AN ALTERNATIVE DUAL-LAUNCH ARCHITECTURE FOR A CREWED
ASTEROID MISSION

A Thesis

Presented to

the Faculty of California Polytechnic State University

San Luis Obispo

In Partial Fulfillment

of the Requirements for the Degree

Master of Science in Aerospace Engineering

by

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September 2012

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Abstract

An Alternative Dual-Launch Architecture for a Crewed Asteroid Mission

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This thesis is a feasibility study for a crewed mission to a Near Earth Asteroid (NEA). An alternate dual-launch architecture is proposed and analyzed against a more established architecture. Instead of a rendezvous in a low-Earth parking orbit, the new architecture performs the rendezvous while the two spacecraft are on an Earth-escape trajectory to the destination NEA. After selecting a target asteroid, 2000 SG344, each architecture will have its best mission compared to the best mission of the other architecture.

Using the new architecture, a mission is created to the chosen NEA, 2000 SG344. A back-up Orion MPCV and a Habitation Module are launched first on a cargo configuration SLS. A crew of two astronauts is launched two hours later in the primary Orion MPCV by a crewed configuration SLS. Both of these launches are on an Earth-escape trajectory and begin rendezvous after two full days in outer space. The completed spacecraft journeys the rest of the trip to the NEA. For a period of eight days, the spacecraft remains in a tight control sphere near the asteroid by using a control algorithm and the rendezvous thrusters. The astronauts have this period to perform their EVAs and accomplish their mission objectives at the NEA. The spacecraft then departs the NEA and returns to Earth. The entire mission is 134 days and requires 2.054 km/s of Δv maneuvers to complete.

An analysis of multiple Lambert's methods is also done due to their extensive use in this thesis. Many of the most popular Lambert algorithms are compared by evaluating each on its accuracy, speed, and singularities. The best Lambert

method to use for the orbital analysis in this paper is Battin's method because it is accurate, quick, and robust for all cases that will be observed.

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Chapter 1

Introduction

1.1 Previous Studies

A crewed asteroid mission is a popular topic in the aerospace community. Many papers are similar to this paper, but the new architecture is not evaluated for a crewed mission to a NEA.

1.1.1 Asteroid Finders

Although it is not the official name of any of these, many papers perform a brute force analysis that provides a list of potential asteroid candidates for humans to visit. The goal of these papers is to provide a list of likely NEAs for mission designers to choose among. The output list often provides data for a sample mission including the asteroid name and classification, size, round trip duration, total mission Δv , and the launch date. From the information provided in these papers, mission planners are supposed to choose an asteroid to design a mission. This thesis uses an asteroid finder to help find and decide upon an

asteroid, then it designs a mission to this chosen NEA.

1.1.2 Plymouth Rock

In recent years, Lockheed Martin performed a study, entitled Plymouth Rock[1], that utilized a dual-launch, dual-Orion concept for a human mission to a NEA. Two Orion capsules is an excellent idea that eliminates the need to design a newer, complex vehicle for this specific mission.

In order to prevent cabin fever and allow for proper health and hygiene, NASA specifies a minimum amount of volume per crewmember for various mission lengths. Two Orion Crew Modules do not provide enough space for the mission length that was mentioned. Even with the smallest possible crew, two Orion Crew Modules only allow for a mission duration of 60 days. This is not even enough time for an outbound trip. Therefore, there must be an additional module to allow for longer missions.

The Plymouth Rock study chose to visit asteroid 2008 EA9. This asteroid is likely $\leq 20\text{m}$ and it requires more Δv than other, larger NEAs. A different asteroid candidate is chosen for a mission design in this thesis.

The study also uses the low-Earth parking orbit rendezvous mission architecture. The architecture proposed in this thesis has rendezvous during an Earth-escape trajectory.

1.1.3 NASA/JPL

NASA and JPL have chosen asteroid 1999 AO10 as their destination for human asteroid missions[2]. The main appeal for this asteroid is that it is large and

it has a launch window around the year 2025. However, 1999 AO10 is a NEA that is difficult to reach, just like 2008 EA9 from the Plymouth Rock study, without a longer mission. Because of the difficulty in reaching 1999 AO10, a different asteroid candidate is chosen for a mission design for this thesis.

Like the Plymouth Rock study, the mission to 1999 AO10 is described using the low-Earth parking orbit rendezvous mission architecture[3]. The new architecture being proposed will be compared to this other architecture, but with a different target asteroid.

1.2 About NEAs

Near Earth Asteroids (NEAs) are small bodies of rock and other materials that orbit the sun. There are many thousands of NEAs orbiting the sun and new NEAs are being discovered and catalogued on a daily basis. To be classified as a NEA, the asteroid orbit must come closer to Earth than any other planet[4]. A flyby of another planet during transfer to a NEA is unlikely for a human mission.

1.2.1 NEA Size

Most NEAs are smaller than a typical office building, but the largest ones are about 10 km in mean diameter. The smallest known NEAs are about the size of a large car. Their small size makes them difficult to accurately photograph at large distances[1]. However, the diameter of a NEA can be estimated as a function of its absolute magnitude and albedo. Because the albedo of every asteroid is not known, a range of values is assumed to give an interval of likely asteroid diameters[5]. Different sources yield a different range of diameters for a given

NEA, but this is due to a different range of albedos that are assumed.

Because NEAs are so small, it should be assumed that most NEAs will not provide enough surface gravity to “land” an object on the surface. This would be different for the largest of NEAs, including 433 Eros, but these largest NEAs are not on the list of possible candidates being considered in this project. Therefore, a landing on a NEA is not possible in a typical planetary sense. Instead, it would be more feasible to “attach” to the NEA or simply fly in formation with the NEA. If the craft remains in formation flight, humans will need to EVA to reach the NEA.

1.2.2 NEA Spin Rate

The spin rates of most NEAs are unknown, but those that are known are a few minutes per revolution[1]. This presents a challenge for any attachments to the NEA. In order to attach to the NEA, the spacecraft must adjust itself to match the spin rate in an orbit. To conserve Δv , it would be wise to try to attach to the axis that the NEA spins around or remain in formation flight and use a propulsive system that astronauts can use by themselves.

Chapter 2

Mission Vehicles

New vehicles for human space travel will not be designed for this thesis. No new launch vehicles will be designed either. The two mission architectures will be compared using vehicles that are currently in existence or will be in existence during the time of the mission.

2.1 Orion MPCV

The Orion Multi-Purpose Crew Vehicle (MPCV) is a recently designed spacecraft that will be used for crewed missions for NASA. It consists of three integrated parts that are joined together during launch[\[6\]](#).

The upper part of the Orion MPCV is the Launch Abort System. This is a tall nose needle that sits at the very top of the SLS launch vehicle. It is designed to be able to quickly and safely carry the crewed Orion spacecraft away if the launch vehicle experiences catastrophic failures. If the launch is successful, this part is jettisoned from the orbiting spacecraft.

The inner part of the Orion MPCV is the Crew Module. This is the part where the crew will spend all of its time in outer space unless the Orion docks with another vehicle. It is able to carry up to 4 people on a mission, but more crew means shorter duration missions due to the lower volume per person and the increased rate of consumption for air, food, and water. If a mission does not involve an EVA or rendezvous, the crew will spend the entire mission in the crew capsule.

The bottom part of the Orion MPCV is the Service Module. This provides the propulsion, electrical power, and fluid storage for the Orion. This part of the Orion stays attached throughout most of the mission in order to provide the necessary propulsion, power, air and water for the mission and its crew. Unlike the crew module, this part does not survive re-entry. This part of the Orion separates from the Crew Module and does not return to the surface of the Earth.

The three parts come together to form a highly versatile vehicle that is a starting point for NASA to place humans in outer space. An exploded view of the Orion MPCV and its spacecraft launch adapter is shown below in Fig. 2.1.

These parts come together to form a flexible vehicle that will allow for a new era of human exploration. A summary of some of the information on the Orion MPCV and its individual parts is shown below in Table 2.1.

2.1.1 Orion MPCV Rendezvous Abilities

The Orion MPCV was designed to have the ability to autonomously rendezvous with another spacecraft. Autonomous rendezvous is defined as needing no communication with the ground during the rendezvous and docking processes[8]. The Orion still requires a crew of two to rendezvous and dock, but a lot of tasks

Figure 2.1: An exploded view of the Orion MPCV with all of its parts.[\[7\]](#)

are controlled by onboard computers[\[9\]](#). A manual override option also exists in case of malfunctions.

2.1.2 Interim Cryogenic Propulsion Stage

For some missions, an Interim Cryogenic Propulsion Stage (ICPS) will be added to provide some extra Δv for the crewed Orion MPCV assembly. This will be able to administer an additional 3050 m/s to the mission once it is developed[\[10\]](#). This extra boost is tremendously helpful to augment the abilities of the Orion MPCV.

Table 2.1: Summary of the Orion MPCV and its parts.

Orion Part	Parameter	Value[6]
Orion MPCV	Typical Crew	2-4
	Crewed Mission Duration	21-210 days
	Total Possible Δv	4920 ft/s $\approx$ 1.5 km/s
	Mass to Orbit	50231 lbs $\approx$ 22785 kg
	Gross Liftoff Weight	69181 lbs $\approx$ 31380 kg
Launch Abort System	Gross Liftoff Weight	16125 lbs $\approx$ 7315 kg
Crew Module	Habitable Volume	316 ft ³ $\approx$ 8.95 m ³
	Gross Liftoff Weight	21650 lbs $\approx$ 9820 kg
Service Module	Propellant Weight	17433 lbs $\approx$ 7908 kg
	Gross Liftoff Weight	27198 lbs $\approx$ 12337 kg

2.2 Space Launch System

The Space Launch System (SLS) is a next generation launch vehicle. The SLS will be the most powerful launch vehicle in human history by being able to bring the most mass to orbit[11]. Two configurations are being planned, the crewed and the cargo configuration[12].

2.2.1 Crewed Configuration

The crewed configuration will be ready first. It is designed to be human rated and have the Orion MPCV as its payload at the top. The first scheduled launch of the Block 1 will be a flight in 2017[11]. The initial crewed configuration will be able to lift 70 metric tons (1 mt = 1000 kg) to LEO[12]. After some added development, the Block 1A and 2 crewed configurations will be able to lift up to 105mt and 130mt to LEO[13].

The C3 graphs for the SLS are not available at this moment. Because the two vehicles have similar mass to LEO values, it will be assumed that the C3 values from the Saturn V directly match with the C3 values for the SLS crewed and cargo versions. If the C3 graphs of the SLS vehicles are published, then the values used in this thesis will need to be replaced. The graph of the C3 capabilities of the Saturn V is shown below in Fig. 2.2.

Figure 2.2: The C3 Capabilities of the Saturn V Rocket from the Payload Planner's Guide[14]. It is assumed in this paper that these values are the same as the C3 values of the SLS.

2.2.2 Cargo Configuration

A cargo configuration is being designed along with the crewed configuration of the SLS. This configuration will have its first launch after the crewed configuration SLS. The cargo configuration is able to bring more mass to orbit than previous launch vehicles. The fully evolved lift capability of the cargo configuration will be 130mt (130,000 kg) to LEO[12].

Chapter 3

Mission Architectures

In order for this mission to happen, there are a series of events that must take place in the following order:

- A human rated launch vehicle must bring astronauts into outer space in a spacecraft.
- A spacecraft must bring astronauts to a Near Earth Asteroid (NEA).
- A spacecraft must remain at the NEA for the astronauts to do some work.
- A spacecraft must return the astronauts to Earth and safely bring them to the surface.

These are unavoidable requirements to accomplish the objective of bringing a living human astronaut to a Near Earth Asteroid and back to Earth.

3.1 Single Launch Architecture

In order to visit a NEA, a single Orion MPCV as described in Sec. 2.1 will not be enough. It is an excellently designed vehicle that will be used for a long time, but by itself, its capabilities are limited. There is not enough volume for a crew of 2 for a mission longer than 30 days. The Orion MPCV can only provide 1.5 km/s of Δv , however missions will need at least 1.8 km/s. There may be room inside the SLS fairing at launch to store the Orion and another spacecraft to meet one of these criteria, but not both.

3.2 Two Launches

The multi-launch approach to a mission can greatly raise the cost, therefore, the mission shall have the minimum amount of launches necessary to successfully complete its objectives. A dual-launch mission allows for more creativity in mission design, but it relies heavily upon both of the launches to succeed. Any single launch failure can be catastrophic to the entire mission.

Another critical part of a multi-launch mission is the ability to rendezvous. A great deal of time and effort has been placed into optimizing the rendezvous process of the Orion MPCV. The Orion MPCV rendezvous process is largely controlled by a computer master timeline and can be done with only 2 crew members[9].

