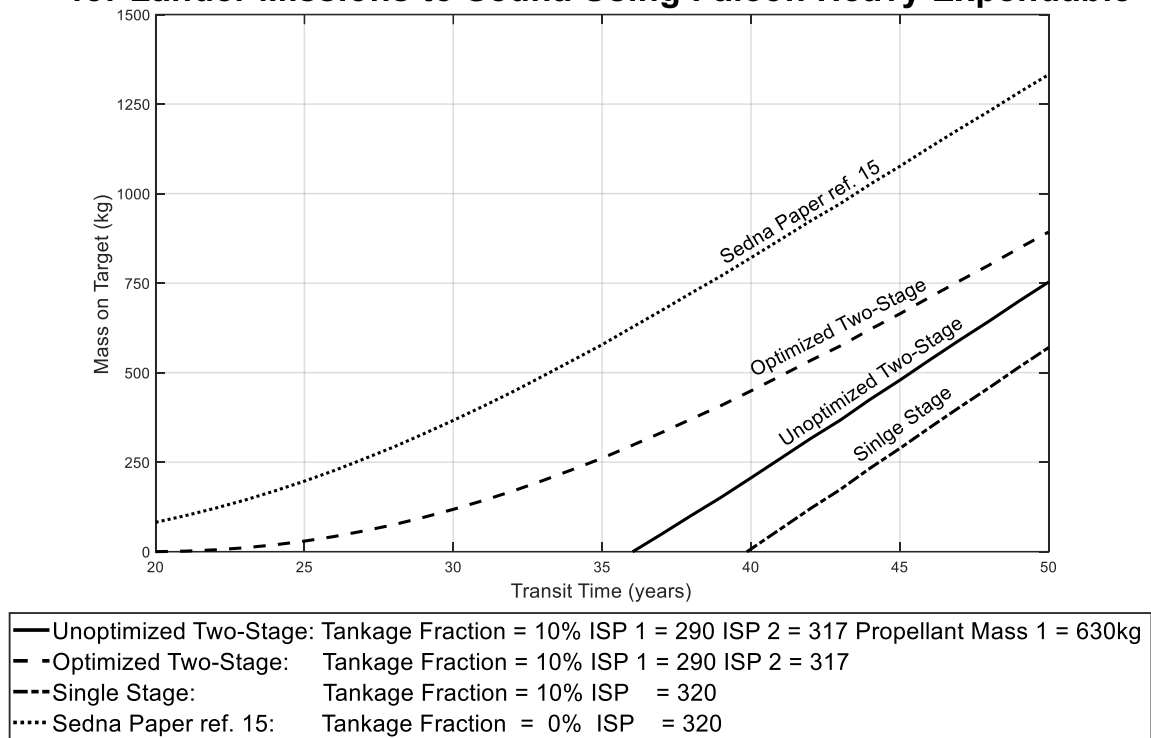


Figure 5.18. Various mass on target curves for lander missions to Sedna.

Table 5.3. Tabulated payload masses for various payload insertion options.

Transit Time (years)	Two Stage Burn Unoptimized (kg)	Two Stage Burn Optimized (kg)	One Stage Burn (kg)	One Stage Burn 0% Tankage Fraction (Original Results [15]) (kg)
20	-576.81	0.07	-805.17	82.05
21	-557.42	1.59	-784.76	100.6
22	-535.64	5.22	-761.88	121.4
23	-511.74	10.99	-736.81	144.19
24	-485.02	19.16	-708.83	169.62
25	-456.2	29.6	-678.7	197.02
26	-424.97	42.45	-646.09	226.66
27	-390.74	58.04	-610.4	259.11
28	-355.53	75.44	-573.72	292.45
29	-316.84	95.9	-533.48	329.04
30	-277	118.23	-492.08	366.67
31	-235.31	142.76	-448.79	406.02
32	-192.13	169.26	-404	446.74
33	-146.24	198.49	-356.44	489.98
34	-99.49	229.27	-308.03	533.99
35	-52.42	261.16	-259.34	578.25
36	-1.42	296.64	-206.62	626.18
37	49.26	332.75	-154.25	673.79
38	101.38	370.67	-100.45	722.7
39	152.07	408.27	-48.15	770.24
40	206.12	449.07	7.58	820.91
41	260.79	491.02	63.92	872.13
42	315.76	533.84	120.54	923.6
43	367.37	574.57	173.66	971.89
44	425.19	620.79	233.16	1025.98
45	478.9	664.24	288.39	1076.19
46	535.26	710.33	346.33	1128.86
47	590.38	755.86	402.96	1180.35
48	643.83	800.42	457.86	1230.25
49	699.56	847.3	515.08	1282.27
50	753.12	892.72	570.04	1332.24

Conclusion

Few missions have ever been conducted to visit the most distant reaches of the Solar System. New Horizons visited only two trans-Neptunian objects out of the hundreds that have been discovered. This is due, in part, to the incredibly lengthy transit times required to complete missions to these far-away objects. Investigation into methods to shorten these transit times and arrive with enough usable payload mass provided insight into viable solutions to make missions to trans-Neptunian objects more enticing.

Summary of Previous Research

Research completed by The University of Tennessee over the last 14 years was compiled, standardized for fair comparison, and analyzed. Each of the three mission types, flyby, orbiter, and lander, revealed insights into how different trajectory properties influenced the outcome of the mission.

It was found that a delta-VEGA could send larger payload masses for missions exceeding 12 years of transit times while more direct Earth to Jupiter to Target missions performed better for transit times less than 12 years. While orbiters and landers need more research, the two most promising targets are currently Ixion and Huya due to the low excess speeds at each target.

Jupiter is an essential part of all the trajectories studied in this paper. Its enormous mass allows for the most effective gravity assists when leaving Earth, decreasing both C3 and transit time. However, there is a limit to how close a spacecraft can get to Jupiter due to its intense radiation belts.

The radiation encountered during a Jupiter flyby depends on a multitude of variables including hyperbolic excess speed, periapsis radius, and inclination. However, the general trend is the closer the spacecraft passes by Jupiter, the more radiation it receives. This is roughly an exponential relation which levels off around 15 Jupiter radii for all data gathered from UTK's previous research. Therefore, a pass distance greater than 15 Jupiter radii result in minimally less radiation while still increasing transit times.

Errors were found in mass calculations for prior missions (Sedna landers and Makemake/Orcus dual-mission) which have been corrected. Moreover, it was realized

that no prior UTK studies of TNO orbiters considered missions using delta-VEGA or other inner solar system flybys to reduce C3 and increase payload mass.

Summary of Trajectory Optimization

Methods were evaluated to improve trajectories and enable reduced transit time and increased payload mass. The research done was completed in a way to show trends and provide tools for future research.

A previously-described, two-probe mission to Orcus and Makemake utilizing a powered Jupiter gravity assist was optimized for the minimum summed transit time. The same process was applied to another two-probe mission to Eris and Borasisi, and relevant trajectory parameters for both mission sets were identified and presented. Based on the New Horizons mission, the probes' final masses were set at 500 kg, and launch was assumed to take place on a Falcon Heavy in Recovery mode. The optimal summed transit time was 24 years for the Makemake/Orcus mission, while for Eris/Borasisi it was 38.9 years due to the extreme distance to Eris.

To improve orbiter and lander payload masses, a method was devised to optimize a two-stage capture burn. Previous research selected a solid rocket booster for the first stage with a set propellant mass. Instead, the optimization process iterates through all possible first-stage propellant masses (from zero to mass before the burn) and solves for the payload mass. The largest payload mass is selected and the optimum first and second-stage propellant masses can then be derived.

A mission mass fraction approach was implemented to evaluate missions for a range of potential launch vehicles quickly. By setting the outbound mass from Earth as 1 instead of using the C3 and launch vehicle performance curve fit, the final mass of the mission will be a value between 0 and 1. This is the usable payload mass arriving on target as a percentage (in decimal form) of the outbound mass from Earth.

A deeper analysis into powered Jupiter gravity assists brought forth two major conclusions. The effectiveness of the maneuver greatly depends on the pass distance and how large the burn is. To investigate this relationship, a contour plot to a target of transit time to 30 AU as a function of Jupiter pass distance and delta-v (also dependent on C3 which was held constant). This showed that the shorter the mission's transit time, the

larger the benefit from the lowering the pass distance at Jupiter. That is, a greater deduction of delta-v was provided when decreasing the pass distance the shorter the transit time. It was found that it requires a Jupiter flyby burn of approximately 3500 m/s to increase the heliocentric radial rate by 1 AU per year.

Future Work

From the UTK compiled data, a few holes in the research were found. The most prominent discovery is the lack of delta-VEGA maneuvers utilized for orbiter missions. These orbiter missions require large capture burns and would benefit from a larger allowable launch mass resulting in more available propellant mass. More complex trajectories such as multi-planet flybys should also be investigated to decrease C3. While lander missions showed more promise for mass on target, they had much longer transit times and fewer targets researched. Expanding the list of possible targets for landing is also a topic of future research which could be done in tandem with further orbital investigation. Both would also benefit from the optimized capture burn method previously discussed.

Dual-target mission optimization can also be completed for more targets to expand the list of viable TNO pairs. Stern et al [30] proposed a single probe, two target orbital encounters with Pluto followed by Quaoar. A dual-target mission to Pluto and Quaoar using methods previously described could be investigated and compared to the results of Stern.

A comparison of different trajectory architectures (e.g. powered Jupiter gravity assist and delta-VEGA) was done for a single target (Orcus). This process provided insights into which architecture would prove more beneficial over different transit times. To glean more information, this process could be repeated for other targets in search of more generalized conclusions.

More research is also needed to prove the generalization of the effectiveness of powered Jupiter gravity assists. More targets with varying trajectory parameters should be investigated and compared to prove or disprove the generalizations made in this paper.

C3 must also be accounted for when investigating the most optimum method of decreasing transit times while maximizing payload mass. There is a tradeoff between C3,

flyby distance at Jupiter, and delta-v at Jupiter when determining usable mass on target. Investigating this tradeoff and how it relates to transit times would provide a more in-depth understanding of which trajectories would provide the most optimum outcomes. There is also a payload mass tradeoff to investigate which should include research into the mass required for radiation shielding for different Jupiter flyby distances.

