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SHORT DURATION MISSIONS TO EARTH CROSSING ASTEROIDS

THESIS

James B. Millar, III, 2d Lt, USAF

AFIT-ENY-MS-16-M-228

DEPARTMENT OF THE AIR FORCE
AIR UNIVERSITY

AIR FORCE INSTITUTE OF TECHNOLOGY

Wright-Patterson Air Force Base, Ohio

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SHORT DURATION MISSIONS TO EARTH CROSSING ASTEROIDS

THESIS

Presented to the Faculty

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in Partial Fulfillment of the Requirements for the

Degree of Master of Science

James B. Millar, III, B.S.

2d Lt, USAF

March 2016

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SHORT DURATION MISSIONS TO EARTH CROSSING ASTEROIDS

THESIS

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AFIT-ENY-MS-16-M-228

Abstract

My investigation of the Near Earth Object (NEO) catalog has led to identification of numerous short duration, under 40 days, mission opportunities in the future for three different mission types: uncrewed fly-by, uncrewed arrival, and crewed arrival. 2-body propagation techniques were used to model the orbits of various asteroid candidates and the Earth to determine when a close approach would occur. Once the dates were calculated, distance between the bodies was computed to estimate the ΔV to complete the mission. From the mission ΔV values, a possible mission duration was also computed. The values were analyzed to determine the best options for the mission types described above. One candidate is presented for the uncrewed fly-by opportunity, three for the uncrewed arrival mission, and four more for a potential crewed mission. The results show that a short duration mission is not only possible but should be strongly considered in the near future. These short duration missions are in sharp contrast to the common multi-month or year long duration proposals. Among the other wealth and resource benefits, short duration asteroid missions are of supreme importance for planetary defense and maintaining a powerful US space presence.

Acknowledgments

I would like to thank everyone who supported me along this journey, Stephanie, my family, my friends and finally my advisor, Dr. Wiesel. His guidance and leadership throughout my research has been unmatched and I could not have done this without him.

James B. Millar, III

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List of Acronyms

2BP	-	2-Body Problem
AFIT	-	Air Force Institute of Technology
AJAX	-	Advanced Jovian Asteroid Explorer
ARM	-	Asteroid Redirect Mission
AU	-	Astronomical Unit
BASiX	-	Binary Asteroid in-situ Explorer
CalTech	-	California Institute of Technology
COEs	-	Classical Orbital Elements
DARe	-	Dark Asteroid Rendezvous
JPL	-	Jet Propulsion Laboratory
LEO	-	Low Earth Orbit
NASA	-	National Aeronautics and Space Administration
NEA	-	Near Earth Asteroid
NEOs	-	Near Earth Objects
OSIRIS-Rex	-	Origins Spectral Interpretation Resource Identification Security - Regolith Explorer
SLS	-	Space Launch System

SHORT DURATION MISSIONS TO EARTH CROSSING ASTEROIDS

1. Introduction

1.1 Overview

The push to the moon in the 1950s and 1960s was one of the most impressive accomplishments of the human race. Weeklong missions to the lunar surface became almost common, and with such short mission durations, mission planning and execution became somewhat second nature to mission control. Since the final lunar mission in 1972, the National Aeronautics and Space Administration (NASA) and other organizations have been discussing numerous missions to Mars or Earth crossing asteroids [1]. These mission proposals, discussed extensively in Chapter 2, describe trips ranging from months to years but nothing along the lines of days or weeks.

NASA's proposed missions would be the longest trip any human has ever made in the vastness of outer space. The safety concerns that arise on a multi-month mission are incredible and lead to a lot of 'guess work' in regards to preparation and planning. A long duration mission also lacks the same excitement to the general public as the short duration lunar landings did. It is difficult to get excited about a trip to an asteroid in which the spacecraft will not even arrive until 6 months after launch. The solution to this problem is to determine potential candidates and develop fuel cost and time values to Earth crossing asteroids that lead to short missions of, at most, 40 days in duration.