One launch will consist of an Orion MPCV in a crewed configuration SLS. The other launch will carry the necessary support spacecraft for the success of the mission. These two will rendezvous to form a completed spacecraft that has enough pressurized volume and fuel to accomplish the proposed mission in this

thesis.

3.3 Parking Orbit Rendezvous Architecture

The first of the two mission architectures being compared in this thesis is the parking orbit rendezvous. This is the architecture that is mentioned in the Plymouth Rock study[1] and others. This mission architecture is listed below:

- The two launches bring their cargo to identical parking orbits to rendezvous.
- The two launches rendezvous in this parking orbit and prepare for Earth Departure.
- The completed spacecraft will utilize an Earth Departure Stage booster to provide the necessary Δv to escape Earth orbit on a hyperbolic trajectory to the NEA. The Earth Departure Stage is jettisoned after it has been used.
- The completed spacecraft will match orbits with the NEA and remain in formation flight.
- The astronaut crew will perform their EVAs to the NEA while in formation flight.
- The spacecraft will depart the NEA and return to Earth.

3.4 New Architecture - C3 Mission

The other mission architecture being considered in this thesis is the C3 mission architecture. This mission architecture has a few differences from the previous

mission architecture. This architecture is expected to be a hybrid of a dual-launch approach for a mission and the C3 escape that many robotic spacecraft use for interplanetary mission. The C3 mission architecture is outlined below:

- The support vehicles launch first aboard the cargo configuration SLS. This launch goes on a course to meet the crewed launch in a few days time (even though the crewed launch hasn't occurred yet).
- After an hour or two, the crewed Orion MPCV will launch on course to the NEA. The crewed configuration SLS will carry the Orion MPCV on a hyperbolic Earth-escape trajectory to the NEA. This is the master trajectory that the support vehicles of the first launch will match en route to the NEA.
- The support vehicles will match the trajectory of the crewed Orion with a Δv maneuver. The two will join together to form the completed spacecraft.
- The completed spacecraft will continue its outbound journey to the NEA.
- The completed spacecraft will match orbits with the NEA and remain in formation flight.
- The astronaut crew will perform their EVAs to the NEA while in formation flight.
- The spacecraft will depart the NEA and return to Earth.

3.5 Architecture Differences

The difference between the two architectures is the timing of the rendezvous. The traditional architecture completes the rendezvous in a low-Earth parking or-

bit before departing for the NEA. The new architecture performs the rendezvous while on an Earth-escape trajectory to the NEA.

The best mission of the parking orbit rendezvous architecture will be compared against the best mission of the new C3 architecture. Each architecture will receive the same analysis to find its best mission. No missions from previous work will be used. The two architectures will be evaluated on:

- Total Mission Δv
- Total Mission Duration
- Total Mass to Orbit
- Astronaut Safety

A better mission will require less total Δv , less total duration, less mass to orbit, and greater astronaut safety.

Chapter 4

NEA Selection

4.1 NEA Trade Study

The full details of the NEA initial selection and the final trade study are mentioned in Appendix [B.1](#). Asteroids were rated on three categories: the orbital analysis, the year of the launch window, and the size of the asteroid. The analysis of these asteroids was performed with custom written code and not with custom software to reduce cost and maintain transparency and editability of the computer code. All of the computer calculations in this thesis was done with custom code instead of commercial software for these reasons. The candidate asteroids that were considered in the trade study were:

- 2009 BD
- 2000 SG344
- 1991 VG
- 2006 RH120

- 2008 UA202
- 2001 GP2
- 2001 QJ142
- 2008 HU4

4.1.1 Orbital Analysis

Each NEA received a score for the orbital analysis part of the trade study from its pork chop plots. Both the outbound and return pork chop plot were considered in this score. The scores were determined by the required Δv for a complete mission.

4.1.2 Launch Window

The best launch opportunities for each asteroid were found during the creation of the pork chop plots mentioned in the previous section. The year of these launch opportunities determined the score for the launch window section of the trade study. The NEA received a top score if the launch window fell between 2025 and 2030. A lower score was awarded by a launch window that was outside of this range.

4.1.3 Size of NEA

A larger estimated diameter range yielded a higher score in the trade study. These scores were normalized to ensure that the largest NEA being considered received a top score.

4.1.4 Weight

The weight for each category was selected according to its importance. The most important category was the orbital analysis. The other two categories were lower, but the launch window category received a higher score because it was determined that it would be better to have a mission to a smaller NEA than rush or wait for a larger NEA.

4.2 NEA Trade Study Results

The trade study results are shown below in Fig. 4.1. From the trade study, the best three options, in order, are:

1. 2000 SG344
2. 2006 RH120
3. 2001 QJ142

	<u>Orbital Analysis</u> Wt. = 5		<u>Launch Window</u> Wt. = 3		<u>Size of NEA</u> Wt. = 2			
NEA	Value	Score	Year	Score	Diameter	Score	Total	Rank
2009 BD	<i>n/a</i>	2	2034	3	3.7-17m	1	21	7
2000 SG344	<i>n/a</i>	4	2029	5	19-86m	4	43	1
1991 VG	<i>n/a</i>	3	2039	2	3.7-16m	1	23	5
2006 RH120	<i>n/a</i>	5	2028	5	2.2-10m	1	42	2
2008 UA202	<i>n/a</i>	2	2029	5	2.3-10m	1	27	4
2001 GP2	<i>n/a</i>	1	2019	1	7.3-33m	2	12	8
2001 QJ142	<i>n/a</i>	3	2024	3	36-161m	5	34	3
2008 HU4	<i>n/a</i>	1	2026	5	3.9-18m	1	22	6

Figure 4.1: NEA Selection Trade Study. A higher number represents heavier weighting or a better score.

These three NEAs are the best of the candidates that were found based upon the metrics outlined in the trade study. For the rest of this thesis, these three will be the asteroids of choice to visit. Mission analysis will be done for these NEAs in the upcoming sections.

A visual comparison of the three NEAs and Earth is shown below in Fig. 4.2. This plot shows the orbits of the Earth and the top three NEAs. The z -axis is scaled differently than the x -axis and y -axis which makes the NEA orbits look more inclined than they actually are. The inclination between Earth and 2001 QJ142 is only 3.1° . The year prefix will be occasionally dropped from the asteroid names as well.

Figure 4.2: The orbits of Earth and the top three NEAs. The z -axis is stretched and makes the NEA orbits look more inclined than they actually are.

Chapter 5

Mission Analysis

5.1 Asteroid 2000 SG344

The top asteroid in the trade study was asteroid 2000 SG344. This asteroid was the best because it offered low Δv requirements, a well-placed launch window, and it is a larger NEA. At the time of writing, this NEA does not have an official name beyond its current classification. The orbits of Earth and SG344 are shown below in Fig. [5.1](#).

Figure 5.1: The heliocentric orbits of Earth and asteroid SG344. Note that the vertical axis is stretched and gives the illusion of a greater inclination difference than what actually exists.

Other than the orbital data, there is a lot of important information that should be documented for an asteroid. The asteroid is fairly large compared to many other NEAs, but it is not large enough to be classified as a Potentially Hazardous Asteroid (PHA). It has a relatively low inclination with Earth, even if Fig. 5.1 above does not suggest this. Finally, the eccentricity is low enough to be used in the Clohessy-Wiltshire (C-W) equations, which is done in Sec. 6.5 to find the relative motion during the time the spacecraft is near the asteroid. Asteroid 2000 SG344 has the following parameters listed below in Table 5.1.

Table 5.1: Some important information of Asteroid 2000 SG344

Parameter	Value (units)[15]
Absolute Magnitude	24.8
Approximate Mean Diameter	19-86m
Launch Opportunity	2029
Inclination	0.108°
Eccentricity	0.0662
Mean Motion	1.013 deg/day

5.1.1 Pork Chop Plot

Pork Chop plots are valuable tools to find launch windows and Δv requirements for a mission. From the information on the pork chop plot, asteroid SG344 has a best launch window that may be somewhere around Day 6040. All transfers in this plot are Type 1 transfers. The pork chop plot for SG344 is shown below in Fig. 5.2 for the 2029 launch window.

Figure 5.2: Pork Chop Plot for Asteroid SG344. Day 5900 = Feb 25, 2029.

5.1.2 Return Pork Chop Plot

Analysis must be made for the return trip. A robotic mission would not need a return trip, but since humans are involved, they must be brought back to Earth. Round trips will always have higher Δv requirements than one-way trips. The return trip was considered when making the orbital analysis for the NEA selection process. The return pork chop plot for SG344 is shown below in Fig. 5.3. The spacecraft does not need a Δv at Earth arrival.

The main objective of the mission is to place humans at a NEA for as many days as possible. This exact parameter is likely something that will be set early on in mission planning. The longest possible stay duration is desired in order to get all of the EVA activities completed successfully. It is unknown what these activities will consist of, nor how long they will take, but it is assumed that they can be done in around a week.

In this section, “departure” refers to the moment that the spacecraft begins its outbound transfer to the NEA. If the two launches rendezvous in LEO, then the actual surface-to-orbit launch and LEO rendezvous must take place before this date.

In order to find the departure windows, sample mission scenarios were created and analyzed. The departure date, outbound and return transfer times, and stay duration were variables in this problem. A brute force approach was taken by using a series of nested loops to check all possible combinations at one-day intervals for a series of departure dates. For each combination, a stay of 6, 7, and 8 days was assumed. The mission generator gathered the Δv requirements and summed up the total mission duration.

This mission creation code was validated by comparing results with the NASA NHATS table. If the same asteroid target, departure date, transfer times, and stay duration were used, the same resulting velocity values were obtained from the NHATS table and the mission creation code[\[16\]](#).

If the mission had a total Δv that was below the maximum permitted amount and was quicker than 160 days, the mission parameters were stored. In the end, the code found the fastest possible missions, below the maximum Δv , for a NEA stay duration of 6, 7, and 8 days. The code also provides a range of departure

dates that have a mission below the maximum Δv and at most a set number of days. This is better explained with the data.

Parking Orbit Architecture

The first of the two mission architectures is the parking orbit one. The two launches rendezvous in LEO before departing to the NEA. The day number in the table represents the date at which the outbound transfer begins. A summary of the launch window data from this is shown below in Table 5.2. The stay duration at the NEA was varied from 6 to 8 days. Missions with a stay of 21 days were also created. A longer stay at the NEA will result in longer mission durations and a higher pressurized volume requirement for the spacecraft, but it allows for more objectives to be completed at the asteroid.

Table 5.2: SG344 Launch Window for missions that yield $\Delta v \leq 5$ km/s.

Stay at NEA	Fastest Mission Duration	Departure Window
6 days	146 days	July 15 – 20, 2029
7 days	147 days	July 14 – 21, 2029
8 days	147 days	July 16 – 18, 2029
21 days	159 days	July 9 – 14, 2029
8 days	160 days	June 26 – July 26, 2029

The table above lists the initial departure dates to check for mission planning. These provide insight for mission development to determine the ideal mission. For example, if the mission required a stay of 8 days at the NEA, the fastest possible mission (total $\Delta v \leq 5$ km/s) should depart Earth on July 16 – 18, 2029 for a 147 day mission.

A trend can be seen from the table above. As the duration of the stay at

the NEA increases by one day, the fastest possible mission usually increases by one day as well. If the duration does not increase, the launch window for the fastest mission greatly shrinks. The duration of the asteroid stay will be a major mission driver. For this thesis, the mission will be designed assuming a stay at the asteroid of 8 days.

These small launch windows represent an ideal mission. This may not happen due to schedule slips. Therefore, in order to better have a concept of the true launch window, the last line shows the launch window for missions with a stay duration of 8 days that can be done in a total of 160 days (still total $\Delta v \leq 5$ km/s). In other words, a 160-day or less mission exists, with an 8-day NEA stay duration, total $\Delta v \leq 5$ km/s for Earth departure dates June 26 – July 26, 2029. Different launch days have a different number of possible missions, but there has been no correlation seen between the apparent “center departure date” of the table information and the mean or median of the histogram. The distribution of possible missions for each day is shown below in Fig. 5.4.

C3 Mission Architecture

The same analysis was performed with a direct departure. In this case, the two payloads rendezvous while on escape from Earth-orbit. Because there is no Earth Departure Δv , the total fuel costs will be lower. Instead of the Earth Departure Δv , there is a required Δv for the support vehicles to match the trajectory of the crewed vehicles to the NEA. Some trial runnings using two hours between launches yielded values of around 0.05 - 0.07 km/s for this trajectory matching maneuver. Minimizing this value was far less of a mission driver than the arrival and departure burns at SG344. The algorithm focused on minimizing the arrival and departure Δv values.

Figure 5.4: Histogram of Launch Windows for SG344. More missions exist for certain days compared to others. This departure window has missions shown for June 26 – July 26, 2029.