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APPENDIX

Flyby Payload Mass Example³

Assumptions:

$$g_0 = 9.81 \text{ m/s}^2$$

$$ISP = 317 \text{ s}$$

$$\text{Tankage Fraction} = 10\%$$

$$C3 = 30 \text{ km}^2/\text{s}^2$$

$$\text{Space Burn} = 800 \text{ m/s}$$

$$\text{Jupiter Burn} = 1000 \text{ m/s}$$

Launch Vehicle: Delta IV Heavy with Star-48

Find allowable launch mass with launch vehicle performance curve

$$\begin{array}{ccc}
 \text{Delta IV Heavy} & & \\
 \text{w/ Star-48} & & \\
 C3 = 30 \text{ km}^2/\text{s}^2 & \xrightarrow{\text{Performance Curve}} & \text{Allowable Launch Mass} = m_{ALM} = m_i = 3688 \text{ kg}
 \end{array}$$

Find mass after space burn

$$\begin{aligned}
 m_{f,SB} &= \frac{m_i}{\exp\left(\frac{\Delta v}{g_0 ISP}\right)} \\
 m_{f,SB} &= \frac{3688 \text{ kg}}{\exp\left(\frac{800 \text{ m/s}}{9.81 \text{ m/s}^2 \cdot 317 \text{ s}}\right)}
 \end{aligned}$$

³ To calculate mission mass fraction, set allowable launch mass to 1 and solve using the same process.

$$m_{f,SB} = 2851 \text{ kg}$$

Find mass after Jupiter burn

$$m_{f,JB} = \frac{m_i}{\exp\left(\frac{\Delta v}{g_o ISP}\right)}$$

$$m_{f,JB} = \frac{2851 \text{ kg}}{\exp\left(\frac{1000 \text{ m/s}}{9.81 \text{ m/s}^2 \cdot 317 \text{ s}}\right)}$$

$$m_{f,JB} = 2067 \text{ kg}$$

Find payload mass

$$m^* = 1.1m_{f,JB} - 0.1m_i$$

$$m^* = 1.1(2067 \text{ kg}) - 0.1(3688 \text{ kg})$$

$$m^* = 1905 \text{ kg}$$

Continue Flyby Exampe for Orbiter

Assumptions:

$$m_i = 2067 \text{ kg}$$

$$m_{ALM} = 3688 \text{ kg}$$

$$R_{Earth} = 6,378 \text{ km}$$

$$R_{Huamea} = 797 \text{ km}$$

$$R_p = 897 \text{ km}$$

$$\mu_{Earth} = 3.986 \cdot 10^{14} \text{ m}^3/\text{s}^2$$

$$v_\infty = 4,000 \text{ m/s}$$

$$ISP_1 = 290 \text{ s}$$

$$ISP_2 = 317 \text{ s}$$

$$m_{p,1} = 630 \text{ kg}$$

Circular orbit 100 km above the surface of Haumea

1) Find standard gravitational parameter of Haumea

$$\mu_{Haumea} = \left(\frac{R_{Haumea}}{R_{Earth}} \right)^3 \cdot \mu_{Earth}$$

$$\mu_{Haumea} = \frac{797 \text{ km}}{6,378 \text{ km}} \cdot 3.986 \cdot 10^{14} \text{ m}^3/\text{s}^2$$

$$\mu_{Haumea} = 7.7778 \cdot 10^{11} \text{ m}^3/\text{s}^2$$

2) Find periapsis velocity

$$v_p = \sqrt{2 \left[\frac{v_\infty^2}{2} + \frac{\mu_{Haumea}}{R_p} \right]}$$

$$v_p = \sqrt{2 \left[\frac{(4000 \text{ m/s})^2}{2} + \frac{7.7778 \cdot 10^{11} \text{ m}^3/\text{s}^2}{897,000 \text{ m}} \right]}$$

$$v_p = 4211 \text{ m/s}$$

3) Find velocity required for circular orbit

$$v_c = \sqrt{\frac{\mu_{Huamea}}{R_p}}$$

$$v_p = \sqrt{\frac{7.7778 \cdot 10^{11} \text{ m}^3/\text{s}^2}{897,000 \text{ m}}}$$

$$v_p = 931 \text{ m/s}$$

4) Find required burn for orbital capture

$$\Delta v_{\text{Capture}} = v_{\infty} - v_c$$

$$\Delta v_{\text{Capture}} = 4000 \text{ m/s} - 913 \text{ m/s}$$

$$\Delta v_{\text{Capture}} = 3069 \text{ m/s}$$

5) Find delta-v of first capture burn

$$\Delta v_1 = g_0 \cdot ISP_1 \cdot \ln\left(\frac{m_i}{m_i - m_{p,1}}\right)$$

$$\Delta v_1 = 9.81 \frac{\text{m}}{\text{s}^2} \cdot 317 \text{ s} \cdot \ln\left(\frac{2067 \text{ kg}}{2067 \text{ kg} - 630 \text{ kg}}\right)$$

$$\Delta v_1 = 1130 \text{ m/s}$$

6) Find mass after first capture burn

$$m_{f,1} = m_i - 1.1m_{p,1}$$

$$m_{f,1} = 2067 \text{ kg} - 1.1(630 \text{ kg})$$

$$m_{f,1} = 1374 \text{ kg}$$

7) Find mass after second capture burn

$$m_{f,2} = \frac{m_{f,1}}{\exp\left(\frac{\Delta v_{\text{capture}} - \Delta v_1}{g_o ISP_2}\right)}$$

$$m_{f,2} = \frac{1374 \text{ kg}}{\exp\left(\frac{3069 \frac{m}{s} - 1130 m/s}{9.81 \frac{m}{s^2} \cdot 317 \text{ s}}\right)}$$

$$m_{f,2} = 737 \text{ kg}$$

8) Find payload mass

$$m^* = 1.1m_{f,2} - 0.1m_{ALM}$$

$$m^* = 1.1(737 \text{ kg}) - 0.1(3688 \text{ kg})$$

$$m^* = 442 \text{ kg}$$

Continue Flyby Example for a Lander

Assumptions:

$$m_i = 2067 \text{ kg}$$

$$m_{ALM} = 3688 \text{ kg}$$

$$R_{\text{Earth}} = 6,378 \text{ km}$$

$$R_{\text{Huamea}} = 797 \text{ km}$$

$$\mu_{\text{Earth}} = 3.986 \cdot 10^{14} \text{ m}^3/\text{s}^2$$

$$v_{\infty} = 4,000 \text{ m/s}$$

$$ISP_1 = 290 \text{ s}$$

$$ISP_2 = 317 \text{ s}$$

$$m_{p,1} = 630 \text{ kg}$$

- 1) Find standard gravitational parameter of Haumea

$$\begin{aligned}\mu_{Huamea} &= \frac{R_{Huamea}}{R_{Earth}} \cdot \mu_{Earth} \\ \mu_{Huamea} &= \frac{797 \text{ km}}{6,378 \text{ km}} \cdot 3.986 \cdot 10^{14} \text{ m}^3/\text{s}^2 \\ \mu_{Huamea} &= 7.7778 \cdot 10^{11} \text{ m}^3/\text{s}^2\end{aligned}$$

- 1) Find velocity required to land

$$\begin{aligned}\Delta v_{land} &= \sqrt{2 \left[\frac{v_{\infty}^2}{2} + \frac{\mu_{Huamea}}{R_{Huamea}} \right]} \\ \Delta v_{land} &= \sqrt{2 \left[\frac{(4000 \text{ m/s})^2}{2} + \frac{7.7778 \cdot 10^{11} \text{ m}^3/\text{s}^2}{797,000 \text{ m}} \right]} \\ \Delta v_{land} &= 4237 \text{ m/s}\end{aligned}$$

- 1) Find delta-v of first capture burn

$$\begin{aligned}\Delta v_1 &= g_0 \cdot ISP_1 \cdot \ln \left(\frac{m_i}{m_i - m_{p,1}} \right) \\ \Delta v_1 &= 9.81 \frac{\text{m}}{\text{s}^2} \cdot 317 \text{ s} \cdot \ln \left(\frac{2067 \text{ kg}}{2067 \text{ kg} - 630 \text{ kg}} \right)\end{aligned}$$

$$\Delta v_1 = 1130 \text{ m/s}$$

2) Find mass after first capture burn

$$m_{f,1} = m_i - 1.1m_{p,1}$$

$$m_{f,1} = 2067 \text{ kg} - 1.1(630 \text{ kg})$$

$$m_{f,1} = 1374 \text{ kg}$$

3) Find mass after second capture burn

$$m_{f,2} = \frac{m_{f,1}}{\exp\left(\frac{\Delta v_{\text{capture}} - \Delta v_1}{g_o ISP_2}\right)}$$

$$m_{f,2} = \frac{1374 \text{ kg}}{\exp\left(\frac{4237 \text{ m/s} - 1130 \text{ m/s}}{9.81 \frac{\text{m}}{\text{s}^2} \cdot 317 \text{ s}}\right)}$$

$$m_{f,2} = 500 \text{ kg}$$

4) Find payload mass

$$m^* = 1.1m_{f,2} - 0.1m_{ALM}$$

$$m^* = 1.1(500 \text{ kg}) - 0.1(3688 \text{ kg})$$

$$m^* = 181 \text{ kg}$$

Two-Stage Burn Propellant Masses Optimization⁴

Description:

- Define:
 - ISP for both stages
 - g_0
- Set desire capture burn (CB) and initial mass (m_i). Both can also be iterated for desired range.
- Iterate from $m_{p,1} = 0, 1, 2, \dots, m_i$

Assumptions:

$$\begin{aligned}g_0 &= 9.81 \text{ m/s}^2 \\ \text{ISP}_1 &= 290 \text{ s} \\ \text{ISP}_2 &= 317 \text{ s} \\ m_i &= 2000 \text{ kg} \\ CB &= 6000 \text{ m/s}\end{aligned}$$

For these assumptions, the code converges to ideal values of:

$$\begin{aligned}m_{p,1} &= 1160 \text{ kg} \\ m_{p,2} &= 491 \text{ kg} \\ m^* &= 183 \text{ kg}\end{aligned}$$

⁴ To calculate mass fraction, set initial mass to 1 and solve using the same process.