1.2 Motivation

NASA, many other people, and organizations have tried to develop ideas on what is the best way forward with the US's current space program. With Mars being the ultimate long-term goal, many people are asking if there is anything we can focus on in the immediate future that will help accomplish that goal. The general consensus is to plan a mission to an asteroid prior to going to Mars [2]. This will not only enhance the knowledge of asteroids and the universe but present an opportunity to test different mission components that are necessary to a successful Mars landing. This consensus was solidified by President Barack Obama during a speech at Kennedy Space Center in Florida in 2010. The President discussed a set of crewed flights to test systems that will give the ability to explore beyond Low Earth Orbit (LEO). He claimed, "By 2025, we expect new spacecraft designed for long journeys to allow us to begin the first-ever crewed missions beyond the moon into deep space. We will start by sending astronauts to an asteroid for the first time in history" [3].

Developing short duration missions allows the public to get excited about the space program again while also giving NASA, or another organization, the ability to study numerous asteroids that come close to the Earth, leading to even more information about asteroid formation and composition. A short duration mission will require precise trajectories in order to minimize ΔV cost, which is sure to be fairly high. Due to the shorter distance and smaller amount of time in each mission, potential errors will be much more costly than if over a longer period of time. Though the costs are high, these short duration missions are necessary to expand our understanding of space and improve the public image of the space program.

The importance of identifying these short duration mission candidates cannot be understated in the planetary defense realm. Asteroids orbit at incredibly high speeds with trajectories that are not perfectly defined. For this reason, an Earth impact is,

unfortunately, more likely than desired. This risk, though, can be dramatically diminished through the introduction and completion of short duration missions. These missions could not only provide more accurate orbital properties of the asteroid but can also demonstrate how to operate around an asteroid. This information can then be used in the future to potentially detect and deter an asteroid from an Earth impact. Therefore, numerous short duration missions can have a great impact for the security of the nation.

Near Earth Objects (NEOs) are incredibly common. Figure 1 shows just how numerous these objects are [4]. Some of them orbit the Sun in very elliptical orbits

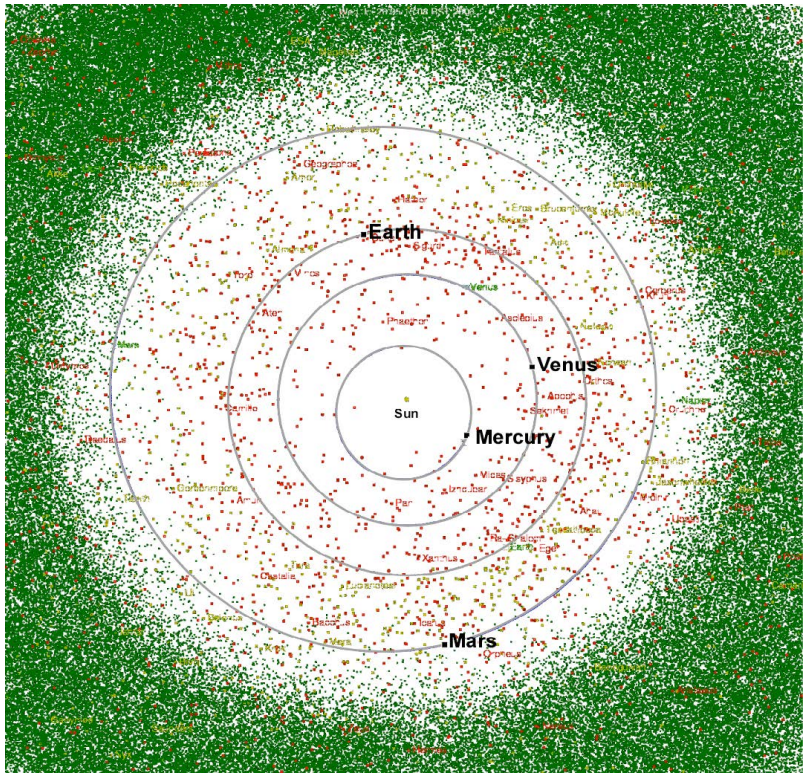

Figure 1. NEOs in the solar system. Red are Earth crossing, green are not. Reproduced from Scott Manley of the Armagh Observatory [4]**.**

and fly by the Earth at high speeds, often modeled as hyperbolic orbits with respect to the Earth. These asteroids are part of the Aten class, with a semi-major axis (orbital element used to define size of the orbit) less than 1 Astronomical Unit (AU)

and are about 6% of all Near Earth Asteroids (NEAs). An AU is further defined in Section 3.1.2, but it is the term to represent the mean distance from the Sun to the Earth. About 62% of known NEAs make up the Apollo class, and are defined by a semi-major axis larger than that of the Earth's. They also have similar Earth-like orbits, making potential ΔV small but time of mission high [5, 6]. These classes can be seen in Figure 2, reproduced from the NASA Near Earth Object Program website. There is also the Amor class of asteroids that is not Earth crossing but simply has an orbit around the orbit of the Earth, making up about 32% of known NEAs [5]. For this research these asteroids will not be considered for close approaches because they are not Earth crossing.