More analysis was performed to calculate the Δv requirement for the support vehicles to match the trajectory of the crewed Orion. If the two launches were spaced two hours apart, the Δv requirement would be 0.0636 km/s for one mission scenario. For the same mission scenario, if the launches were spaced one day apart, the Δv requirement would be 0.5305 km/s. If the launches were spaced two days apart, the requirement increases to 0.7963 km/s. In order to minimize the required propellant, the two launches should be launched as close together in time as possible.

The launch window algorithm run was repeated like in the previous section with the parking orbit rendezvous. However, the total Δv cut-off was able to be

lowered from 5 km/s down to 2.06 km/s because of the numerous amounts of successful missions. The value of 2.06 km/s represents the sum of the rendezvous maneuver and the arrival and departure burns at the NEA. The stay durations of 6, 7, 8, and 21 days were kept as well as the goal of having the fastest possible mission. The launch window statistics are shown below in Table 5.3.

Table 5.3: SG344 Launch Window for missions with $\Delta v \leq 2.06$ km/s.

Stay at NEA	Mission Duration	Launch Window
6 days	133 days	July 21 – 28, 2029
7 days	134 days	July 19 – 29, 2029
8 days	134 days	July 21 – 27, 2029
21 days	143 days	July 17 – 23, 2029
8 days	160 days	June 19 – Aug 5, 2029

The launch window for the C3 mission is larger for this mission because the total mission Δv seems to change more slowly with different launch days. These launch windows also appear to have a slightly different “center departure date” than the parking orbit rendezvous table above. Compared to the results in Table 5.2, the values for the C3 mission yield much more favorable results.

The distribution of possible launches per day is shown below in Fig. 5.5. It can be clearly seen that there are significantly more mission possibilities per day and more days that have possible missions. This provides much greater flexibility for mission designers.

Figure 5.5: Histogram of Launch Windows for SG344. More missions exist for certain days compared to others. The range of departure dates for this mission extends between June 19 – Aug 5, 2029.

5.1.4 SG344 Mission Comparison

The best mission of each architecture is shown below in Table 5.4. The best mission was decided by choosing from the missions with the longest stay at the NEA and the shortest possible duration. Then, the mission with the lowest total Δv requirement was chosen. This table compares the two architectures following the criteria outlined before in this chapter. The stay duration at the asteroid will be set at 8 days to maximize the time allowed for astronauts to complete their EVAs.

Out of the two different architectures, the C3 mission to launch directly to

Table 5.4: Mission Architecture Comparison for mission to SG344.

Mission Parameter	Parking Orbit	C3 Mission
Launch Date	July 17, 2029	July 24, 2029
Outbound Transfer (days)	69	61
Stay Duration (days)	8	8
Return Transfer (days)	70	65
Support Vehicle Rendezvous (km/s)	0	0.0636
Earth Departure (km/s)	3.27	0
Stop at NEA (km/s)	0.7028	0.8524
Depart NEA (km/s)	1.0237	1.1379
Total Duration (days)	147	134
Total Δv (km/s)	4.998	2.054

the NEA is by far the best option. It allows for quicker missions with half the fuel costs and slightly later launch dates. Since there is no Earth departure stage and less fuel is needed, this also means less mass-to-orbit which drives down the cost of the mission. The C3 mission does require an extremely high assurance that the Earth escape rendezvous will work. If the likelihood of a successful rendezvous is not acceptable, then this mission architecture should not be used. The crewed spacecraft must be able to safely return the crew to Earth if the rendezvous is unsuccessful.

For actual mission design in this thesis, the C3 mission architecture will be chosen. The mission will be to asteroid 2000 SG344 and it will have the parameters shown below in Table 5.5. These were chosen to maximize the stay at the NEA while minimizing the total mission duration and Δv requirements.

The numbers here are based upon a preliminary Lambert analysis. A more thorough analysis will be performed as the mission is described.

Table 5.5: Mission Planning Parameters for C3 mission to SG344.

Mission Parameter	Value [Standard Units]
Launch Date	July 24, 2029
Outbound Transfer	61 days
Stay Duration	8 days
Return Transfer	65 days
Support Vehicle Rendezvous	0.0636 km/s
Stop at NEA	0.8524 km/s
Depart NEA	1.1379 km/s
Total Duration	134 days
Total Δv	2.054 km/s

5.2 Asteroid 2006 RH120

The second place asteroid in the trade study was asteroid 2006 RH120. This asteroid was placed high because it has the most favorable orbit for visiting and a good launch window, but it ended up second best because it is a very small asteroid, around the size of a large SUV. At the current time of writing, the NEA does not have a name beyond the normal classification. The orbits of Earth and RH120 are shown below in Fig. 5.6.

Figure 5.6: The orbits of Earth and RH120. Note that the vertical axis is stretched and gives the illusion of a greater inclination difference than what actually exists.

There is no publication that seems to suggest that a mission to this asteroid is being planned at this time. This NEA approaches Earth, but it is far too small to be considered a PHA. Like SG344, this NEA also has a low inclination relative to Earth, and the eccentricity is low enough to use the C-W equations. Asteroid 2006 RH120 has the following parameters listed below in Table 5.6.

Table 5.6: Some important information of Asteroid 2006 RH120

Parameter	Value (units)[15]
Absolute Magnitude	29.5
Approximate Mean Diameter	2.2-10m
Launch Opportunity	2028
Inclination	1.56°
Eccentricity	0.0205
Mean Motion	0.986 deg/day

5.2.1 Pork Chop Plot

As mentioned before, pork chop plots are valuable tools to find launch windows and Δv requirements for a mission. The pork chop plot suggests that an ideal mission to RH120 has a best launch window that may be somewhere around day 5670, July 10, 2028. The pork chop plot for RH120 is shown below in Fig. 5.7 for the 2028 launch opportunity.

Figure 5.7: Pork Chop Plot for Asteroid RH120. Day 5670 = July 10, 2028.

5.2.2 Return Pork Chop Plot

A return pork chop plot is created to ensure that the return Δv values do not make the mission unfeasible. Just like SG344, these values yield attractive results. The return pork chop plot for RH120 is shown below in Fig. 5.8.

Figure 5.8: Return Pork Chop Plot for Asteroid RH120. Day 5700 = August 9, 2028.

5.2.3 RH120 - Launch Windows

The same analysis from Sec. 5.1.3 was repeated for RH120. The stay durations at the NEA were assumed to be constant for any NEA, regardless of size or potential composition. The maximum duration for a mission was lowered to 140 days because this NEA is easier to reach. Lowering the total mission Δv cut-off by anymore than 0.25 km/s from the previous 5 km/s was not possible to do without severely cutting the size of the launch window.

Parking Orbit Architecture

The first of the two mission architectures is the parking orbit one. The two launches rendezvous in LEO before departing to the NEA. A summary of the launch window data from this is shown below in Table 5.7.

Table 5.7: RH120 Launch Window for missions that yield $\Delta v \leq 5$ km/s.

Stay at NEA	Fastest Mission Duration	Launch Window
6 days	128 days	July 1 – 11, 2028
7 days	129 days	July 2 – 9, 2028
8 days	130 days	July 4 – 7, 2028
8 days	140 days	June 12 – July 22, 2028

This table reads the same way as Table 5.2. If a stay duration of 8 days is needed, then the fastest possible mission (that keeps total mission $\Delta v \leq 5$ km/s) is 130 days and it must begin its outbound transfer on days July 4-7, 2028.

The distribution of possible missions for each day is shown below in Fig. 5.9. This shows the distribution of possible missions in the launch window from June 12 – July 22, 2028. Unlike the histograms from SG344, the missions to RH120 have a maximum duration of 140 days, and it has a larger window size as well.

Figure 5.9: Histogram of Launch Windows for RH120. More viable missions exist for certain days compared to others. The range of 5642-5682 corresponds to the dates June 12 – July 22, 2028.

C3 Mission Architecture

The other mission architecture is the C3 mission. In this case, the two launches rendezvous while on their escape trajectory to the asteroid. The value for the empty launch to match outbound orbits with the crewed vehicles is still assumed to be constant over all missions as before.

The C3 launch window algorithm was run again to find the launch windows for this asteroid. Again, the total Δv cut-off was able to be lowered from 5 km/s down to 2.04 km/s. The stay durations of 6, 7, and 8 days were kept as well as

the goal of having the fastest possible mission. This presents a fairer comparison between the two architectures. The launch window statistics are shown below in Table 5.8.

Table 5.8: RH120 Launch Window for missions with $\Delta v \leq 2.04$ km/s.

Stay at NEA	Mission Duration	Launch Window
6 days	114 days	July 7 – 19, 2028
7 days	115 days	July 7 – 18, 2028
8 days	116 days	July 8 – 16, 2028
8 days	140 days	May 29 – Aug 6, 2028

Again, the C3 mission architecture has wider launch windows. These launch windows also appear to have a slightly different center launch date than the parking orbit rendezvous table above. Compared to the results in Table 5.7, the values for the C3 mission yield much more favorable results. These results match with the results from the same analysis on SG344.

The distribution of possible launches per day is shown below in Fig. 5.10. Like SG344, there are significantly more mission possibilities per day and more days that have possible missions for greater flexibility for mission planning.

Figure 5.10: Histogram of Launch Windows for RH120. C3 Architecture. More missions exist for certain days compared to others.

5.2.4 RH120 Mission Comparison

The best mission of each architecture is shown below in Table 5.9. These missions were decided in the same way as Asteroid 2000 SG344. This table compares the two architectures following the criteria outlined before in this chapter. The stay duration at the asteroid will be set at 8 days to maximize the time allowed for astronauts to complete their EVAs.

Table 5.9: Mission Architecture Comparison for mission to RH120.

Mission Parameter	Parking Orbit	C3 Mission
Launch Date	July 6, 2028	July 12, 2028
Outbound Transfer (days)	58	51
Stay Duration (days)	8	8
Return Transfer (days)	64	57
Support Vehicle Rendezvous (km/s)	0	0.037
Earth Departure (km/s)	3.2058	0
Stop at NEA (km/s)	0.9047	1.007
Depart NEA (km/s)	0.8876	0.9842
Total Duration (days)	130	116
Total Δv (km/s)	4.9992	2.028

Out of the two different architectures, the C3 mission to launch directly to the NEA is by far the best option for the same reasons mentioned in the SG344 mission section, Sec. 5.1.4. While the asteroid 2006 RH120 has better overall orbital analysis results, it was still chosen as the second place finisher in the NEA selection trade study because it is much smaller.

If it is decided that this is the top NEA to visit for a crewed mission, then a best case mission is still presented below in Table 5.10. These were chosen to maximize the stay at the NEA while minimizing the total mission duration and Δv requirements. However, this will not be analyzed in this thesis.

Table 5.10: Mission Planning Parameters for C3 mission to RH120.

Mission Parameter	Value [Standard Units]
Destination NEA	2006 RH120
Launch Date	5672 [July 12, 2028]
Outbound Transfer	51 days
Stay Duration	8 days
Return Transfer	57 days
Empty Rendezvous	0.037 km/s
Stop at NEA	1.007 km/s
Depart NEA	0.9842 km/s
Total Duration	116 days
Total Δv	2.028 km/s

5.3 QJ142

The third place asteroid in the trade study was asteroid 2001 QJ142. This asteroid yields unattractive orbital analysis results compared to the first and second place NEA. On average, it is worse by about 2 km/s and it requires a longer travel time. It was still the definite third place finisher in the trade study, partially due to the large size of this NEA. At the current time of writing, the NEA does not have a name beyond the normal classification. The orbits of Earth and QJ142 are shown below in Fig. 5.11.

Figure 5.11: The orbits of Earth and QJ142. Note that the z -axis is stretched and gives the illusion of a greater inclination difference than what actually exists.

Because of the increased fuel requirements and mission duration, this NEA is a distant third compared to the first two. The analysis of this asteroid is not shown here. In order for the same mission algorithm to be used on this NEA, the Δv and mission duration cut-offs had to be relaxed to 7 km/s and 180 days for the parking orbit rendezvous mission and 3.55 km/s and 180 days for the C3 mission. If technology developed quickly enough to allow for a more difficult mission with a sooner launch window (2024), then it might be wise to visit this NEA instead.

Chapter 6

In-Depth Mission

6.1 Launch

The first phase of the actual mission is the launch phase. Unlike many other missions, this mission will have two launches. This phase of the mission has the following items of the architecture:

1. Support Vehicles launch on trajectory to meet the crewed Orion in 2 days and 2 hours
2. Crewed Orion launches on trajectory to the NEA 2 hours later
3. Each vehicle performs post-launch check-outs as soon as possible prior to the upcoming rendezvous process.