Note: All values of $m_{p,1}$ from 0:2000 kg can be checked via code or hand using the same process detailed below for proof of optimization. Ideal values are shown below to ensure process and code is correct.

- 1) Find delta-v of first burn (Δv_1)

$$\begin{aligned}\Delta v_1 &= g_0 \cdot ISP_1 \cdot \ln\left(\frac{m_i}{m_i - m_{p,1}}\right) \\ \Delta v_1 &= 9.81 \frac{m}{s^2} \cdot 290 s \cdot \ln\left(\frac{2000 kg}{2000 kg - 1160 kg}\right) \\ \Delta v_1 &= 2467.9 m/s\end{aligned}$$

- 2) Calculate mass after burn and tanks are jettisoned

$$\begin{aligned}m_{f,1} &= m_i - 1.1m_{p,1} \\ m_{f,1} &= 2000 kg - 1.1(1160 kg) \\ m_{f,1} &= 724 kg\end{aligned}$$

- 3) Calculate delta-v of second burn (Δv_2)

$$\begin{aligned}\Delta v_2 &= CB - \Delta v_1 \\ \Delta v_2 &= 6000 m/s - 2467.9 m/s \\ \Delta v_2 &= 3532.5 m/s\end{aligned}$$

- 4) Calculate mass after second burn (including tanks) ($m_{f,2}$)

$$m_{f,2} = \frac{m_{f,1}}{\exp\left(\frac{\Delta v_2}{g_0 ISP_2}\right)}$$

$$m_{f,2} = \frac{724 \text{ kg}}{\exp\left(\frac{3532 \text{ m/s}}{9.81 \frac{\text{m}}{\text{s}^2} \cdot 317 \text{ s}}\right)}$$

$$m_{f,2} = 232.5 \text{ kg}$$

5) Calculate propellant mass of stage two burn ($m_{p,2}$)

$$m_{p,2} = m_{f,1} - m_{f,2}$$

$$m_{p,2} = 724 \text{ kg} - 232.5 \text{ kg}$$

$$m_{p,2} = 491 \text{ kg}$$

6) Calculate payload mass (m^*)

$$m^* = m_{f,2} - 0.1m_{p,2}$$

$$m^* = 232.5 \text{ kg} - 0.1(491 \text{ kg})$$

$$m^* = 183 \text{ kg}$$

7) Calculate propellant mass ratio

$$\text{ratio} = \frac{\text{stage 1 propellant mass}}{\text{stage 2 propellant mass}}$$

$$\text{ratio} = \frac{1160 \text{ kg}}{591 \text{ kg}}$$

$$\text{ratio} = 2.36$$

Dual-Target Mission Optimization: Probes Split After Earth Re-encounter and Before Jupiter Encounter

Assumptions:

$$\begin{aligned}
 m_{ALM} &= 8090 \text{ kg} \\
 C3 &= 30.9 \text{ kg}^2/\text{s}^2 \\
 g_0 &= 9.81 \text{ m/s}^2 \\
 \text{Tankage Fraction} &= 10 \% \\
 m^* &= 500 \text{ kg} \\
 \text{Space Burn} &= 1,055 \text{ m/s} \\
 ISP &= 317 \text{ s} \\
 \text{Launch Vehicle} &= \text{Falcon Heavy Expendable}
 \end{aligned}$$

Makemake:

$$\Delta v \text{ at Jupiter} = 3784 \text{ m/s (Iterated)}$$

Mass After Space Burn:

- 1) Solve for mass after space burn

$$\begin{aligned}
 \text{Mass After Space Burn} &= \frac{m_{ALM}}{\exp\left(\frac{\Delta v}{g_0 ISP}\right)} \\
 m_{SB} &= \frac{8090 \text{ kg}}{\exp\left(\frac{1055 \text{ m/s}}{9.81 \text{ m/s}^2 \cdot 317 \text{ s}}\right)} \\
 m_{SB} &= 5762 \text{ kg}
 \end{aligned}$$

- 2) Mass after tanks are jetisoned

$$\begin{aligned}
\text{Mass After Tanks} &= m_{SB} - 0.1(m_{ALM} - m_{SB}) \\
m_{SB,T} &= 5762 \text{ kg} - 0.1(8090 \text{ kg} - 5762 \text{ kg}) \\
m_{SB,T} &= 5530 \text{ kg}
\end{aligned}$$

Mission 1 (Makemake):

- 1) Solve for initial mass of mission using set delta-v at Jupiter

$$\begin{aligned}
m_{i,M_1} &= \frac{\exp\left(\frac{\Delta v}{g_0 ISP}\right) m^*}{1.1 - 0.1 \exp\left(\frac{\Delta v}{g_0 ISP}\right)} \\
m_{i,M_1} &= \frac{\exp\left(\frac{3784 \text{ m/s}}{9.81 \text{ m/s}^2 \cdot 317 \text{ s}}\right) 500 \text{ kg}}{1.1 - 0.1 \exp\left(\frac{3784 \text{ m/s}}{9.81 \text{ m/s}^2 \cdot 317 \text{ s}}\right)} \\
m_{i,M_1} &= 2214 \text{ kg}
\end{aligned}$$

- 2) Solve for final mass of mission

$$\begin{aligned}
m_{f,M_1} &= \frac{m^* + 0.1 m_{i,M_1}}{1.1} \\
m_{f,M_1} &= \frac{500 \text{ kg} + 0.1(2214 \text{ kg})}{1.1} \\
m_{f,M_1} &= 656 \text{ kg}
\end{aligned}$$

- 3) Solve for propellant mass

$$\begin{aligned}
m_{p_1} &= m_{i,M_1} - m_{f,M_1} \\
m_{p_1} &= 2214 \text{ kg} - 656 \text{ kg} \\
m_{p_1} &= 1559 \text{ kg}
\end{aligned}$$

- 4) Solve for tankage mass

$$m_{T_1} = 0.1m_{p_1}$$

$$m_{T_1} = 0.1(1159 \text{ kg})$$

$$m_{T_1} = 156 \text{ kg}$$

Mission 2 (Orcus):

- 1) Solve for initial mass of mission using set delta-v at Jupiter

$$m_{i,M_2} = m_{SB,T} - m_{i,M_1}$$

$$m_{i,M_2} = 5530 \text{ kg} - 2214 \text{ kg}$$

$$m_{i,M_2} = 3316 \text{ kg}$$

- 2) Solve for final mass of mission

$$m_{f,M_2} = \frac{m^* + 0.1m_{i,M_2}}{1.1}$$

$$m_{f,M_1} = \frac{500 \text{ kg} + 0.1(3316 \text{ kg})}{1.1}$$

$$m_{f,M_1} = 756 \text{ kg}$$

- 3) Solve for allowable Δv at Jupiter

$$\Delta v_2 = g_0 ISP \ln \left(\frac{m_{i,M_2}}{m_{f,M_2}} \right)$$

$$\Delta v_2 = 9.81 \text{ m/s}^2 \cdot 317 \text{ s} \cdot \ln \left(\frac{3316 \text{ kg}}{756 \text{ kg}} \right)$$

$$\Delta v_2 = 4597 \text{ m/s}$$

- 4) Solve for propellant mass

$$m_{p_2} = m_{i,M_2} - m_{f,M_2}$$

$$m_{p_2} = 3316 \text{ kg} - 756 \text{ kg}$$

$$m_{p_2} = 2560 \text{ kg}$$

5) Solve for tankage mass

$$m_{T_2} = 0.1m_{p_2}$$

$$m_{T_2} = 0.1(2560 \text{ kg})$$

$$m_{T_2} = 256 \text{ kg}$$

Dual-Target Mission Optimization: Probes Split After Earth Launch and Before Space Burn

Assumptions:

$$\begin{aligned}
 m_{ALM} &= 8610 \text{ kg} \\
 C3 &= 28 \text{ kg}^2/\text{s}^2 \\
 g_0 &= 9.81 \text{ m/s}^2 \\
 \text{Tankage Fraction} &= 10 \% \\
 m^* &= 500 \text{ kg} \\
 \text{Eris Space Burn} &= 800 \text{ m/s} \\
 \text{Borasisi Space Burn} &= 625 \text{ m/s} \\
 ISP &= 317 \text{ s} \\
 \text{Launch Vehicle} &= \text{Falcon Heavy Expendable}
 \end{aligned}$$

Eris:

$$\Delta v \text{ at Jupiter} = 4852 \text{ m/s (Iterated)}$$

Mission 1 (Eris):

- 1) Solve for initial mass before Jupiter burn using set delta-v at Jupiter (also mass after space burn)

$$\begin{aligned}
 m_{i,J_1} &= \frac{\exp\left(\frac{\Delta v}{g_0 ISP}\right) m^*}{1.1 - 0.1 \exp\left(\frac{\Delta v}{g_0 ISP}\right)} \\
 m_{i,J_1} &= \frac{\exp\left(\frac{4852 \text{ m/s}}{9.81 \text{ m/s}^2 \cdot 317 \text{ s}}\right) 500 \text{ kg}}{1.1 - 0.1 \exp\left(\frac{4852 \text{ m/s}}{9.81 \text{ m/s}^2 \cdot 317 \text{ s}}\right)} \\
 m_{i,J_1} &= 3814 \text{ kg}
 \end{aligned}$$

2) Solve for mass initial mission mass before space burn

$$m_{i,M_1} = \exp\left(\frac{\Delta v}{g_0 ISP}\right) m_{i,J_1}$$

$$m_{i,M_1} = \exp\left(\frac{800 \text{ m/s}}{9.81 \text{ m/s}^2 \cdot 317 \text{ s}}\right) 3814 \text{ kg}$$

$$m_{i,M_1} = 4933 \text{ kg}$$

3) Solve for final mass of mission

$$m_{f,M_1} = \frac{m^* + 0.1m_{i,M_1}}{1.1}$$

$$m_{f,M_1} = \frac{500 \text{ kg} + 0.1(4933 \text{ kg})}{1.1}$$

$$m_{f,M_1} = 903 \text{ kg}$$

4) Solve for propellant mass

$$m_{p_1} = m_{i,M_1} - m_{f,M_1}$$

$$m_{p_1} = 4933 \text{ kg} - 903 \text{ kg}$$

$$m_{p_1} = 4030 \text{ kg}$$

5) Solve for tankage mass

$$m_{T_1} = 0.1m_{p_1}$$

$$m_{T_1} = 0.1(4030 \text{ kg})$$

$$m_{T_1} = 403 \text{ kg}$$

Mission 2 (Borasisi):

- 1) Solve for initial mass of mission using set delta-v at Jupiter

$$m_{i,M_2} = m_{ALM} - m_{i,M_1}$$

$$m_{i,M_2} = 8610 \text{ kg} - 4933 \text{ kg}$$

$$m_{i,M_2} = 3677 \text{ kg}$$

- 2) Solve for mass after space burn

$$m_{SB_2} = \frac{m_{i,M_2}}{\exp\left(\frac{\Delta v}{g_0 ISP}\right)}$$

$$m_{SB_2} = \frac{3677 \text{ kg}}{\exp\left(\frac{625 \text{ m/s}}{9.81 \text{ m/s}^2 \cdot 317 \text{ s}}\right)}$$

$$m_{SB_2} = 3008 \text{ kg}$$

- 3) Solve for final mass of mission

$$m_{f,M_2} = \frac{m^* + 0.1m_{i,M_2}}{1.1}$$

$$m_{f,M_1} = \frac{500 \text{ kg} + 0.1(3677 \text{ kg})}{1.1}$$

$$m_{f,M_1} = 789 \text{ kg}$$

- 4) Solve for allowable Δv at Jupiter

$$\Delta v_2 = g_0 ISP \ln\left(\frac{m_{i,M_2}}{m_{f,M_2}}\right)$$

$$\Delta v_2 = 9.81 \text{ m/s}^2 \cdot 317 \text{ s} \cdot \ln\left(\frac{3008 \text{ kg}}{789 \text{ kg}}\right)$$

$$\Delta v_2 = 4161 \text{ m/s}$$

- 5) Solve for propellant mass

$$m_{p_2} = m_{i,M_2} - m_{f,M_2}$$

$$m_{p_2} = 3677 \text{ kg} - 789 \text{ kg}$$

$$m_{p_2} = 2888 \text{ kg}$$

6) Solve for tankage mass

$$m_{T_2} = 0.1m_{p_2}$$

$$m_{T_2} = 0.1(2888 \text{ kg})$$

$$m_{T_2} = 289 \text{ kg}$$

Table 5.4 List of flyby trajectories previously researched by The University of Tennessee and used for the summary section of this paper.

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust 2001 UR163	E-Delta-VEGA-JGA-TNO	2/19/2030	25	33.4	1603.2	-	10.33	-	-	11	3688.6	0.557
High Thrust 2002 AW197	E-Delta-VEGA-JGA-TNO	10/6/1934	17.2	27.11	890	978	9.73	-	-	12.5	5038.2	0.726
	E-Delta-VEGA-JGA-TNO	10/9/1946	13.1	26.65	800	891	7.08	-	-	12.5	5245.1	0.750
High Thrust 2002 KX14	E-Delta-VEGA-JGA-TNO	9/16/2036	22	32.4	973.7	-	56.7	-	-	10	4489.1	0.704
High Thrust 2002 MS4	E-Delta-VEGA-JGA-TNO	10/18/2037	16.7	28.15	1320	636	2.7	-	-	14.98	4226.9	0.620
High Thrust 2002 XW93	E-Delta-VEGA-JGA-TNO	5/12/2032	17	34.3	712.7	-	11.22	-	-	14.64	4790.2	0.775
High Thrust 2003 QA91	E-Delta-VEGA-JGA-TNO	1/6/2041	18	42	257.1	-	13.54	-	-	9.5	4991.7	0.913
High Thrust Arrokoth	E-Delta-VEGA-JGA-TNO	12/19/2037	21	34.4	2500	-	3.92	-	-	10.98	2422.0	0.392
High Thrust Borrasis	E-Delta-VEGA-JGA-TNO	1/20/2028	18	33	100.4	-	2.65	-	-	14.19	6092.4	0.965
High Thrust Chaos	E-Delta-VEGA-JGA-TNO	3/18/2030	24	33.4	692.7	-	3.99	-	-	7.84	4895.0	0.780
High Thrust Deucalion	E-Delta-VEGA-JGA-TNO	10/27/2037	21	36.3	847.9	-	3.04	-	-	9.95	4416.7	0.737
High Thrust Eris	E-JGA-TNO	4/3/2032	24.66	130.13	-	-	9.86	-	-	17.88	1479.4	1.000
	E-JGA-TNO	4/7/2044	24.66	131.21	-	-	10.86	-	-	17.56	1458.9	1.000

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Humeca	E-JGA-TNO	10/20/2014	16.45	145.12	-	-	12.9	-	-	13.45	1229.4	1.000
	E-JGA-TNO	9/25/2025	16.45	115.53	-	-	6.79	-	-	13.89	1798.9	1.000
	E-JGA-TNO	9/25/2025	14.25	125.53	-	-	4.65	-	-	16.58	1571.1	1.000
	E-JGA-TNO	11/1/2026	16.45	134.51	-	-	12.51	-	-	13.23	1398.9	1.000
	E-JGA-TNO	9/23/2037	16.45	110.2	-	-	6.01	-	-	13.61	1938.0	1.000
	E-JGA-TNO	10/29/2038	16.45	121	-	-	12.27	-	-	12.93	1669.2	1.000
	E-JGA-TNO	9/21/2025	26.16	102.4	-	-	15.36	50.6	-	7.49	2166.5	1.000
	E-JGA-TNO	9/21/2025	23.42	104	-	-	13.32	56.6	-	8.72	2117.0	1.000
	E-JGA-TNO	9/21/2025	22.06	105.1	-	-	12.16	224	-	9.47	2083.8	1.000
	E-JGA-TNO	9/21/2025	20.69	106.6	-	-	10.92	660	-	10.32	2039.6	1.000
	E-JGA-TNO	9/21/2025	19.32	108.5	-	-	9.61	1610	-	11.29	1985.2	1.000

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
Low Thrust Huamea	E-JGA-TNO	6/1/2025-12/31/2025	30	55.4	-	-	19.4	67.4	-	6.16	4424.8	1.000
	E-JGA-TNO	6/1/2025-12/31/2025	27	57	-	-	17.4	70.2	-	7.17	4315.0	1.000
	E-JGA-TNO	6/1/2025-12/31/2025	24	57.4	-	-	14.9	71.9	-	8.46	4288.0	1.000
	E-JGA-TNO	6/1/2025-12/31/2025	20	57.9	-	-	11.6	2910	-	10.8	4254.5	1.000
	E-JGA-TNO	6/1/2025-12/31/2025	17	58.4	-	-	7.78	14500	-	13.4	4221.3	1.000
	E-JGA-TNO	6/1/2025-12/31/2025	14	59.1	-	-	4.64	25600	-	17	4175.2	1.000
High Thrust Huya	E-JGA-TNO	10/22/2026	20.15	91.32	-	-	7.58	11300	-	5.67	2549.1	1.000
	E-JGA-TNO	10/22/2026	18.78	91.36	-	-	6.81	16600	-	6.32	2547.5	1.000
	E-JGA-TNO	10/22/2026	17.41	91.39	-	-	5.9	21000	-	7.09	2546.4	1.000
	E-JGA-TNO	10/22/2026	16.05	91.78	-	-	4.89	21600	-	8.01	2531.7	1.000
	E-JGA-TNO	10/22/2026	14.69	92.87	-	-	3.81	25400	-	9.12	2491.1	1.000
	E-JGA-TNO	10/22/2026	13.32	95.3	-	-	2.78	39900	-	10.48	2403.2	1.000

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
Low Thrust Huy	E-LT-JGA-TNO	5/1/2026-8/31/2026	25.4	47.4	-	-	12.1	2290	-	4.02	5020.1	1.000
	E-LT-JGA-TNO	5/1/2026-8/31/2026	22	48.9	-	-	10.8	5970	-	4.97	4902.3	1.000
	E-LT-JGA-TNO	5/1/2026-8/31/2026	19	49.6	-	-	8.79	14700	-	6.24	4848.4	1.000
	E-LT-JGA-TNO	5/1/2026-8/31/2026	16	51	-	-	6.02	39300	-	8.09	4742.4	1.000
	E-LT-JGA-TNO	5/1/2026-8/31/2026	14	53	-	-	4	53700	-	9.83	4595.1	1.000
	E-LT-JGA-TNO	5/1/2026-8/31/2026	12	57	-	-	2.27	116000	-	12.2	4315.0	1.000
High Thrust Ixion	E-JGA-TNO	10/25/2026	25.9	88.6	-	-	18.9	752	-	5.58	2654.7	1.000
	E-JGA-TNO	10/25/2026	24.5	89.2	-	-	16.7	703	-	5.97	2630.9	1.000
	E-JGA-TNO	10/25/2026	23.2	90.2	-	-	14.5	756	-	6.38	2592.0	1.000
	E-JGA-TNO	10/25/2026	21.8	91.5	-	-	12.3	984	-	6.84	2542.2	1.000
	E-JGA-TNO	10/25/2026	20.4	92.14	-	-	9.79	2460	-	7.34	2518.2	1.000
	E-JGA-TNO	10/25/2026	19.05	92.2	-	-	8.56	5370	-	8.17	2515.9	1.000