6.1.1 First Launch - Support Vehicles

This launch is scheduled to be sent into space on July 24, 2029, or day 6049 in units used in the code. This is the same day as the human launch. It is best to

have as little lead time as possible between launches to keep the rendezvous Δv minimal. If the launches were spaced two days apart, an additional 0.77 km/s of Δv would be needed for the rendezvous. If the two launches are two hours apart, the rendezvous requires only 0.0636 km/s of Δv . However, there needs to be some time allotted for a quick check-out of this first launch. If it turns out that one or more failures have occurred that will make the mission impossible, the human launch should not be undergone. If the human launch has already happened by the time the failures are discovered, then the human launch should immediately abort the mission and return to Earth.

The first launch of the mission will be the uncrewed support vehicle launch. This launch will have no humans onboard and therefore does not need to use a human rated launch vehicle. The payload of this launch must still be able to safely rendezvous and house humans later in the mission. In order to get the most mass to orbit for this long mission, the wisest choice would be the cargo configuration SLS. This launch vehicle will be able to carry the most mass to LEO during the time this mission will take place. Currently, the Block 2 SLS (130mt = 130,000 kg) is not scheduled to be ready until 2032, but the previous iteration (105mt to LEO) will be available by 2024[17].

This launch will begin rendezvous with the crewed launch on July 26, 2029. In other words, two days and two hours after its launch, this first launch performs its Δv maneuver to match trajectories with the crewed Orion MPCV. In order to reach this point, this launch will need a $v_\infty = 1.5265$ km/s. This orbit matching maneuver is expected to need 1949 lbs of propellant. Unlike the other maneuvers of the mission, this maneuver is only performed by the empty vehicles. Therefore, there is less concern about acceleration forces on the vehicles due to a human payload. However, great care must still be taken because this is a rendezvous

maneuver.

This launch needs a v_∞ of 1.5265 km/s. As mentioned before, the C3 capabilities of the cargo configuration SLS are unknown at this time. Because of similar mass to LEO values, the C3 data from the Saturn V is assumed to be identical to the cargo configuration SLS. The total payload mass of this launch must remain under around 97500 lbs. The masses of the extra Orion MPCV parts (Crew Module and Service Module) are known, and, when subtracted, leave 48652 lbs left for the rest of the launch.

As mentioned in Sec. 3.1, a single Orion MPCV will not suffice for the entire human mission to a NEA. This launch must augment the abilities by bringing more volume, consumables, propellant, and flexibility to outer space. This will be accomplished by a Habitation Module attached to a second Orion MPCV inside the payload fairing of a cargo configuration SLS.

Augmenting Flexibility - A Second Orion

A great way to supplement the capabilities of the single Orion MPCV would be to use a second Orion MPCV. This would quickly double the amount of volume and consumables that could be brought on the mission, without requiring vehicle development, qualification testing, or astronaut training for a new vehicle. This also provides the flexibility of having an extra volume that remains pressurized during EVAs. If there was only a single Crew Module, the entire crew volume must be depressurized for an EVA[18]. A second Orion also provides a lot of redundancy for the mission. If something on one Orion failed, the backup on the other Orion should be able to successfully do the job. Redundancy on this mission is highly important because it is unlikely that any rescue will be possible.

Augmenting Volume - The Habitation Module

For this mission, two Orion Crew Modules will not provide enough pressurized volume for a two person crew. A mission of 134 days with two crew members needs 49.76m^3 of pressurized volume. Since each of the two Crew Modules provides 19.55m^3 of pressurized volume, the Habitation Module (HM) must provide the remaining 10.66m^3 . The shape will most likely be cylindrical and an ISS node can be used as a starting point for design. This shape would also fit well inside the payload fairing of the cargo configuration SLS. If the inner volume was a 2m diameter and 4m long, this would provide more than enough pressurized volume.

In order to estimate the actual mass of the module, a few methods can be employed. NASA specifies an estimate on the “Habitation + Consumables Mass” for missions with varying durations and crew sizes. According to this, there should be approximately 25 metric tons of Habitation + Consumables Mass for the mission described here (this contains margin). After taking into account the two Orion Crew Modules, this yields a Habitation Module mass of around 12000 lbs. Another possible method to estimate the module mass is to carry over the ratio of pressurized volume:total mass. Since the HM must provide 10.66m^3 , it would weigh approximately 12000 lbs. This second method is slightly inaccurate because the HM doesn’t need as much equipment and technologies as the Crew Modules, such as re-entry shielding. Because these two primitive estimating methods provide similar results, a total of 12000 lbs is assumed to be the weight for mission calculations.

During the mission, the best place for the Habitation Module is between the two Orion capsules. This provides a continuous vehicle when all connecting hatches are opened. When all hatches are closed, it allows for either Orion to

be depressurized to support an EVA. This requires the Habitation Module to have docking hatches that are fully compatible with the Crew Module of the Orion MPCV. The completed dual-Orion spacecraft with the Habitation Module inbetween is shown below in Fig. 6.1.

Figure 6.1: The completed spacecraft with the two Orion MPCVs[6] and the Habitation Module connected inbetween.

This vehicle would need to flip to use the thruster from the other side. Either Orion thruster would be able to perform any maneuver with proper propellant management. The burn maneuvers are spaced far enough apart for a complete rotation of the spacecraft if necessary as well. This vehicle (without a Habitation Module) worked for the Plymouth Rock study[1].

Inside the payload fairing, from the bottom to top, will be the Orion Service Module, Orion Crew Module, and the Habitation Module. This configuration means that these vehicles do not need to separate, spin around, and attach. Some support structure will be needed to hold the Habitation Module in place and carry the launch loads. This support structure will likely consist of a cutout region that will also enclose the Crew Module. This aspect is similar to the Launch Abort System of the Orion on the crewed launch and it would provide some shielding the Crew Module from launch forces. This extra mass would be jettisoned once in outer space to reduce the spacecraft mass. It is assumed that

2000 lbs of support structure will be needed for launch. If the support structure is not jettisoned, then this Orion CM cannot be used for reentry or spacewalks via the EVA hatch because the Crew Module is still enclosed. This may cause other problems as well.

Augmenting Propellant

Adding a second Orion MPCV does not increase the amount of total Δv possible for the mission. There must be some additional fuel that is brought to orbit. A few options exist to provide for the additional Δv , but the best option is the one with the best fuel to total mass ratio.

A simple option would be to place the extra fuel into the Habitation Module. This will increase the size of the HM, but it may be the simplest and best option. The ability to transfer fuel between the vehicles is necessary, because the HM will not have a thruster of its own. This may require some extra planning for the Orion docking system as fuel control will now become important as well as the quantity of fuel.

Another option to increase the propellant would be to have an extra booster attached to the back of one of the Orion MPCVs. This could easily provide the necessary propulsion, but it must be able to survive until it is needed. The ICPS can provide 3050 m/s of Δv , but it is unknown if this will work for this purpose[10]. The ICPS requirements specify three engine ignitions, with the third for a separation. This suggests that precise control for a specific Δv might be impossible. This would require additional testing and development that may be too costly compared to having extra fuel for the Orion.

Depending on the mission architecture chosen, it may be necessary to have an

Earth Departure Stage booster. This would provide the necessary Δv to escape Earth's orbit and reach the NEA. For all viable mission scenarios, the Earth Departure burn is the most expensive burn. This Earth Departure Stage would need to provide a Δv of approximately 3.2 km/s for most missions. If possible, a lunar flyby will be done, however, simply reaching the moon would take over 3.0 km/s of Δv from the parking orbit. The flyby is also extremely sensitive due to its human payload and precise outbound trajectory. If the launch vehicles are powerful enough to place the launches en-route to the NEA, then an Earth Departure Stage would not be needed. The Earth Departure Stage should be placed underneath the Orion MPCV on the cargo configuration SLS launch.

The Habitation Module will also provide the additional fuel needed for this mission. If no additional fuel was brought on the mission, the two Orions and the HM would only be able to perform approximately 1.2 km/s of Δv . The rest of the allowable mass of the first launch will be propellant. This allows for 34652 lbs of propellant in the Habitation Module. With all of this propellant, there is just enough to complete the mission with 10% margin.

First Launch - Mass

The total amount of allowable mass in this first launch is 97500 lbs. With all of the major contributors added in, the mass breakdown for the launch is shown below in Fig. 6.2. As much propellant as possible is included in order to allow the most margin for Δv maneuvers.

Figure 6.2: The relative masses for the first launch.

6.1.2 Second Launch - Crewed Orion MPCV

The second launch will be a crewed configuration SLS launch of a Orion MPCV with two astronauts. This is the lowest number of astronauts possible for a mission on an Orion MPCV. A two-person crew means that a minimal amount of consumables and volume is required for the mission. This yields lower mass requirements for the overall mission.

It makes more sense to have the human launch as the second launch. This is the better option because the crew spends less time in space. That means that less consumables and less pressurized volume is required for the mission. Also, if there is a catastrophic failure with the other launch, human astronauts are on the surface of the Earth and not in outer space in need of rescue.

This launch will take place on July 24, 2029. This launch should take place after the launch of the support vehicles, but as soon as possible. This launch and trajectory serves as the master trajectory for the mission because this launch contains the human payload. The prior launch of empty vehicles will match this trajectory for rendezvous and docking to augment the capabilities of the

spacecraft.

This launch is designed to place its human payload on an outbound course to SG344. There are insufficient amounts of consumable air, water, and propellant to complete the mission without assistance from the support vehicles of the first launch. The rendezvous must be successful or the mission must be aborted in order for the human payload to survive. The Interim Cryogenic Propulsion Stage (ICPS) will be able to provide an additional 3050 m/s of Δv to a crewed Orion MPCV[10]. For this mission, the ICPS is needed as an abort rocket to safely return the astronauts to Earth. With the ICPS attached, this launch will look like Fig. 6.3 shown below.

Figure 6.3: Crewed Configuration SLS with all of its components.[19]

6.2 Rendezvous and Abort Trajectory

A single Orion MPCV does not have the capabilities to complete this mission on its own with one launch. Because of this, an extra launch must be done to augment the ability of the completed spacecraft. Once both vehicles are in space, they will rendezvous and dock to complete the larger spacecraft that can fully meet all of the requirements of a human mission. The extra vehicles must provide the extra volume, propellant, and flexibility for the mission to succeed. This is done with a second Orion MPCV and the Habitation Module.

The rendezvous procedure will begin on July 26, 2029. Following each launch, they will travel separately and begin their individual system check-outs. During this time, they are each headed to the rendezvous destination. The crewed Orion MPCV trajectory takes it directly through the rendezvous point and to the NEA. The support vehicle trajectory goes to the rendezvous point and performs a Δv maneuver to match orbits with the crewed Orion. This begins the rendezvous process where the crewed Orion is the target and the support vehicles are the chaser. The support vehicles must be the chaser because the crewed Orion will still have the ICPS attached.

6.2.1 Rendezvous Process

A 0.0636 km/s Δv is required for the empty vehicles to match the trajectory of the crewed vehicles. With a 10% margin, 1949 lbs of propellant would be needed to adjust the trajectory of the empty vehicles for rendezvous. This burn is done by the Service Module thrusters ($I_{sp} = 346s$) with either the Service Module propellant or the extra Habitation Module propellant. The rest of the maneuvering for the rendezvous should not take significant amounts of fuel

compared to the major burns of the mission.

6.2.2 Abort

The joining of the two spacecraft is difficult. There is a lot of room in space and many things can go wrong. If any of the following things go wrong, it will likely or certainly mean a mission failure. The list below is certainly not a comprehensive list, but it details some of the possible causes for a failure to rendezvous and dock.

- The support vehicles do not match orbits with the crewed Orion. This results in a complete miss of the two spacecraft.
- The seal on the docked spacecraft is not sufficient for the duration of the mission and air is leaked at too high of a rate.
- A failure in the support vehicle launch results in catastrophic damage to its vehicles.
- Correct attitude or positioning is not possible for rendezvous.
- Rendezvous systems aboard each spacecraft do not establish and undergo the proper rendezvous procedure.
- The two vehicles collide during rendezvous causing catastrophic damage.

Nearly all of these failures could result with the crewed Orion MPCV fully intact. The mission must then be aborted for the astronaut crew to return to Earth.

6.2.3 Abort Return Trajectory

Rendezvous begins after two full days in space. The rendezvous process is expected to take 2 days at the most to complete. The worst case is to have the astronaut crew as far away from Earth as possible with as few days as possible to return. The worst case situation would be an abort after 4 days of mission. This represents the furthest starting point and the fewest possible 16 days to return to Earth.

There are two options for the return trajectory to Earth. The spacecraft can transfer directly to Earth. The spacecraft can also perform a burn to maneuver itself to a lunar flyby that places the spacecraft on a free return trajectory to Earth.