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
Low Thrust Ixion	E-LT-JGA-TNO	7/1/2028-10/31/2028	28.1	54.6	-	-	28.5	54	-	4.11	4480.8	1.000
	E-LT-JGA-TNO	7/1/2028-10/31/2028	25	56.5	-	-	28.5	70	-	4.74	4349.0	1.000
	E-LT-JGA-TNO	7/1/2028-10/31/2028	22	57.2	-	-	27.8	73	-	5.69	4301.5	1.000
	E-LT-JGA-TNO	7/1/2028-10/31/2028	18	58.3	-	-	24.8	84	-	7.67	4227.9	1.000
	E-LT-JGA-TNO	7/1/2028-10/31/2028	15	59.8	-	-	20.4	102	-	9.99	4129.7	1.000
	E-LT-JGA-TNO	7/1/2028-10/31/2028	12	63.4	-	-	14	145	-	13.6	3904.0	1.000
	E-Delta-VEGA-JGA-TNO	11/17/2038	14.8	27.46	1140	514	10.79	-	-	11.8	4569.7	0.662
High Thrust Kagara	E-Delta-VEGA-JGA-TNO	7/13/2034	19	32.9	786.3	-	8.33	-	-	11	4769.2	0.754
High Thrust Lempo	E-Delta-VEGA-JGA-TNO	3/28/2038	21	38.3	802.3	-	10.55	-	-	7.36	4349.9	0.750
High Thrust Makemake	E-JGA-TNO	8/21/2024	16.04	117.95	-	-	3.99	-	-	15.62	1740.0	1.000
	E-JGA-TNO	8/24/2036	16.19	113.02	-	-	3.43	-	-	15.65	1862.8	1.000
	E-Delta-VEGA-JGA-TNO	7/19/2034	18	29.55	1720	945	3.21	-	-	16.02	3553.7	0.533
	E-PJGA-TNO	8/22/2036	16.4	110.9	-	-	4.8	-	1168.2	15.6	1257.2	0.656
	E-PJGA-TNO	8/22/2036	13.7	110.9	-	-	2.8	-	1956.2	19.6	932.9	0.486

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Makemake	E-PJGA-TNO	8/22/2036	13.0	110.9	-	-	2.5	-	2221.4	20.9	840.9	0.438
	E-PJGA-TNO	8/22/2036	12.3	110.9	-	-	2.2	-	2526.0	22.3	744.6	0.388
	E-PJGA-TNO	8/22/2036	11.6	110.9	-	-	1.9	-	2879.9	24.0	643.9	0.336
	E-PJGA-TNO	8/22/2036	11.0	110.9	-	-	1.7	-	3294.0	25.9	539.7	0.281
	E-PJGA-TNO	8/22/2036	10.3	110.9	-	-	1.7	-	4092.3	28.1	374.1	0.195
	E-PJGA-TNO	8/22/2036	9.6	110.9	-	-	1.7	-	5406.0	30.7	179.1	0.093
	E-PJGA-TNO	10/27/2048	16.44	117.66	-	-	3.09	-	698.1	15.57	1360.5	0.779
	E-PJGA-TNO	10/27/2048	13.70	117.66	-	-	1.96	-	1425.7	19.58	1040.2	0.595
	E-PJGA-TNO	10/27/2048	13.01	117.66	-	-	1.73	-	1666.4	20.88	949.8	0.544
	E-PJGA-TNO	10/27/2048	12.33	117.66	-	-	1.7	-	1939.5	22.34	855.2	0.490
	E-PJGA-TNO	10/27/2048	11.64	117.66	-	-	1.7	-	2672.7	24	638.9	0.366
	E-PJGA-TNO	10/27/2048	10.96	117.66	-	-	1.7	-	3797.2	25.91	392.0	0.224
	E-PJGA-TNO	10/27/2048	10.27	117.66	-	-	1.7	-	5163.3	28.1	190.6	0.109
	E-PJGA-TNO	10/27/2048	9.59	117.66	-	-	1.7	-	6815.1	30.67	40.0	0.023

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Orcus	E-Delta-VEGA-JGA-TNO	5/12/2031	19	34.8	701	-	3.93	-	-	14.51	4772.3	0.778
	E-Delta-VEGA-TNO	8/27/2033	20	27.403	742.2	-	6.17	-	1032.9	12.21	3601.4	0.522
	E-Delta-VEGA-TNO	8/27/2033	19	27.365	739.2	-	5.39	-	1149	13.18	3450.3	0.499
	E-Delta-VEGA-TNO	8/27/2033	18	27.328	736.6	-	4.63	-	1286.7	14.28	3276.1	0.474
	E-Delta-VEGA-TNO	8/27/2033	17	27.294	734.4	-	3.91	-	1450.6	15.54	3076.8	0.445
	E-Delta-VEGA-TNO	8/27/2033	16	27.263	732.7	-	3.24	-	1646.9	17	2849.6	0.412
	E-Delta-VEGA-TNO	8/27/2033	15	27.235	731.6	-	2.62	-	1884.3	18.72	2591.6	0.374
	E-Delta-VEGA-TNO	8/27/2033	14	27.213	731.2	-	2.07	-	2175	20.76	2299.8	0.332
	E-Delta-VEGA-TNO	8/27/2033	13	27.094	703.5	-	1.7	-	2649.9	23.26	1902.6	0.274
	E-Delta-VEGA-TNO	8/27/2033	11.9945 205	27.181	647.3	-	1.7	-	4060.9	26.54	984.2	0.142
	E-Delta-VEGA-TNO	2045	20	27.119	813	-	4.53	-	509.3	11.95	4293.7	0.619
	E-Delta-VEGA-TNO	2045	19	27.101	811.5	-	3.9	-	631.2	12.9	4105.5	0.592
	E-Delta-VEGA-TNO	2045	18	27.083	810.1	-	3.31	-	770.4	13.99	3898.6	0.562

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Orcus	E-Delta-VEGA-TNO	2045	17	27.067	809.1	-	2.75	-	930.2	15.23	3670.9	0.529
	E-Delta-VEGA-TNO	2045	16	27.053	808.8	-	2.25	-	1114. ₅	16.67	3420.9	0.493
	E-Delta-VEGA-TNO	2045	15	27.041	809.6	-	1.8	-	1329. ₁	18.36	3146.1	0.453
	E-Delta-VEGA-TNO	2045	14	26.847	689.9	-	1.7	-	1895. ₉	20.56	2640.0	0.379
	E-Delta-VEGA-TNO	2045	13	27.189	728.3	-	1.7	-	2883. ₆	23.04	1692.9	0.244
	E-Delta-VEGA-TNO	2045	12	27.561	740.6	-	1.7	-	4652. ₃	26.08	648.8	0.094
	E-Delta-VEGA-TNO	2045	11.8398 63	27.627	742.6	-	1.7	-	5000. ₈	26.63	505.7	0.073
	E-PJGA-TNO	7/29/2035	10	102.8	-	-	1.7	-	4218. ₅	26.74	394.8	0.183
	E-PJGA-TNO	7/29/2035	11	105.71	-	-	1.7	-	2646. ₄	23.49	763.7	0.370
	E-PJGA-TNO	7/29/2035	12	117.35	-	-	1.7	-	1547	20.67	998.1	0.569
	E-PJGA-TNO	7/29/2035	13	131.59	-	-	1.7	-	833.5	18.53	1076.4	0.741
	E-PJGA-TNO	7/29/2035	14	136.03	-	-	1.89	-	457.4	16.84	1165.9	0.850
	E-PJGA-TNO	7/29/2035	15	128.3	-	-	2.4	-	173.7	15.42	1424.5	0.940

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Orcus	E-PJGA-TNO	7/29/2035	16	122.58	-	-	3.01	-	0	14.2	1634.1	1.000
	E-PJGA-TNO	7/29/2035	17	118.31	-	-	3.72	-	0	13.14	1731.4	1.000
	E-PJGA-TNO	7/29/2035	18	115.1	-	-	4.5	-	0	12.21	1809.7	1.000
	E-PJGA-TNO	7/29/2035	19	112.66	-	-	5.32	-	0	11.37	1872.2	1.000
	E-PJGA-TNO	7/29/2035	20	110.79	-	-	6.16	-	0	10.62	1922.0	1.000
	E-PJGA-TNO	7/28/2036	10	190.96	-	-	2.43	-	0	23.73	805.0	1.000
	E-PJGA-TNO	7/28/2036	11	170.16	-	-	3.3	-	0	21.26	945.9	1.000
	E-PJGA-TNO	7/28/2036	12	156.08	-	-	4.33	-	0	19.19	1087.3	1.000
	E-PJGA-TNO	7/28/2036	13	146.29	-	-	5.5	-	0	17.44	1212.8	1.000
	E-PJGA-TNO	7/28/2036	14	139.31	-	-	6.78	-	0	15.93	1318.1	1.000
	E-PJGA-TNO	7/28/2036	15	132.82	-	-	8.13	-	0	14.62	1429.2	1.000
	E-PJGA-TNO	7/28/2036	16	127.47	-	-	9.51	-	0	13.49	1531.4	1.000
	E-PJGA-TNO	7/28/2036	17	125.22	-	-	10.9	-	0	12.48	1577.5	1.000
	E-PJGA-TNO	7/28/2036	18	123.35	-	-	12.28	-	0	11.59	1617.3	1.000

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Orcus	E-PJGA-TNO	7/28/2036	19	121.85	-	-	13.61	-	0	10.8	1650.3	1.000
	E-PJGA-TNO	7/28/2036	20	120.67	-	-	14.9	-	0	10.09	1676.7	1.000
	E-PJGA-TNO	7/29/2057	11	96.314	-	-	1.7	-	4080. ₄	23.96	464.4	0.196
	E-PJGA-TNO	7/29/2057	12	95.375	-	-	1.7	-	2053. ₇	20.93	1124.2	0.468
	E-PJGA-TNO	7/29/2057	13	108.69	-	-	1.7	-	1260. ₅	18.55	1254.1	0.633
	E-PJGA-TNO	7/29/2057	14	121.24	-	-	1.7	-	808.8	16.74	1244.6	0.748
	E-PJGA-TNO	7/29/2057	15	127.74	-	-	1.83	-	509.4	15.27	1272.5	0.834
	E-PJGA-TNO	7/29/2057	16	121.36	-	-	2.35	-	281.2	14.02	1503.2	0.905
	E-PJGA-TNO	7/29/2057	17	116.55	-	-	2.94	-	64.9	12.93	1733.4	0.977
	E-PJGA-TNO	7/29/2057	18	112.91	-	-	3.59	-	0	11.99	1865.7	1.000
	E-PJGA-TNO	7/29/2057	19	110.12	-	-	4.35	-	0	11.16	1940.2	1.000
	E-PJGA-TNO	7/29/2057	20	107.97	-	-	5.15	-	0	10.41	2000.0	1.000
	E-PJGA-TNO	7/28/2048	10	182.04	-	-	1.92	-	247.5	23.57	784.5	0.916

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Oeris	E-PJGA-TNO	7/28/2048	11	161.43	-	-	2.64	-	0	21.06	1028.5	1.000
	E-PJGA-TNO	7/28/2048	12	147.58	-	-	3.57	-	0	18.99	1194.8	1.000
	E-PJGA-TNO	7/28/2048	13	138.04	-	-	4.66	-	0	17.24	1338.7	1.000
	E-PJGA-TNO	7/28/2048	14	129.36	-	-	5.89	-	0	15.74	1494.1	1.000
	E-PJGA-TNO	7/28/2048	15	126.53	-	-	7.24	-	0	14.43	1550.5	1.000
	E-PJGA-TNO	7/28/2048	16	120.95	-	-	8.64	-	0	13.29	1670.4	1.000
	E-PJGA-TNO	7/28/2048	17	118.05	-	-	10.09	-	0	12.29	1737.6	1.000
	E-PJGA-TNO	7/28/2048	18	118.19	-	-	11.29	-	138.5	11.39	1651.2	0.952
	E-PJGA-TNO	7/28/2048	19	119.22	-	-	12.35	-	335.1	10.59	1517.8	0.888
	E-PJGA-TNO	7/28/2048	20	119.99	-	-	13.4	-	500.1	9.88	1415.7	0.837
High Thrust Quaoar	E-JGA-TNO	12/25/2016	13.97	122.49	-	-	16.44	-	-	13.93	1636.1	1.000
	E-JGA-TNO	11/22/2027	13.57	101.79	-	-	5.03	-	-	15.09	2185.7	1.000
	E-JGA-TNO	12/22/2028	13.57	111.54	-	-	12.1	-	-	14.13	1901.8	1.000
	E-JGA-TNO	1/22/2030	13.57	142.22	-	-	24.65	-	-	13.34	1272.4	1.000

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Quaoar	E-JGA-TNO	12/20/2040	13.58	105.89	-	-	9.2	-	-	14.39	2060.4	1.000
	E-JGA-TNO	11/21/2027	25.1	90.51	-	-	17.64	91	-	6.5	2580.0	1.000
	E-JGA-TNO	11/21/2027	22.3	91.08	-	-	15.5	98.1	-	7.69	2558.2	1.000
	E-JGA-TNO	11/21/2027	20.9	91.51	-	-	14.19	548	-	8.42	2541.9	1.000
	E-JGA-TNO	11/21/2027	19.6	92.11	-	-	12.7	1540	-	9.26	2519.3	1.000
	E-JGA-TNO	11/21/2027	18.2	93	-	-	11.07	4480	-	10.24	2486.3	1.000
	E-JGA-TNO	11/21/2027	15.5	94.4	-	-	9.3	10500	-	11.38	2435.3	1.000
Low Thrust Quaoar	E-JGA-TNO	4/1/2025-10/31/2025	24.6	50.3	-	-	19.4	119	-	6.72	4795.1	1.000
	E-JGA-TNO	4/1/2025-10/31/2026	23	52.1	-	-	17.1	127	-	7.89	4660.8	1.000
	E-JGA-TNO	4/1/2025-10/31/2027	18	53.5	-	-	12	2430	-	10.5	4559.1	1.000
	E-JGA-TNO	4/1/2025-10/31/2028	16	55.1	-	-	9.03	14700	-	12.3	4445.7	1.000
	E-JGA-TNO	4/1/2025-10/31/2029	14	58.1	-	-	6.06	42500	-	14.6	4241.2	1.000
	E-JGA-TNO	4/1/2025-10/31/2030	12	63.9	-	-	3.52	71100	-	17.7	3873.7	1.000

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Salacia	E-Delta-VEGA-JGA-TNO	1/20/2028	17	33	100.4	-	15.41	-	-	15.4	6092.4	0.965
	E-Delta-VEGA-JGA-TNO	1/15/2029	18.4	27.16	1230	611	6.26	-	-	13	4441.0	0.641
	E-Delta-VEGA-JGA-TNO	2/27/2042	18.2	28.18	1620	865	10.49	-	-	13.3	3773.6	0.553
High Thrust Sedna	E-JGA-TNO	5/6/2033	24.48	110.88	-	-	10.26	-	-	15.06	1919.5	1.000
	E-JGA-TNO	6/23/2046	24.48	147	-	-	21.01	-	-	14.15	1202.8	1.000
	E-Delta-VEGA-JGA-TNO	2/24/2042	25.5	28.94	960	0	2.63	-	-	15.97	4768.7	0.708
	E-Delta-VEGA-JGA-TNO	1/17/2041	29.2	27.25	620	0	2.2	-	-	14.88	5545.7	0.801
	E-PJGA-TNO	5/15-17/2045	12	157.8	-	-	1.89	-	3404.4	33.06	286.2	0.268
	E-PJGA-TNO	5/15-17/2045	13	158	-	-	2.19	-	2389.5	30.17	437.0	0.410
	E-PJGA-TNO	5/15-17/2045	14	154.8	-	-	2.57	-	1710.1	27.75	589.4	0.535
	E-PJGA-TNO	5/15-17/2045	15	153.4	-	-	3.01	-	1260.1	25.68	709.0	0.634
	E-PJGA-TNO	5/15-17/2045	16	150.9	-	-	3.55	-	955.4	23.88	815.8	0.709
	E-PJGA-TNO	5/20-21/2057	12	150.6	-	-	1.81	-	3674	32.61	274.2	0.238
	E-PJGA-TNO	5/20-21/2057	13	150.6	-	-	2.1	-	2721	29.75	413.9	0.359

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Sedna	E-PJGA-TNO	5/20-21/2057	14	150.6	-	-	2.42	-	1931	27.32	567.1	0.491
	E-PJGA-TNO	5/20-21/2057	15	150.7	-	-	2.76	-	1265	25.24	729.2	0.632
	E-PJGA-TNO	5/20-21/2057	16	150.7	-	-	3.12	-	695.9	23.44	898.8	0.779
	E-Delta-VEGA-JGA-TNO	2042	20	26	620	-	4.32	-	-	14.1	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	21	26	620	-	4.69	-	-	13.46	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	22	26	620	-	5.07	-	-	12.87	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	23	26	620	-	5.44	-	-	12.33	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	24	26	620	-	5.8	-	-	11.82	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	25	26	620	-	6.16	-	-	11.35	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	26	26	620	-	6.5	-	-	10.91	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	27	26	620	-	6.83	-	-	10.49	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	28	26	620	-	7.16	-	-	10.11	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	29	26	620	-	7.46	-	-	9.74	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	30	26	620	-	7.76	-	-	9.4	5658.0	0.801

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Sedna	E-Delta-VEGA-JGA-TNO	2042	31	26	620	-	8.04	-	-	9.08	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	32	26	620	-	8.3	-	-	8.78	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	33	26	620	-	8.55	-	-	8.49	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	34	26	620	-	8.79	-	-	8.22	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	35	26	620	-	9.01	-	-	7.97	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	36	26	620	-	9.22	-	-	7.72	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	37	26	620	-	9.42	-	-	7.49	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	38	26	620	-	9.6	-	-	7.27	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	39	26	620	-	9.77	-	-	7.07	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	40	26	620	-	9.93	-	-	6.87	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	41	26	620	-	10.07	-	-	6.68	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	42	26	620	-	10.21	-	-	6.5	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	43	26	620	-	10.33	-	-	6.34	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	44	26	620	-	10.45	-	-	6.17	5658.0	0.801

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Sedna	E-Delta-VEGA-JGA-TNO	2042	45	26	620	-	10.55	-	-	6.02	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	46	26	620	-	10.64	-	-	5.87	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	47	26	620	-	10.73	-	-	5.73	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	48	26	620	-	10.8	-	-	5.6	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	49	26	620	-	10.87	-	-	5.47	5658.0	0.801
	E-Delta-VEGA-JGA-TNO	2042	50	26	620	-	10.93	-	-	5.35	5658.0	0.801
	E-JGA-TNO	2032	20	110.5	-	-	1.7	-	-	19.74	1929.8	1.000
	E-JGA-TNO	2032	21	105.1	-	-	2.05	-	-	18.67	2083.8	1.000
	E-JGA-TNO	2032	22	101	-	-	2.45	-	-	17.69	2210.9	1.000
	E-JGA-TNO	2032	23	97.7	-	-	2.88	-	-	16.8	2319.9	1.000
	E-JGA-TNO	2032	24	95.2	-	-	3.35	-	-	15.98	2406.7	1.000
	E-JGA-TNO	2032	25	93.2	-	-	3.84	-	-	15.23	2478.9	1.000
	E-JGA-TNO	2032	26	91.6	-	-	4.35	-	-	14.53	2538.5	1.000
	E-JGA-TNO	2032	27	90.4	-	-	4.87	-	-	13.89	2584.2	1.000

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Sedna	E-JGA-TNO	2032	28	89.3	-	-	5.39	-	-	13.29	2627.0	1.000
	E-JGA-TNO	2032	29	88.5	-	-	5.91	-	-	12.73	2658.6	1.000
	E-JGA-TNO	2032	30	87.8	-	-	6.42	-	-	12.22	2686.7	1.000
	E-JGA-TNO	2032	31	87.3	-	-	6.93	-	-	11.73	2706.9	1.000
	E-JGA-TNO	2032	32	86.8	-	-	7.42	-	-	11.28	2727.3	1.000
	E-JGA-TNO	2032	33	86.4	-	-	7.89	-	-	10.86	2743.7	1.000
	E-JGA-TNO	2032	34	86.1	-	-	8.34	-	-	10.46	2756.2	1.000
	E-JGA-TNO	2032	35	85.8	-	-	8.77	-	-	10.08	2768.6	1.000
	E-JGA-TNO	2032	36	85.6	-	-	9.19	-	-	9.73	2777.0	1.000
	E-JGA-TNO	2032	37	85.4	-	-	9.58	-	-	9.4	2785.4	1.000
	E-JGA-TNO	2032	38	85.2	-	-	9.95	-	-	9.09	2793.8	1.000
	E-JGA-TNO	2032	39	85	-	-	10.3	-	-	8.79	2802.2	1.000
	E-JGA-TNO	2032	40	84.9	-	-	10.62	-	-	8.51	2806.4	1.000
	E-JGA-TNO	2032	41	84.8	-	-	10.93	-	-	8.24	2810.7	1.000