The analysis for both of these was performed and the Δv requirements were plotted against each other. Initially, the analysis was done for an abort after 4 days of space travel. It is clear that doing a flyby of the Moon provides no benefit in Δv cost reduction and the added mission time is only a further hindrance. The process was repeated for aborts after 2 full days and 3 full days of mission, all yielded the same result. It is best to abort the mission directly back to Earth. The graph of Δv requirements is shown below in Fig. 6.4.

The abort trajectory will go directly back to Earth. If the 2 person mission is aborted after 4 full days of transfer, the crewed Orion with the ICPS has the capability to safely return within the 20 day window.

Figure 6.4: Abort Trajectory comparison between a transfer directly back to Earth or to a Lunar flyby. The direct transfer to Earth costs about the same and avoids an extra few days of Moon to Earth transfer.

6.3 Transfer to NEA

After completing the rendezvous process, the completed spacecraft will be on course to the asteroid. Because this is a human mission, it is necessary to keep the total mission duration short. Therefore, there are no flyby maneuvers around another planet. The Moon is not in position to save any Δv for the outbound transfer, and visiting another planet would tremendously increase the mission duration. The launch vehicles have enough power to place both launches in the directions and velocities they need.

The spacecraft will arrive at the NEA on September 23, 2029. This is a travel

time of 61 days since launch. Since the rendezvous process began after two days of travel and took 2 days to complete, the completed spacecraft will have traveled for 57 days. The transfer to the NEA is shown below in Fig. 6.5. The position data is shown in the heliocentric frame. The spacecraft (green line) begins at the Earth (blue line) and arrives at SG344 (black line).

Figure 6.5: Heliocentric position data for the all major nearby bodies during the transfer to SG344. The different axes are stretched differently. The inclination between the Earth orbit and SG344 is only 0.1° .

As the spacecraft flies to SG344, it will need to perform mid-course corrections as necessary. This will correct its trajectory from errors caused by perturbations, simulation accuracy, thruster capabilities, and more.

6.4 Arrival at the NEA

At the NEA, the spacecraft will perform the second major burn of the mission. The spacecraft will match velocities of the NEA as best as possible. This maneuver will take place on September 23, 2029.

The Δv required for the orbit matching is calculated at 0.8524 km/s. This shall be done by the Orion MPCV thrusters and is expected to require 34368 lbs of propellant (with a 10% margin). The spacecraft should be able to match relative velocities with the NEA to under 0.01 m/s based upon the maneuvering capabilities of the Orion capsule.

The spacecraft will aim for a destination that is 100m inside the orbit of the NEA. As the spacecraft approaches the desired location in the C-W frame, the spacecraft will be doing a lower-risk rendezvous process. There will be a desired target with a position and velocity. However, this target is simply a point in space and not an actual spacecraft or object. The risk of collision is lower, but the rendezvous process still requires precision and accuracy to minimize wasted propellant and any collisions. A lower offset may be chosen in the future when the orbit of SG344 and the rendezvous thruster capabilities of the completed spacecraft are better known.

6.5 Relative Motion at the NEA

The main purpose of the mission is to place humans at the NEA. Midway through the mission, at this destination, the spacecraft will spend several days in formation flight with the NEA. Unlike other parts of the mission, this analysis is done in the C-W frame. This frame is used to find the relative motion drift of

one body versus another through the Clohessy-Wiltshire Hill Equations.

The C-W frame is a moving frame that is attached to the target. It only works with circular or near circular orbits. In real applications, no orbit is circular for more than a split second. Positive x -direction is defined as the direction from the center of the orbit through the target body. Positive y -direction is the direction of motion of the target. Positive z -direction is the direction that completes the right-handed frame centered at the target. If the orbit was not circular, this coordinate frame would not consist of 3 independent directions unless the target was at periapsis or apoapsis. The relative motion calculations were tested against the sample case in Vallado[20] to validate their accuracy.

Eccentricities for the 8 asteroids are shown below in Table 6.1. Although the chosen asteroid is 2000 SG344, all eccentricities are presented to show that they are all similar to each other, and it is evident that the eccentricity of SG344 is not an outlier.

Table 6.1: Eccentricities of the 8 asteroids in the study.

Asteroid	Eccentricity[21]
2006 RH120	0.0245
1991 VG	0.0491
2009 BD	0.0515
2000 SG344	0.0669
2008 UA202	0.0686
2008 HU4	0.0733
2001 GP2	0.0739
2001 QJ142	0.0864

From the eccentricities mentioned above, it will be assumed that each of these target NEAs have an orbit that is circular enough for the Clohessy-Wiltshire

equations to hold. A hard cutoff could be imposed if the eccentricity exceeded 0.1, but none of these target NEAs reach this. For a stay that is expected to be only 8 days, the errors could be corrected through control algorithm.

6.5.1 Control Logic

The purpose of the formation flying control architecture is to keep the spacecraft in close proximity to the target NEA through the use of small rendezvous thrusters on the spacecraft. If the spacecraft drifts too far from the NEA, it will become more difficult for astronauts to travel to the NEA and back during their EVAs. A farther trip will take more time and more fuel from the EVA transfer equipment used by the astronauts. The spacecraft also cannot drift too close to the NEA because this increases the risk of collision and thruster contamination onto the asteroid or any equipment placed on the asteroid.

The control architecture for the spacecraft is designed to keep the spacecraft at its desired goal position, but is satisfied if the spacecraft remains close enough, in a region called the “goal sphere.” For the purposes of creating and testing the control architecture, the goal position was defined as $[-200, 0, 0]$ meters and the radius of the goal sphere is set at 50 meters in the C-W frame. The control architecture discretizes the stay into multiple blocks per day for the full stay duration. At each of the 400 blocks over the 20 day period, the algorithm checks if the spacecraft should perform a burn maneuver to better adjust the velocity of the spacecraft. All velocity changes are assumed to be instantaneous.

If the spacecraft exceeds a certain distance away from its goal position, a burn maneuver will be performed. The spacecraft thrusters will fire and apply their minimum Δv to the spacecraft. This burn is applied directly towards the goal

position of the spacecraft. Although simple in its design, this control method will require frequent burning as the spacecraft continuously drifts to opposite sides of the goal sphere. Left alone, this control algorithm would require many burns as the spacecraft reverses its direction many times. Lots of fuel would be used as well. A sample of this control architecture is shown below in Fig. 6.6.

Figure 6.6: The path of the spacecraft frequently exceeds the boundary of the Goal Sphere. Each correction burn overpowers the drift.

The overshoot distance could be reduced by minimizing the time between blocks, but that does not help reduce the amount of burns, nor does it reduce the amount of fuel used. The real problem is the excessive speed. In order to improve the control architecture, a second condition statement is added. If the spacecraft is inside the goal sphere, and it is traveling fast, it will burn to slow itself down. The spacecraft is considered to be “traveling fast” if a minimum Δv against the current velocity will slow down the spacecraft, but not stop it or change the direction of motion. For example, if the spacecraft thrusters could

apply a minimum 0.003 m/s burn, the control architecture would apply a burn to stop the spacecraft once it reached or exceeded 0.003 m/s by firing thrusters in the opposite direction. This “braking” is applied only inside the goal sphere in order to lengthen the amount of time the spacecraft spends inside.

A braking maneuver would not be applied if the spacecraft was traveling at slightly over half of the minimum Δv . For example, if the spacecraft was traveling at 0.002 m/s, it would not apply a burn to change the velocity to 0.001 m/s in the opposite direction. A braking maneuver such as this may end up pushing the spacecraft outside of the goal sphere too often. This would also increase the total amount of burns.

Once this secondary control step is applied, the overall results are vastly improved. The results clearly show that much more precise control of the spacecraft is possible. Because of the extra precision, the radius of the goal sphere was shrunk to 10m and the initial displacement of the spacecraft from the asteroid was reduced to 100m. As a trade off, the 20 day period had to be discretized into more time blocks due to the smaller radius of the goal sphere. The number of time blocks was increased from 400 to 2000 to prevent large drift outside of the goal sphere between checks. Despite being only one-fifth of the previous duration, one time block is still 14.4 minutes.

The amount of correction burns made by the spacecraft is reduced from 100-200 to 5-7. The total Δv also drops to 0.017 m/s from the former 0.47 m/s. The spacecraft spends significantly more time in the goal sphere. On some cases, the spacecraft does not escape the goal sphere during the 20 day period except for an initial correction. If it manages to drift out, the drift distance is much less due to the slower speeds.

This control architecture is not perfect. First of all, when the spacecraft is outside of the goal sphere, it should not always burn directly towards the goal. Doing this means you will miss the center of the goal sphere. In the end, you may be maneuvering around the edges of the goal sphere. This would outline a fullerene-type shape that consists of straight edges between correction burns. However, the simplicity and the impressive results of the current system do not need improvement and the fullerene shape orbit has not been seen for 20 days of relative motion calculation.

Secondly, when the spacecraft drifts outside of the goal sphere and performs a burn to fix positioning, this burn is often immediately followed by a braking burn. This is seen when there is a square-wave pattern on the velocity graph. The first burn is too strong and must be softened immediately by a subsequent braking burn against motion. A more fuel-efficient fix would be able to perform one burn to take the place of these two, but this could be a limitation from the spacecraft thrusters.

6.5.2 Testing and Simulations

The control architecture needs to be tested under varying conditions that might actually be experienced during the real mission. It is unlikely that the spacecraft will start perfectly at its goal position with no relative velocities. Position displacements will be small, because the Orion capsule has been designed for efficient rendezvous procedures. In addition, the displacements will eventually be negated by the control architecture's desire to keep the spacecraft within the goal sphere.

The more troublesome variant is the residual velocities that remain after the

spacecraft reaches its goal position. These can greatly vary the amount of Δv and number of burns required for accurate positioning. In order to test the robustness of the architecture, the spacecraft will start with small random velocities. These are created through the *rand* command in Matlab and scaled accordingly to the fine and coarse thruster capabilities of the Orion thrusters as used for ISS docking simulations[22].

Without the braking maneuver, it would take a lucky set of starting velocities to remain mostly in the goal sphere. Some initial velocity cases brought the spacecraft very far outside the goal sphere. When the braking maneuver is re-added into the control architecture, the spacecraft consistently remains inside or close to the goal sphere. A sample case of these positions and velocities is shown below in Fig. 6.7.

Figure 6.7: A sample case of random initial velocities that undergo the control architecture for 20 days of formation flying. The goal position is at $[-100, 0, 0]$ meters with a 10m goal sphere radius.

But testing one or two, or even 10 random cases is not sufficient. Therefore, a few Monte Carlo simulations were performed to find some worst case values. This is the true test of robustness of the control architecture. If it is apparent that for all the cases, the control architecture handles the situation, then the control architecture should be able to handle a real initial state like the ones in the simulation. The Monte Carlo simulation was performed 4 times. Each simulation consisted of 2000 random velocities that are used in the formation flying algorithm mentioned earlier. The statistical results for the four simulations are shown below in Fig. 6.8.

Figure 6.8: The four Monte Carlo simulations with their Δv statistics. Each simulation yields similar results, therefore it is probable that they encompass enough data.

From the simulations, a case that uses more than 0.1 m/s of fuel over 20 days is unlikely. Out of the 8000 cases, the spacecraft reaches a maximum distance of 119.03m away from the NEA. This value is obviously outside of the goal sphere, but the moment before and after must be closer to the goal position and the asteroid. If the spacecraft needed to be more tightly controlled and remain closer to the NEA at all times, then the initial positioning and the size of the goal sphere should be adjusted. This must also take the size and spin rate of the asteroid into account as well as the knowledge of the position of the asteroid.

The closest the two objects get is 86.19m. The closest approach distance must be at least large enough to cover the combined distances of:

- The maximum radius of the chosen asteroid (Asteroid 2000 SG344 has an estimated radius of 20m)
- The maximum distance between the center of the spacecraft and its furthest appendage
- The maximum required distance for equipment placed on the asteroid
- The maximum distance for the astronauts to work on the asteroid
- The maximum required distance for the EVA equipment to not interfere with the spacecraft nor the astronauts
- The maximum distance for astronauts to re-enter the spacecraft
- Any extra margin or factors of safety

The worst-case scenario would be for all of these items to line up directly along the vector joining the spacecraft and asteroid. Adequate clearance and positioning must be allotted for and maintained to prevent a collision.

One safe method to set a goal position would be to find the error ellipsoid for the NEA and remain at least 3-sigma away from the center at all times. This would define a distance magnitude to be at relative to the NEA, but not a vector. On the scale of this mission, whichever distance vector is chosen as the goal position relative to the NEA, it will not significantly change the Δv requirements for transfers or positioning, nor the communication delay.