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Sedna	E-JGA-TNO	2032	42	84.7	-	-	11.22	-	-	7.99	2814.9	1.000
	E-JGA-TNO	2032	43	84.6	-	-	11.48	-	-	7.75	2819.2	1.000
	E-JGA-TNO	2032	44	84.5	-	-	11.73	-	-	7.53	2823.4	1.000
	E-JGA-TNO	2032	45	84.4	-	-	11.96	-	-	7.31	2827.7	1.000
	E-JGA-TNO	2032	46	84.3	-	-	12.17	-	-	7.11	2832.0	1.000
	E-JGA-TNO	2032	47	84.3	-	-	12.37	-	-	6.91	2832.0	1.000
	E-JGA-TNO	2032	48	84.2	-	-	12.55	-	-	6.72	2836.2	1.000
	E-JGA-TNO	2032	49	84.1	-	-	12.71	-	-	6.55	2840.5	1.000
	E-JGA-TNO	2032	50	84.1	-	-	12.86	-	-	6.38	2840.5	1.000
	E-JGA-TNO	2044	20	112.9	-	-	1.4	-	-	19.19	1865.9	1.000
	E-JGA-TNO	2044	21	106.9	-	-	1.7	-	-	18.14	2030.9	1.000
	E-JGA-TNO	2044	22	102.3	-	-	2.04	-	-	17.19	2169.6	1.000
	E-JGA-TNO	2044	23	98.8	-	-	2.42	-	-	16.32	2282.8	1.000
	E-JGA-TNO	2044	24	96.1	-	-	2.82	-	-	15.52	2375.0	1.000

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Sedna	E-JGA-TNO	2044	25	93.9	-	-	3.25	-	-	14.79	2453.3	1.000
	E-JGA-TNO	2044	26	92.3	-	-	3.7	-	-	14.11	2512.2	1.000
	E-JGA-TNO	2044	27	91	-	-	4.16	-	-	13.48	2561.2	1.000
	E-JGA-TNO	2044	28	90	-	-	4.62	-	-	12.9	2599.7	1.000
	E-JGA-TNO	2044	29	89.3	-	-	5.09	-	-	12.36	2627.0	1.000
	E-JGA-TNO	2044	30	88.7	-	-	5.55	-	-	11.86	2650.7	1.000
	E-JGA-TNO	2044	31	88.2	-	-	6	-	-	11.39	2670.6	1.000
	E-JGA-TNO	2044	32	87.9	-	-	6.44	-	-	10.95	2682.7	1.000
	E-JGA-TNO	2044	33	87.6	-	-	6.86	-	-	10.53	2694.7	1.000
	E-JGA-TNO	2044	34	87.4	-	-	7.27	-	-	10.15	2702.8	1.000
	E-JGA-TNO	2044	35	87.3	-	-	7.66	-	-	9.78	2706.9	1.000
	E-JGA-TNO	2044	36	87.2	-	-	8.02	-	-	9.44	2711.0	1.000
	E-JGA-TNO	2044	37	87.2	-	-	8.37	-	-	9.12	2711.0	1.000
	E-JGA-TNO	2044	38	87.1	-	-	8.7	-	-	8.82	2715.0	1.000

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta-V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta-V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Sedna	E-JGA-TNO	2044	39	87.1	-	-	9.01	-	-	8.53	2715.0	1.000
	E-JGA-TNO	2044	40	87.1	-	-	9.3	-	-	8.26	2715.0	1.000
	E-JGA-TNO	2044	41	87.2	-	-	9.57	-	-	8	2711.0	1.000
	E-JGA-TNO	2044	42	87.2	-	-	9.82	-	-	7.75	2711.0	1.000
	E-JGA-TNO	2044	43	87.3	-	-	10.05	-	-	7.52	2706.9	1.000
	E-JGA-TNO	2044	44	87.3	-	-	10.27	-	-	7.3	2706.9	1.000
	E-JGA-TNO	2044	45	87.4	-	-	10.47	-	-	7.1	2702.8	1.000
	E-JGA-TNO	2044	46	87.5	-	-	10.65	-	-	6.9	2698.8	1.000
	E-JGA-TNO	2044	47	87.6	-	-	10.81	-	-	6.71	2694.7	1.000
	E-JGA-TNO	2044	48	87.7	-	-	10.96	-	-	6.53	2690.7	1.000
	E-JGA-TNO	2044	49	87.8	-	-	11.1	-	-	6.36	2686.7	1.000
	E-JGA-TNO	2044	50	87.9	-	-	11.22	-	-	6.19	2682.7	1.000
High Thrust Typhon	E-Delta-VEGA-JGA-TNO	5/12/2032	16	35.1	726.5	-	1.73	-	-	12.91	4705.7	0.771
High Thrust Varda	E-Delta-VEGA-JGA-TNO	1/8/2051	17	32	224	-	13.53	-	-	12.66	5924.4	0.924

Table 5.4 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Earth Delta- V (m/s)	Jupiter Pass Distance (radii)	Jupiter Radiation Dose (rads)	Delta- V at Jupiter (m/s)	Excess Speed at TNO (km/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Varuna	E- Delta- VEGA- JGA- TNO	7/13/2032	22.5	26.96	1380	842	7.87	-	-	9.42	4212.7	0.606

Table 5.5 List of orbiter trajectories previously researched by The University of Tennessee and used for the summary section of this paper.

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Jupiter Pass Distance (radii)	Excess Speed at TNO (km/s)	Δv to Orbit (m/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust 2002 AW197	E-JGA-TNO	7/14/2023	20	97.59	-	9.84	9.48	9075.0	-33.6	0.015
	E-JGA-TNO	7/14/2023	25	96.26	-	12.69	6.93	6531.9	131.8	0.060
	E-JGA-TNO	8/23/2036	20	105.77	-	14.87	8.78	8376.5	5.2	0.023
	E-JGA-TNO	8/23/2036	25	102.79	-	18.33	6.41	6014.0	172.8	0.076
	E-JGA-TNO	8/28/2048	20	100.97	-	11.83	8.71	8306.7	5.2	0.024
	E-JGA-TNO	8/27/2048	25	97.94	-	15.43	6.33	5934.4	191.5	0.079
High Thrust 2002 TC302	E-JGA-TNO	3/24/2020	20	96.41	-	10.19	9.15	8895.6	-27.4	0.019
	E-JGA-TNO	3/24/2020	25	98.3	-	11.25	6.74	6488.3	133.4	0.065
	E-JGA-TNO	5/6/2033	20	108.76	-	11.81	8.09	7836.6	33.0	0.034
	E-JGA-TNO	5/6/2033	25	107.35	-	12.98	5.9	5649.7	201.1	0.096
	E-JGA-TNO	5/12/2045	20	106.19	-	8.18	7.9	7646.8	42.9	0.037
	E-JGA-TNO	5/11/2045	25	104.49	-	9.4	5.74	5490.0	226.5	0.103

Table 5.5 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Jupiter Pass Distance (radii)	Excess Speed at TNO (km/s)	Δv to Orbit (m/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust 2003 AZ84	E-JGA-TNO	7/15/2023	20	102.77	-	21.91	8.77	8390.8	2.4	0.023
	E-JGA-TNO	7/15/2015	25	100.41	-	26.81	6.38	6007.5	178.2	0.077
	E-JGA-TNO	7/18/2035	20	96.58	-	19.59	8.41	8031.6	17.4	0.028
	E-JGA-TNO	7/18/2035	25	94.65	-	24.21	6.07	5698.8	228.3	0.089
	E-JGA-TNO	7/23/2047	20	90	-	16.96	8.02	7642.5	40.9	0.035
	E-JGA-TNO	7/22/2047	25	88.39	-	21.33	5.72	5350.3	299.5	0.104
High Thrust 2007 UK126	E-JGA-TNO	4/30/2021	20	87.87	-	15.97	8.24	7922.8	19.8	0.031
	E-JGA-TNO	4/30/2021	25	86.46	-	19.39	6.05	5737.5	247.3	0.090
	E-JGA-TNO	6/12/2034	20	94.31	-	23.53	7.41	7094.2	83.3	0.047
	E-JGA-TNO	6/12/2034	25	92.45	-	26.34	5.54	5229.2	302.4	0.112
	E-JGA-TNO	6/16/2046	20	89.53	-	16.68	7.54	7224.0	74.4	0.044
	E-JGA-TNO	6/16/2046	25	88.7	-	18.19	5.59	5279.0	309.7	0.110
High Thrust Huamea	E-JGA-TNO	9/25/2037	20	102.25	-	9.68	10.55	9700.7	-49.5	0.008
	E-JGA-TNO	9/25/2037	24	98.87	-	13.41	8.21	7383.8	59.7	0.032

Table 5.5 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Jupiter Pass Distance (radii)	Excess Speed at TNO (km/s)	Δv to Orbit (m/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Huya	E-JGA-TNO	11/26/2027	24	90.82	-	20.01	3.99	3791.5	600.9	0.217
	E-JGA-TNO	11/26/2027	22	91.2	-	19.04	4.67	4469.9	444.0	0.163
	E-JGA-TNO	11/26/2027	19	91.97	-	17.75	5.44	5238.6	302.9	0.117
	E-JGA-TNO	11/26/2027	17	92.74	-	15.59	6.66	6457.1	144.2	0.068
	E-JGA-TNO	12/26/2039	25	88.55	-	6.75	4.7	4499.8	454.6	0.161
	E-JGA-TNO	11/29/2039	24	87.24	-	6.8	5	4799.3	402.1	0.142
	E-JGA-TNO	11/24/2039	23	87.35	-	6.85	5.32	5118.7	342.7	0.124
	E-JGA-TNO	11/29/2039	22	87.42	-	6.9	5.68	5478.2	283.0	0.106
	E-JGA-TNO	11/29/2039	21	87.59	-	6.99	6.09	5887.7	222.9	0.088
	E-JGA-TNO	11/29/2039	20	87.8	-	7.1	6.55	6347.2	164.4	0.071
	E-JGA-TNO	11/29/2039	19	87.98	-	7.2	7.06	6856.7	109.2	0.056
	E-JGA-TNO	11/29/2039	18	88.44	-	7.35	7.64	7436.2	56.9	0.042
	E-JGA-TNO	11/26/2027	19	91.97	-	17.75	5.44	5238.6	302.9	0.117
	E-JGA-TNO	11/26/2027	24	90.82	-	20.01	3.99	3791.5	600.9	0.217