A tighter goal sphere would require more frequent correction burns. A trade-off would need to be made to determine a final size. This would highly depend on the thrust capabilities that define a closest distance and a more accurate measure of the size of the asteroid. The Δv requirements for this phase of the mission

are nearly negligible compared against orbit transfers. It would also be simple to arbitrarily choose a radius based upon the distance magnitude to the NEA, like what was initially done in this thesis. A modification could be made mid-mission if there was a concern of collision or thruster contamination.

Another concern for the size of the goal sphere would be the frequency of the burns and its impact on astronauts. The spacecraft should not undergo a burn such as this when astronauts are in the process of entering or departing the spacecraft. However, this can be mitigated by planning EVAs around the burn schedules.

The calculations done in this section assumed a NEA stay of 20 days. But this is a maximum stay duration being considered. The actual mission design involves a stay duration of only around 8 days, therefore all appropriate data would need to be scaled accordingly. The number of burns and the total Δv would be reduced by a little more than half. However, because the mission impact of this relative motion Δv is negligible, half of negligible, is still negligible.

6.6 Tasks for the Astronauts

The spacecraft will remain in proximity with the NEA between the days of September 23, 2029 and October 1, 2029. This leaves 8 full days for work to be done.

This mission shall accomplish many tasks at the NEA, including the following:

- Study and refine rendezvous and formation flying procedures with small bodies.
- Place a commemorative symbol on the NEA, likely a flag and/or plaque.

- Collect one or more physical samples of the NEA.
- Place any equipment that requires specific placement on the NEA. This may also require Earth-based verification or re-placement.
- Augment any scientific data concerning the NEA such as photography, LIDAR, and other sensors.

This list is not intended to be a comprehensive list and more objectives could be added at a later time.

6.6.1 EVA to the NEA

At a distance of 100m, a tethered astronaut could not reliably jump out of the spacecraft and expect to reach the NEA. It is simply too small of a target to reach. Some directional course control must be possible for the astronaut to reach the NEA. One of the simplest solutions to reach the NEA is to re-use the Manned Maneuvering Units (MMU) that was previously used on space shuttle missions. An MMU stored 40 lbs of compressed gas with an $I_{sp} = 60s$. Depending on the mass of the crew and the cargo being brought along, the MMU could provide a Δv of up to around 135 ft/s. This also had the ability to recharge itself through the parent spacecraft, but in the past, the space shuttle could only recharge to a lower psi that reduced the max Δv of the MMU slightly. The MMU could provide life support for 6 hours of EVA[23].

An improved and re-designed MMU should be able to increase all of these capabilities, but the older design would be good enough to provide the propulsive ability to reach and return from the NEA. The NEA can be reached in a more fuel-efficient manner by accelerating to lower velocities and coasting for longer

periods of time. A journey of 100m would take approximately 120 seconds if the astronaut only accelerated to 3 ft/s. Depending on the mass of the cargo brought along, this would use around 5% of the available propellant inside the MMU.

The MMU may need some modifications in order to allow an astronaut to have the range of motion that is needed. However, this is more dependent on the plan for operations at the NEA and any complex astronaut tasks. A potential solution to this is to use a space scooter that an astronaut would ride back and forth between the spacecraft and the NEA like a motorcycle. At the NEA, the astronaut would use a smaller MMU or use some sort of tether from this point.

A big challenge that must be overcome would result from any rotation of the asteroid. Even a small rotation would make it difficult to perform labor on the asteroid. The rotation of the asteroid must be compensated for using almost constant propellant from the MMU. Since larger NEAs are more likely to have lower spin rates, this once again stresses the importance of visiting a larger target. The spin of a NEA essentially eliminates the possibility of using a tethered system because the tether would become tightly wrapped around the asteroid. It is unknown how much Δv will be required to match the spin of the NEA. This is highly dependent on its size, spin rate, and shape. These things will be determined prior to the mission in order to find the best solution. It will be likely that an MMU-type solution will be utilized.

It is possible that a long term attachment could be made to the NEA. This process may be similar to attaching a rock climber hook. Multiple anchors could be used to secure an object to the NEA. Once an initial attachment such as this is first made, a more solid platform could be constructed. The final constructed platform does not need to last beyond the stay duration of the current mission unless it is required by an instrument left on the NEA.

6.7 Departure and Return Trip from the NEA

The spacecraft will depart the asteroid on October 1, 2029. On this day, the spacecraft will perform a burn to place itself on trajectory to meet arrive at Earth in 65 days. This burn provides 1.1379 km/s of Δv and uses 33326 lbs of propellant. The spacecraft is scheduled to arrive back at Earth on December 5, 2029 to complete the 134 day mission. The return trajectory is shown below in Fig. 6.9.

Figure 6.9: Fully populated spacecraft return trajectory back to Earth. Like before, the z -axis is stretched again.

6.8 Re-entry

The spacecraft will re-enter Earth's atmosphere on December 5, 2029 after the 65 day return trip from the NEA. At this time, there will be frequent communications with Earth without significant time delay of signal. The successful re-entry and retrieval marks the end of the 134 day mission. The spacecraft approaches Earth with a $v_{\infty} = 1.5456$ km/s. It will hit the upper atmosphere of 122 km at a relative speed of 11.65 km/s (including the worst-case angular velocity of the air). Lunar flyby tests will demonstrate the re-entry abilities of the heat shield on the Orion Crew Module[\[17\]](#).

The two Orion Crew Modules will survive re-entry. The Habitation Module and the two Orion Service Modules are destroyed during re-entry.

6.9 Mission Summary

The first human mission to an asteroid has many mission phases. It involves two launches that rendezvous to form a completed spacecraft which allows for all of the mission to be accomplished. The actual mission events are described here. Each event is described in greater detail in the corresponding section.

1. Support vehicles launch on trajectory to meet the crewed Orion in 2 days.
2. Crewed Orion launches on trajectory to the NEA 2 hours after the first launch.
3. Each vehicle performs system checkouts for 2 days prior to rendezvous procedure.

4. The support vehicles perform a Δv (0.0636 km/s) to match the trajectory of the crewed Orion.
5. The two vehicles rendezvous and dock to complete the spacecraft for the rest of the mission.
6. The completed spacecraft finishes its post-launch check-out and undergoes the rest of the 61 day transfer to the NEA.
7. Once at the NEA, the spacecraft matches orbits as best as possible 100m inside the NEA orbit (0.8524 km/s). This is similar to a rendezvous procedure. The spacecraft will observe the NEA at this time and plan out a course of action for astronauts to approach the asteroid.
8. For the rest of the 8 day stay, astronauts will conduct regular EVAs to the NEA to accomplish the mission objectives.
9. The spacecraft will depart the NEA (1.1379 km/s) and begin a 65 day return trip to Earth.
10. As the spacecraft approaches Earth, it will perform the appropriate atmospheric entry operations. These will ensure that both Crew Module capsules survive re-entry and that the Habitation Module and both Service Modules are safely destroyed in the atmosphere.

6.9.1 Timeline of Events

The important events of the mission are all listed below in Tab. 6.2 according to the dates on which they happen.

Table 6.2: Timeline of Proposed Mission Events

Date	Event
July 24, 2029 (two hours later)	Support Vehicles Launch Crewed Orion Launches
July 26, 2029	Support Vehicles Match Trajectory (0.0636 km/s) Rendezvous Begins
July 28, 2029	Rendezvous Completes (or sooner)
September 23, 2029	Spacecraft Arrives at NEA (0.8524 km/s) Astronauts Perform Duties at NEA
October 1, 2029	Spacecraft Departs NEA (1.1379 km/s)
December 5, 2029	Spacecraft Re-enters Earth Atmosphere
Total Mission Duration	134 Days
Total Mission Δv	2.054 km/s

6.9.2 Mission Analysis Graphs

The first graph shows the distance from Earth over the course of the mission. There are lines that show the distance from Earth for both the spacecraft and asteroid SG344. Of course, the spacecraft is near the asteroid during the time it spends in formation flight with the NEA. These days are marked with the red vertical lines that are seen. The plot is shown below in Fig. 6.10.

Figure 6.10: Geocentric Euclidean Distance to the human spacecraft and asteroid 2000 SG344.

This graph is normalized to units of light-seconds to show the estimated communication delay for radio signals. It is seen that the signal delay doesn't exceed 30 seconds. Therefore a round trip command and response from the spacecraft such as a ping should not take longer than a full minute.

The second mission graph shows a 3-D representation of the trajectory of the spacecraft throughout the entire mission. Like the previous graph, this plot is geocentric. This graph is labeled with a few markers that will show the direction of motion along these lines. The 3-D position data for both the spacecraft and the NEA is shown below in Fig. [6.11](#).

Figure 6.11: ECI 3C Position Data to the human spacecraft and asteroid 2000 SG344. Two markers indicate the arrival and departure locations and give a general direction to all of the lines.

This plot allows Earth based observation and tracking. Because the coordinates are based upon the ECI frame, any object would need to account for its displacement from the center of the Earth before pointing a receiving antenna. For a transmission to the spacecraft, the appropriate amount of lead time would need to be added to reach the spacecraft. The first figure of this section, Fig. 6.10 will provide this value.

Chapter 7

Conclusion

The alternative mission architecture offers an attractive prospect for a crewed asteroid mission. It was the better option for two separate asteroid missions by offering a mission with lower Δv requirements and shorter mission durations than the low-Earth parking orbit rendezvous mission architecture. The new architecture depends heavily upon the ability to rapidly launch spacecraft and the rendezvous abilities of those spacecraft, but the savings in Δv and mission duration may compensate. For this mission, the abort may happen as late as 7 days into the mission and still return the crew safely to Earth. If these challenges of the new architecture are surpassed, then this architecture might ease some of the difficulty of a longer duration, higher Δv mission.

A mission to NEA SG344 is created using the new architecture. A set of support vehicles, consisting of a back-up Orion MPCV and a Habitation Module, is launched first on a cargo configuration SLS. After 2 hours, a crew of 2 astronauts in the primary Orion MPCV is launched by a crew configuration SLS. These two launches both occur on July 24, 2029 and travel to the rendezvous destination on an Earth-escape trajectory to asteroid SG344. The rendezvous begins on July

26 to form a spacecraft that is capable of completing the mission to the NEA. Rendezvous is completed by July 28 on the outbound journey to the NEA. At the NEA, the spacecraft will spend 8 days, September 23 to October 1, in formation flight while astronauts perform their EVAs. The spacecraft remains inside a tight control sphere to remain close to, but prevent collision with, the asteroid. The astronauts will then return to Earth and re-enter the atmosphere on December 5. The total mission lasts 134 days and requires 2.054 km/s of Δv maneuvers.

Appendix A

Lambert Analysis

Lambert’s Problem is typically used with a “Two Position Vectors and a Δt ” ballistic problem. Many methods have been developed over hundreds of years to solve the problem. Lambert algorithms are heavily used in interplanetary or Earth-departure trajectory studies to create Pork Chop plots. These show mission launch windows and arrival dates and their associated Δv costs. Pork Chop plots are discussed and shown later in Section [A.3.1](#) on page [91](#).

Many methods exist to solve Lambert’s problem. At a minimum, they all take in two position vectors, $\vec{R}_i$ and $\vec{R}_f$, representing the initial and final vector position measurements, and the time between measurements. The algorithm will go through a series of steps to determine the velocities at the initial and final positions. The calculation for the final velocity is much quicker once the first has been found. Many of the Lambert algorithms find the initial velocity of the object through the Lagrange Coefficients, f and g . Once the algorithm finds these coefficients, it is a simple calculation to find the initial velocity vector, $\vec{R}_1$. In addition, the algorithm would only need very little more time to calculate $\dot{f}$ and/or $\dot{g}$, in order to solve for the final velocity vector, $\vec{V}_2$. This expansion is not

always needed because present-day orbital propagators do not require multiple velocity measurements for accurate propagation of space objects.

However, for ballistic trajectories, it may be faster to use this calculation to find the arrival Δv needed for the mission. A mission from Earth to Mars would need to burn to escape Earth, then burn again to obtain a desired orbit around Mars if the mission called for that.

Sometimes, Lambert algorithms will take additional arguments as well. More complex algorithms can account for full orbits in between position vector measurements. In Fig. [A.1](#) below, the object in orbit completes one full orbit between observations. The overlap is shown in the darker region. The figure only shows a single complete orbit, however, there is no limit to the number of orbits an object can complete between measurements other than the computational ability of the analysis equipment.

Figure A.1: Multi-Rev Lambert Case. This object completes one full orbit between measurements. Objects may complete more full orbits between measurements.

In addition, algorithms can also provide an accurate result for going either the short way or the long way between position vectors. Some algorithms handle this ambiguity with a ‘longway’ or ‘shortway’ string input and some use a negative input for the time between measurements. Both methods of travel are shown below in Fig. [A.2](#).

Figure A.2: Long Way - Short Way Lambert Case. The object may go either the short way or the long way between $\vec{R}_1$ and $\vec{R}_2$.

For the two observations in Fig. A.2, it is doubtful that a small Δt would suggest a long-way transfer or a multi-rev case. But with cases where observations are spaced longer apart, sometimes on the order of days, the scenario may get less clear. In Fig. A.2, the $\Delta\theta$ is small. A larger Δt may suggest that a short-way transfer did not happen. This scenario could be a long-way transfer or a short-way + one complete orbit.

A.1 The Methods

In the analysis of Lambert algorithms presented in this paper, each algorithm will be briefly discussed. A quick summary of the process will be mentioned. In addition, some initial observations will be made concerning the performance of the algorithm.

A.1.1 Minimum Energy Method

The Minimum Energy Method is the first of the Lambert algorithms to be looked at in this paper. This algorithm consists of a few simple steps to calculate some simple geometric values. It then makes an assumption that the orbiting spacecraft is at its minimum energy state by solving for the orbit parameter of minimum energy, P_m . Next, the algorithm finishes the quick and explicit calculation of the first observation velocity vector, $\vec{V}_1$.

This first Lambert algorithm is by far the simplest. It is the only method that does not involve iteration loops. The Minimum Energy Method is only accurate when the object in orbit is at or near the minimum energy state. This method should not be used unless as a comparison to other Lambert algorithms to possibly determine if an object is at or near its minimum energy state.

A.1.2 Gauss's Solution

Karl Friedrich Gauss developed a method for a Lambert solution. The method uses a few defined parameters, ℓ and m , and an iterative process that, in the end, solves for the Parameter of the orbit, P and the semi-major axis, a . The angular momentum, h , of the orbit can then be calculated. This is then used to solve for

the parameters f and g and finally the velocity of initial measurement, $\vec{V}_1$. The method could easily be expanded as mentioned before to solve for $\dot{f}$ and $\dot{g}$.

Gauss's Solution is very accurate, fairly fast, and mostly a direct calculation. However, during the calculation, there are parameters that will break the algorithm. If the angle between measurements $\Delta\theta$ is either 0° or 180° . In addition, the iterative process was designed for small angles. If the $\Delta\theta$ between $\vec{R}_i$ and $\vec{R}_f$ is fairly large, then more iterations are often required to complete the algorithm. Furthermore, during the iterative process, a cubic equation must be solved that yields only 1 positive real root. In order to find this root, one must check each root that is given by a root solver algorithm. These three issues make Gauss's Solution to Lambert's Problem unattractive to the modern day astrodynamist.

A.1.3 Universal Variable Method

The Universal Variable method uses a few variables and parameters that were created to apply to all orbits. Similar to the Gauss Solution, this algorithm seeks to find the parameters f , g , and $\dot{g}$ in order to solve for the velocities $\vec{V}_i$ and $\vec{V}_f$. The Universal Variable method applies a lot of operations to a parameter z including iterations and multiple function calls until the solution converges after a Newton iteration.

This method is also very accurate. There are no singularities at $\Delta\theta = 0^\circ$ or 180° . However, it is a fairly expensive algorithm to compute. The procedure undergoes multiple iterations and multiple function calls. Furthermore, there are many calls to the trigonometric functions of sine and cosine. These all add computation time to the algorithm. However, of the methods mentioned so far, it is the most robust (due to not having a singularity) and it is accurate.

A.1.4 Battin’s Method

This method is largely based on Gauss’s Solution and is formally called Gauss’s Solution with Battin’s improvements, but for the purposes of this paper, it will be referred to as Battin’s method. The improvements were developed and published by Richard Battin in 1987[24]. The algorithm uses a few tricks to take care of the present-day shortcomings of Gauss’s Solution without any drop in accuracy. The method also becomes very fast.

Battin’s improvements solve all of the issues previously mentioned with Gauss’s Solution. First, Battin eliminates the 180° *irc* singularity by calculating the ℓ and m parameters in a slightly different way. By doing this, the singularity is removed. However the singularity at 0° and 360° still remain. This should not be a concern though because two measurements that fall into this category are unlikely to exist without being the same measurement. Next Battin removes the long iterative process using some converging continued fractions. The more levels of the continued fraction result in additional decimal places of accuracy in the parameter. By doing this, the iterative process converges in roughly half the number of iterations over all values of $\Delta\theta$. Finally, Battin again uses continued fractions to explicitly solve for the positive real root of the cubic equation. The code being tested now has been written for elliptical orbits.

The improvements made by Battin do not fully remove the singularities inherent in Gauss’s Solution, but they greatly improve the general robustness of the algorithm. Nearly all of the trigonometric functions are also eliminated in this method as well except for the calculation of the g parameter at the tail end of the procedure. These improvements make this method one of the quickest of the accurate Lambert algorithms.

A.1.5 Izzo’s Method

In late 2009, Dr. Dario Izzo (ESA - Advanced Concepts Team) presented an algorithm designed to be extremely quick and reliable. This algorithm avoids the use of the Lagrange Coefficients and instead opts to use a different method that is claimed to provide better numerical results as $\Delta\theta$ approaches 180° . The Matlab code for this method is taken from the MathWorks website[\[25\]](#).

This Lambert algorithm also has the ability to calculate the desired velocities with multiple orbits between observations. It also allows for “left branch” or “right branch” solutions that correspond to the different methods of travel as discussed earlier. The method is said to fail when many full revolutions occur between measurements. There will be no multi-revolution cases for this paper. This method was designed to be compiled using Matlab in order to increase its running speed, however, none of the Lambert algorithms will be compiled in order to keep it even throughout.

A.1.6 Summary of Lambert Algorithms

Shown below in Table [A.1](#) is a summary of what has been discussed about the five Lambert algorithms so far. Actual numeric calculations and comparisons are done in Section [A.3](#).

Table A.1: Summary of Lambert Algorithms

Lambert Method	Summary of Performance Metrics
Minimum Energy	<ul style="list-style-type: none"> + Extremely Quick – Highly Inaccurate for Pork Chop Plots
Gauss’s Solution	<ul style="list-style-type: none"> + Highly Accurate – Has a 0° and 180° singularity – Longer Calculations for larger $\Delta\theta$ values
Universal Variable	<ul style="list-style-type: none"> + Very robust + Highly Accurate – Longer Calculation (Iterations and Function Calls)
Battin’s Method	<ul style="list-style-type: none"> + Highly Accurate + No 180° Singularity + Fast Calculation Time – Retains 0° singularity (not problematic)
Izzo’s Method	<ul style="list-style-type: none"> + Highly Accurate + Fast Calculation Time + No $\Delta\theta$ Singularities – Fails with many revolutions (not problematic)

A.2 Test Cases

Before a lengthy study involving many different scenarios or being used to create a Pork Chop plot, a Lambert algorithm must successfully solve some test cases. There will be two test cases. The first will be to solve an example as outlined below in Table A.2. This case is given, with inputs and expected outputs, in Curtis[26] on page 208–210. This is scenario is done the short-way between observations and does not include multiple revolutions. Please also note that

the $\vec{V}_2$ is shown because some Lambert Algorithms already give both solutions natively, but a comparison will only be done for the first velocity, $\vec{V}_1$. It was found that all algorithms that gave both results were correct in their calculation of the second velocity.

Table A.2: First Test Case for Lambert Algorithms

Parameter	Value
$\vec{R}_1$	[5000, 10000, 2100] km
$\vec{R}_2$	[-14600, 2500, 7000] km
Δt	3600 s
μ	398600 $\frac{km^3}{s^2}$
$\vec{V}_1$	[-5.9925, 1.9254, 3.2456] $\frac{km}{s}$
$\vec{V}_2$	[-3.3125, -4.1966, -0.38529] $\frac{km}{s}$

Using each of the five methods discussed in Section A.1, the results are obtained in Table A.4 shown later in Section A.3.

If a method successfully solves this first test case, it will go on to solve a real world example provided by JPL Horizons system. The JPL Horizons database contains ephemeris data for nearly every celestial body in the solar system. Data can be accessed and received in many different ways including the option to save this data to a text file. The second test that a Lambert Algorithm must solve is to find the correct velocity based on two observations given by JPL Horizons. The chosen case will be assumed to be correct.

In order to ensure that Lambert Algorithms are correct for use outside Earth orbit, a body in a sun-centered orbit is chosen. Near Earth Asteroid GP2 is chosen and the parameters are shown below in Table A.3. Due to this problem being sun-centered instead of Earth-centered, the parameters will all be significantly larger. Units will remain the same as the previous example because the algorithms expect

to see these inputs. Excessive digits have been omitted from being displayed here, but remained in the test. These numbers will more accurately represent the interplanetary trajectories that a Lambert algorithm will be expected to solve in present-day situations.

Table A.3: Second Test Case for Lambert Algorithms

Parameter	Value
$\vec{R}_1$	$[-8.7804\text{E}+07, -1.2312\text{E}+08, 2.0637\text{E}+06]$ km
$\vec{R}_2$	$[1.3652\text{E}+08, -4.9614\text{E}+07, 1.9450\text{E}+06]$ km
t_1	2456293.5 JD
t_2	2456393.5 JD
Δt	100 days = 8640000 sec
μ_{sun}	$1.32712 \times 10^{11} \frac{km^3}{s^2}$
$\vec{V}_1$	$[25.577, -15.676, .50086] \frac{km}{s}$
$\vec{V}_2$	$[11.725, 28.887, -.54172] \frac{km}{s}$

A.3 Results

Using the sample case previously mentioned in Table A.2, each of the 5 Lambert algorithms are run. The true value and each of the calculated values are shown in Table A.4 below.

Table A.4: First Test Case Results

Method	$\vec{V}_1$ Result ($\frac{km}{s}$)
True	[-5.9925, 1.9254, 3.2456]
Minimum Energy	[-3.5206, 3.9780, 3.0862]
Gauss's Solution	[-5.9925, 1.9254, 3.2456]
Universal Variable	[-5.9925, 1.9254, 3.2456]
Battin's Method	[-5.9925, 1.9254, 3.2456]
Izzo's Method	[-5.9925, 1.9254, 3.2456]

From these results, it is clear that the Minimum Energy Method does not produce results that are accurate enough for use. Therefore, it will not be used in any future tests, nor in any studies going forward. The rest of the algorithms returned excellent results, showing no signs of error. The algorithms that yielded a final observation velocity vector also all returned an accurate answer.

The second test case, as described in Table A.3, was run for the 4 remaining Lambert algorithms. Again, the true initial velocity vector and the result of each algorithm is shown below in Table A.5.

Table A.5: Second Test Case Results

Method	$\vec{V}_1$ Result ($\frac{km}{s}$)
True	[25.577, -15.676, 0.50086]
Gauss's Solution	[25.577, -15.676, 0.5008]
Universal Variable	[25.577, -15.676, 0.5008]
Battin's Method	[25.577, -15.676, 0.5008]
Izzo's Method	[25.577, -15.676, 0.5008]

All of the methods returned excellent results. The z -direction values have approximately 0.01% error consistently across all methods. Other axis directions yield less error. All of these methods give results that are accurate enough for

use in our final test.

A.3.1 Pork Chop Plots

Pork Chop plots are tools that astrodynamists and engineers use to determine launch windows and arrival dates for interplanetary missions. A series of launch dates is placed on the x -axis and arrival dates on the y -axis. A Lambert problem is solved for each combination of dates in a 2-D array. The resulting graph shows the Δv requirements on a contour plot. Typically, the information shown is the C3 escape energy and the transfer time. In this case, the outbound v_∞ is shown instead of the C3 ($C3 = v_\infty^2$). Below, Fig. [A.3](#) shows a pork chop plot for one of the asteroids studied in this paper.

All of the calculated cases in this figure are Type 1 trajectories. These trajectories are quicker and cover less distance than Type 2 trajectories by staying inside the outer orbit during the transfer.

Figure A.3: Pork Chop Plot for Asteroid QJ142. Transfer times and escape trajectory v_∞ for a 2-D array of departure and arrival dates from Earth. Transfers of less than 45 days are not calculated.

In order for the Lambert method to be viable in modern-day use, it must be able to create an accurate pork chop plot. This is the primary metric for success of a Lambert algorithm. All of the 4 remaining Lambert algorithms returned nearly identical pork chop plots. The only difference was the location in which the contour line labeling was done by Matlab. Every other pixel of data lined up perfectly between the four methods.

There is no true pork chop plot for comparison. These methods are returning accurate results and it is likely that the algorithm is accurate for all individual scenarios until the pork chop plot is completed.

Because each of the remaining methods could successfully serve the needs of the project in terms of accuracy, another metric must be considered. It has been shown that each method is robust enough to solve any case that is likely to be observed, except for Gauss's Solution. This method has a 180° singularity that is still possible to encounter in Lambert applications.