Table 5.5 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Jupiter Pass Distance (radii)	Excess Speed at TNO (km/s)	Δv to Orbit (m/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Huya	E-JGA-TNO	11/29/2039	20	87.8	-	7.1	6.55	6347.2	164.4	0.071
	E-JGA-TNO	11/29/2039	24	87.24	-	6.8	5	4799.3	402.1	0.142
Low Thrust Huya	E-LT-JGA-TNO	-	25	36	-	-	4.3	4100.7	1201.1	0.191
	E-LT-JGA-TNO	-	24	36	-	-	4.55	4350.1	1067.5	0.172
	E-LT-JGA-TNO	-	23	36	-	-	4.9	4699.4	897.5	0.148
	E-LT-JGA-TNO	-	22	36	-	-	5.2	4998.9	766.2	0.130
	E-LT-JGA-TNO	-	21	36	-	-	5.6	5398.3	609.7	0.110
	E-LT-JGA-TNO	-	20	36	-	-	6.05	5847.8	456.0	0.090
	E-LT-JGA-TNO	-	19	36	-	-	6.55	6347.2	309.3	0.071
High Thrust Ixion	E-JGA-TNO	11/22/2027	25	91.51	-	20.5	4.91	4552.0	425.7	0.148
	E-JGA-TNO	11/22/2027	24	91.51	-	20.28	5.23	4870.1	366.3	0.129
	E-JGA-TNO	11/22/2027	23	91.53	-	19.95	5.59	5228.3	306.3	0.110
	E-JGA-TNO	11/22/2027	22	91.57	-	19.5	5.99	5626.5	247.1	0.092
	E-JGA-TNO	11/22/2027	21	91.64	-	18.89	6.44	6074.7	189.0	0.075

Table 5.5 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Jupiter Pass Distance (radii)	Excess Speed at TNO (km/s)	Δv to Orbit (m/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Ixion	E-JGA-TNO	11/22/2027	20	91.75	-	18.14	6.95	6583.0	132.6	0.059
	E-JGA-TNO	11/22/2027	19	91.91	-	17.21	7.53	7161.4	78.8	0.045
	E-JGA-TNO	11/25/2039	25	89.82	-	13.69	4.48	4124.9	528.3	0.177
	E-JGA-TNO	11/25/2039	24	89.57	-	13.73	4.76	4403.0	468.4	0.157
	E-JGA-TNO	11/25/2039	23	89.41	-	13.61	5.08	4721.0	405.1	0.137
	E-JGA-TNO	11/25/2039	22	89.3	-	13.36	5.45	5089.0	338.8	0.117
	E-JGA-TNO	11/25/2039	21	89.26	-	12.95	5.87	5507.0	272.0	0.097
	E-JGA-TNO	11/25/2039	20	89.27	-	12.39	6.35	5985.1	205.6	0.078
	E-JGA-TNO	11/25/2039	19	89.36	-	11.67	6.9	6533.2	141.0	0.061
	E-JGA-TNO	11/25/2039	18	89.53	-	10.79	7.52	7151.4	80.7	0.045
	E-JGA-TNO	11/25/2039	17	89.84	-	9.74	8.23	7859.7	25.2	0.031
	E-JGA-TNO	11/22/2027	20	91.75	-	18.14	6.95	6583.0	132.6	0.059
	E-JGA-TNO	11/22/2027	24	91.51	-	20.28	5.23	4870.1	366.3	0.129
	E-JGA-TNO	11/25/2039	20	89.27	-	12.39	6.35	5985.1	205.6	0.078

Table 5.5 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Jupiter Pass Distance (radii)	Excess Speed at TNO (km/s)	Δv to Orbit (m/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Ixion	E-JGA-TNO	11/25/2039	24	89.57	-	13.73	4.76	4403.0	468.4	0.157
High Thrust Makemake	E-JGA-TNO	8/17/2024	24	99.7	-	9.41	10.72	9954.0	-61.7	0.007
High Thrust Orcus	E-JGA-TNO	7/14/2023	25	93.9	-	11.56	7.83	7352.6	62.8	0.038
	E-JGA-TNO	7/18/2035	25	91.91	-	10.47	7.72	7243.1	72.0	0.041
	E-JGA-TNO	7/23/2047	25	88.48	-	9.52	7.52	7044.0	90.9	0.045
	E-JGA-TNO	7/23/2047	24	89.06	-	8.75	7.99	7511.9	50.8	0.035
High Thrust Quaoar	E-JGA-TNO	11/21/2027	20	91.82	-	13.29	8.97	8380.3	-6.3	0.021
	E-JGA-TNO	11/22/2027	24	90.61	-	17.03	6.9	6323.6	162.1	0.061
High Thrust Rhadamanthus	E-JGA-TNO	9/21/2025	25	94.31	-	18.75	6.53	6471.8	140.4	0.072
	E-JGA-TNO	9/21/2025	24	94.5	-	17.97	6.92	6861.7	103.1	0.060
	E-JGA-TNO	9/21/2025	23	94.74	-	17.1	7.35	7291.7	67.3	0.049
	E-JGA-TNO	9/26/2037	25	93.28	-	15.15	6.91	6851.7	105.1	0.060
	E-JGA-TNO	9/26/2037	24	93.46	-	14.35	7.33	7271.7	69.2	0.049
	E-JGA-TNO	9/26/2037	23	93.7	-	13.47	7.79	7731.7	35.3	0.039

Table 5.5 Continued

TNO and Mission Type	Mission Type	Launch Date	ToF (years)	C3 (km^2/s^2)	Space Burn (m/s)	Jupiter Pass Distance (radii)	Excess Speed at TNO (km/s)	Δv to Orbit (m/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
High Thrust Salacia	E-JGA-TNO	2/21/2031	25	88.77	-	13.76	7.32	6868.2	107.4	0.049
	E-JGA-TNO	2/21/2031	24	89.19	-	13.07	7.75	7296.4	68.3	0.040
	E-JGA-TNO	2/26/2043	25	85.42	-	12.72	7.54	7087.2	88.7	0.044
	E-JGA-TNO	2/26/2043	24	85.79	-	11.97	7.99	7535.5	48.7	0.035
	E-JGA-TNO	3/3/2055	25	82.3	-	11.47	7.7	7246.6	74.9	0.041
	E-JGA-TNO	3/2/2055	24	82.54	-	10.72	8.16	7704.9	34.2	0.032
High Thrust Varuna	E-JGA-TNO	6/6/2022	25	88.17	-	14.22	6.96	6615.7	133.8	0.059
	E-JGA-TNO	6/6/2022	24	88.36	-	13.6	7.38	7034.7	91.9	0.048
	E-JGA-TNO	6/7/2022	23	88.58	-	12.8	7.83	7483.7	53.0	0.038
	E-JGA-TNO	6/12/2034	25	87.24	-	11.02	7.17	6825.2	113.1	0.053
	E-JGA-TNO	6/12/2034	24	87.42	-	10.38	7.6	7254.2	72.5	0.043
	E-JGA-TNO	6/13/2034	23	87.8	-	9.66	8.1	7753.1	32.1	0.033

Table 5.6 List of lander trajectories previously researched by The University of Tennessee and used for the summary section of this paper.

TNO	Mission Type	Launch Date	ToF (years)	C3 (km ² /s ²)	Space Burn (m/s)	Jupiter Pass Distance (radii)	Excess Speed at TNO (km/s)	Δv to Land (m/s)	Payload Mass (kg) (Delta-IV Heavy with Star-48)	Mission Mass Fraction
1996 TS66	E-Delta-VEGA-JGA-TNO	4/8/2043	46	29.254	978.8	11.07	2.87	2871.5	1661.6	0.2
Albion	E-Delta-VEGA-JGA-TNO	12/19/2039	34	26.937	528.2	2.41	5.24	5240.4	653.8	0.1
Borasisi	E-Delta-VEGA-JGA-TNO	10/22/2037	40	27.764	614	2.17	8.34	8340.3	-79.6	0.0
Chaos	E-Delta-VEGA-JGA-TNO	3/29/2031	36	28.047	656.9	13.51	4	4025.8	1145.4	0.2
Deucalion	E-Delta-VEGA-JGA-TNO	5/27/2032	40	28.788	651.2	13.42	3.91	3911.0	1194.8	0.2
Huya	E-Delta-VEGA-JGA-TNO	10/18/2037	30	28.041	679.2	13.45	4	4011.0	1145.3	0.2
Ixion	E-Delta-VEGA-JGA-TNO	10/17/2037	23	28.258	739.6	12.41	5.92	5945.5	384.6	0.1
Lempo	E-Delta-VEGA-JGA-TNO	1/18/2041	33	27.613	741.4	2.41	4	4004.1	1134.2	0.2
MakeMake	E-Delta-VEGA-JGA-TNO	7/15/2034	40	29.602	840.9	18.79	5	5136.0	606.5	0.1
Manwe	E-Delta-VEGA-JGA-TNO	8/7/2037	29	29.424	790	8.13	3.34	3341.3	1450.5	0.2
Pluto	E-Delta-VEGA-JGA-TNO	12/6/2037	29	28.346	768.8	6.71	5.9	6229.8	307.4	0.1
Rhadmanthus	E-Delta-VEGA-JGA-TNO	4/8/2034	43	28.88	957	1.83	5.37	5370.6	518.1	0.1

VITA

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