The time of computation for the Lambert algorithm is more important than the 180° singularity. The final decision of which Lambert algorithm will be used will be determined by the fastest computation of the 8 Pork Chop plots specified earlier in this paper. The only change necessary will be the selection of the Lambert algorithm and its corresponding arguments to facilitate the correct use of units. The operation is timed using the tic/toc functions of Matlab and rounded to the nearest hundredth. Results from this quick study are shown below in Table A.6. This test was run multiple times and the fastest result was recorded.

Table A.6: Timing of Each Lambert Algorithm. The time required to create all 8 Pork Chop plots.

Method	Time of Computation (sec)
Gauss's Solution	37.44
Universal Variable	215.82
Battin's Method	12.51
Izzo's Method	17.57

From this table, it is clear to see that Battin's Method is the fastest at solving Lambert's Problem and the improvements greatly reduce the total computation time. This result must be taken with a grain of salt. Battin's Method and Gauss's Method were written to be optimized to solve an elliptical case. In order to solve a hyperbolic trajectory, these codes would most likely need to be modified. In addition, the second place finisher, Izzo's method, is taken from a Matlab code

that was written to be compiled and robust for all cases. If each method was compiled, Izzo's method may be the quickest for general use.

A.4 Conclusion

When the Lambert problem will only involve elliptical orbits without any complete orbits, Battin's Method will be the best option. Battin's Method shall be the Lambert algorithm of choice for the project. This method has proven itself to be accurate, robust, and quick enough to provide the data for the rapid creation of Pork Chop plots necessary for interplanetary mission planning. Furthermore, all of the inclinations mentioned in Table [A.1](#) were correct. Each method has the strengths and weaknesses mentioned.

Appendix B

NEA Trade Study

B.1 NEAs Selection

There are many thousands of asteroids to choose from as a destination. A thorough analysis and mission design for each of these is not feasible. A large number of papers and databases exist to help trim the selection of asteroids. These asteroid finders create a top few number of asteroids and provide information on them as well.

In order to remain objective, only the asteroid name was recorded as a NEA was recorded. A repeat orbital analysis was performed after a list of candidates was made. The analysis was based upon raw data to ensure the integrity of candidate asteroids. At such an early point in the procedure, the importance of one factor against another is unknown and therefore candidate asteroids are kept as potential candidates based more on total mission Δv requirements and launch windows rather than asteroid size.

B.1.1 Selection Process

The NEAs mentioned in this thesis do not represent a complete and comprehensive list of viable candidates for a human asteroid mission. These NEAs were selected from accessible candidates in asteroid finder resources and may not be ideal candidates. These were not sorted by “best” candidates to preserve neutrality. The initial 5 candidates are shown below.

- 2009 BD
- 2006 QQ56
- 2003 YN107
- 1991 VG
- 2000 SG344

From this, a comprehensive Lambert analysis was done using Battin’s Method as described in Sec. A.4. Using the state vector data from JPL Horizons (2013 to 2053, one-day steps), a Lambert analysis was done for mission trips to this asteroid. A range of 45 to 100 days was selected as a possible one-way trip time. Every combination of possible departure and arrival dates for the entire 40 years was analyzed to launch window estimates. This created a huge 2-D array of Δv values for the one-way trip to the asteroid. In order to remain a potential candidate, the NEA had to have a Δv transfer value of ≤ 5 km/s at a point in the surface plot. This also had to be at a reasonable date to avoid designing a mission that couldn’t be planned in time. It was decided that the launch window had to begin no sooner than 2017. These criteria were chosen to be forgiving

to avoid pre-maturely eliminating any potentially good candidates based upon a preliminary analysis.

Some of the asteroids were eliminated, and new ones were placed into this group of candidates. The new list of possible candidates is shown below.

- 2009 BD
- 2000 SG344
- 1991 VG
- 2006 RH120
- 2008 UA202
- 2001 GP2
- 2001 QJ142
- 2008 HU4

The same analysis was repeated on the new arrivals, and it was found that all of these met the minimum criteria. Doing this also provided a look into a set of possible launch windows for the asteroids. This did not yield a certain day, but it provided an estimate at the time frame that could be used for the pork chop plots. This success meant that these asteroids were going to be the ones used in the trade study.

B.2 Trade Study

The optimal NEA will be determined by an orbital analysis, the year of the launch window, and the size of the NEA in a trade study.

B.2.1 Orbital Analysis - Pork Chop Plots

As mentioned in Sec. [B.1.1](#), the surface plot provided an estimate at the launch window(s) for the respective NEA. The surface plot revealed the hyperbolic excess velocity, v_∞ , requirement for a one-way transfer from the Earth to the NEA. This successfully narrowed down the wide range of dates for a more in-depth analysis.

The next step in the analysis was to create pork chop plots for the asteroid candidates. Pork chop plots are graphs that show the requirements for a transfer trajectory. A series of departure dates are placed on the x -axis in order to create a launch window. Arrival dates are placed on the y -axis. A Lambert problem is solved for each combination of dates that represent a feasible travel time. The 2-D array of values represents the v_∞ that is necessary for a transfer to occur with those parameters. The v_∞ represents the velocity needed to reach its destination. If the object is currently in orbit around a body and must leave that orbit, then v_∞ is the velocity that must be obtained after hyperbolic escape. Often, the v_∞ value is squared to find the C3 escape energy for the transfer. An example pork chop plot is shown below in Fig. [B.1](#).

Figure B.1: Earth to Mars pork chop plot for the 2005 opportunity.[\[27\]](#)

For the pork chop plots in this paper, the decision was made to leave values as v_∞ . This is a result of setting the cut-off mentioned earlier at a $v_\infty = 5$ km/s. All asteroids had to beat this requirement to remain candidates in this study.

Arrival dates are calculated using the departure dates and the maximum and minimum one-way travel times. As before, the minimum travel time is set to 45 days and the maximum travel time is set to 100 days. These are supposed to be the limits on travel times and not actual possibilities. A travel time of 100 days would result in a mission that was likely too long and a travel time of 45 days would likely be too costly in fuel.

The creation of pork chop plots is an iterative process because the initial launch window isn't known. It is necessary to continually narrow the launch window as appropriate. The pork chop Matlab code was set up so that it would only need a start and end of the departure dates for a NEA to create the pork chop

plots. Travel durations were hard-coded to determine the arrival dates. From this, the dates were easily narrowed down further for each NEA based upon the surface plots from earlier.

All transfers that are shown in these pork chop plots are Type 1 transfers. This can be seen by observing that the transfers are all under diagonal line with the steep contour gradient. These transfers are shorter and quicker than Type 2 transfers because they remain inside orbit of the target object. It makes more sense to use the quicker Type 1 transfers for this human mission.

Before narrowing down the launch dates too much, a wide range of departure dates was made and kept. This is to show what would happen if the launch date was missed by a lot. Once a best opportunity was found, the departure dates were narrowed further. The two sets of departure date ranges are shown below in Table B.1.

Table B.1: The wide and narrow departure ranges for each of the asteroids. The units represent the day number index of the analysis. Day 1 = Jan. 1, 2013.

Asteroid	Wide Departure Range	Narrow Departure Range
2009 BD	7500-8000	7800-7950
2000 SG344	5700-6200	5900-6100
1991 VG	9400-9800	9500-9700
2006 RH120	5400-5900	5550-5750
2008 UA202	5850-6200	6000-6150
2001 GP2	2050-2350	2150-2350
2001 QJ142	3850-4300	3950-4150
2008 HU4	4600-5050	4900-5050

Appropriately entering these values into pork chop plots yielded accurate and nice looking pork chop plots that encompassed the best possible departure

and arrival dates. These plots shall be used to determine the effects of slight alterations to departure and arrival dates based upon the contours.

Return Pork Chop Plots

Since this is a human mission, a return trip to the surface of the Earth must be made. Therefore, pork chop plots to return to Earth can be made. These follow the same procedure to make, except that the mission departs the asteroid and arrives at the Earth. Additionally, the dates are all constrained and can be directly calculated based upon Earth launch date and other previously set parameters. The limits chosen for the NEA stay duration were 6 and 12 days. The earliest that the spacecraft can depart the asteroid is *earliest launch date + minimum transfer time + minimum stay duration* and the latest the spacecraft can depart is *latest launch date + maximum transfer time + maximum stay duration*.

While doing the analysis, the outbound pork chop plot of 2008 UA202 was found to give very attractive results. However, the return pork chop plot showed that returning from the NEA would require large Δv maneuvers.

B.2.2 Launch Window

Each of the 8 NEAs in consideration has a launch window after 2017. The ephemeris data from Horizons goes from 2013 to 2053. An asteroid with a launch window after 2053 was not considered. The year 2017 is also too soon for a mission, especially if it uses the Space Launch System. An ideal launch date is around the years 2025 to 2030.

In the table below, the launch opportunities for each of the 8 NEAs is shown.

Many of these have multiple launch windows in the range of the ephemeris data due to a short synodic period. The best opportunity month and year are shown below in Table B.2. These were found from the orbital analysis in the previous section.

Table B.2: Month and year of launch window for all 8 NEAs.

Asteroid	Launch Window
2009 BD	June, 2034
2000 SG344	July, 2029
1991 VG	Feb, 2039
2006 RH120	June, 2028
2008 UA202	July, 2029
2001 GP2	Feb, 2019
2001 QJ142	Feb, 2024
2008 HU4	July 2026

All of these show good launch windows except for three. Asteroid 2001 GP2 has its launch window a little early and asteroids 1991 VG and 2009 BD have their launch window a little late. Asteroid 2001 QJ142 has a launch window that is slightly outside of the ideal window. Many of these asteroids have mid-summer launch windows which should help avoid the risk of hurricane interference or cold weather delays.

B.2.3 NEA Size

Larger NEAs have traditionally been the goal of previous missions as mentioned in Sec. 1.2.1. A larger NEA will be easier to see, have a slower spin rate, and be easier to attach to due to larger surface area and lower spin rates. The ideal NEA would be as large as possible without providing too much gravity to

affect attachment.

Size is calculated through absolute magnitude measurements and albedo estimates. The measured absolute magnitudes and calculated approximate diameters of the candidate asteroids are shown below in Table B.3.

Table B.3: Absolute magnitudes and estimated diameters of all 8 NEAs.

Asteroid	Absolute Magnitude[21]	Mean Diameter Estimate[21]
2009 BD	28.3	3.7-17m
2000 SG344	24.8	19-86m
1991 VG	28.4	3.7-16m
2006 RH120	29.5	2.2-10m
2008 UA202	29.4	2.3-10m
2001 GP2	26.9	7.3-33m
2001 QJ142	23.4	36-161m
2008 HU4	28.2	3.9-18m

With size as the only consideration, 2001 QJ142 is the best NEA. In second place is 2000 SG344. All of the others are far behind for this category.

B.3 Trade Study Scoring and Results

A detailed study of all 8 NEAs would be too much for this paper. Instead, the 8 asteroids that remain will be narrowed to a top 3 based upon the orbital analysis results in the pork chop plots, the launch window, and the size of the asteroid.

As mentioned, the important criteria for the human mission are the orbital aspects, the time of the launch window, and the size of the asteroid. The orbital

analysis is the most difficult to judge based upon the pork chop plots. These were scored based upon the sizes and overlaps of the contours. A larger area of lower cost transfer requirements for the same transfer time will earn a better score. There is obviously some subjectivity to this, but the order of best to worst remains the same. Launch windows and size are easier to score objectively. In the end, the goal is to find the top 3 and perform a mission analysis on them. The trade study is shown below in Fig. B.2.

	<u>Orbital Analysis</u>		<u>Launch Window</u>		<u>Size of NEA</u>			
	Wt. = 5		Wt. = 3		Wt. = 2			
NEA	Value	Score	Year	Score	Diameter	Score	Total	Rank
2009 BD	<i>n/a</i>	2	2034	3	3.7-17m	1	21	7
2000 SG344	<i>n/a</i>	4	2029	5	19-86m	4	43	1
1991 VG	<i>n/a</i>	3	2039	2	3.7-16m	1	23	5
2006 RH120	<i>n/a</i>	5	2028	5	2.2-10m	1	42	2
2008 UA202	<i>n/a</i>	2	2029	5	2.3-10m	1	27	4
2001 GP2	<i>n/a</i>	1	2019	1	7.3-33m	2	12	8
2001 QJ142	<i>n/a</i>	3	2024	3	36-161m	5	34	3
2008 HU4	<i>n/a</i>	1	2026	5	3.9-18m	1	22	6

Figure B.2: NEA Selection Trade Study. A higher number represents heavier weighting or a better score.

From the trade study, the best three options, in order, are:

1. 2000 SG344
2. 2006 RH120
3. 2001 QJ142

These three NEAs are the top three NEAs from the original list of eight. Mission analysis will be done for each of these to determine a best mission.

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