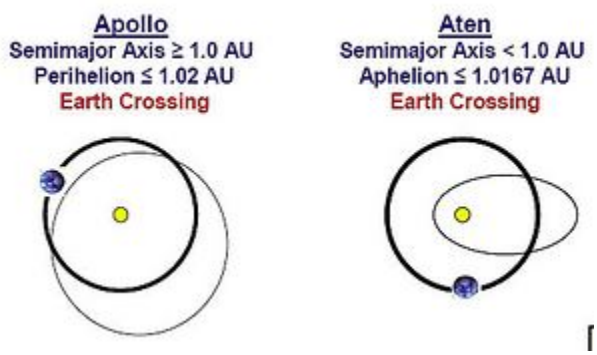

Figure 2. The two classes of earth crossing asteroids and their characteristics. Reproduced from NASA Near Earth Object Program website [5].

In 2009, the Lockheed Martin Space Systems Company detailed five specific reasons as to why we would want to explore an asteroid [6]. The first reason is knowledge. This includes studying the history of our solar system and learning how it was formed. There are billions of asteroids out there, each with its own history and story. Discovering out that story could open up the doors to a number of new scientific discoveries, e.g., origins of the universe, age of the asteroid, and where the asteroid came from. Next is security or planetary defense. This includes being able to "enable deflection of future hazardous impactors by understanding structure, composition, and how to

operate around small asteroids" [6]. If we understand how these asteroids are composed and how they are likely to move, we can develop better orbit determination and if needed, figure out a way to prevent them from impacting the Earth [2]. Planetary safety is of the utmost importance and should bring these asteroid fly-by missions to the top of the priority list. Wealth is another reason to explore an asteroid. There are so many resources available that could be used to extend life on Earth or help us expand into space. National pride is the fourth reason discussed to expand our horizons to NEOs. This involves re-establishing American power and presence in space. Finally, the last reason is to train for either another lunar mission or a Mars mission [6]. Up until now, the Earth has always been used as a safety blanket in terms of supplies and support. In planning for a mission to Mars or an asteroid, this safety blanket gets pulled off. Therefore, an asteroid mission could be a great stepping stone for a trip to Mars. Not only are there similarities in trajectories and navigation out of LEO, but also extensive docking and rendezvous practice that can be coupled with new propulsion methods, preparing the United States very well for a Mars mission [2].

The Committee on the Planetary Science Decadal Survey developed a book discussing the vision for the next decade of space exploration [7]. They describe three themes of exploration, each leading to its own set of questions. The themes are:

1. *Building New Worlds*
2. *Planetary Habitats*
3. *Workings of Solar Systems*

Building New Worlds involves the discovery of the initial stages and processes of the formation of our solar system. They ask how the giant planets and other large bodies came to be and what chemical or physical changes allowed these planets to

develop atmospheres and supply water [7]. *Planetary Habitats* asks where did organic matter arise from and if those processes are still in action today, while *Workings of Solar Systems* seeks to find if there are any solar system bodies that endanger the Earth or its orbiting bodies [7]. Every single one of these themes and questions can be helped through exploration of asteroids, making these short duration missions all the more beneficial and necessary.

1.3 Scope

Inspection of the NEO catalog, or the Horizon's On-Line Ephemeris System, developed by NASA's Jet Propulsion Laboratory (JPL) tells us of hundreds of thousands of NEO targets complete with ephemeris data and propagation using the most sophisticated orbit determination algorithms [8]. Two categories of objects are used, to include major bodies and small bodies. Small bodies are comets and asteroids while major bodies are virtually everything else. The Horizon's System tracks over 600,000 asteroids, as well as 3,200 comets, 176 natural satellites, Lagrange Points of the Sun-Earth system, the planets, and many other bodies [8]. This research focuses only on asteroids, and any targets not currently tracked by JPL were not considered. Propagating and modeling the asteroid used only 2-Body Problem (2BP) methods that do not take perturbations or other planets' gravity into account. The ephemeris data for each asteroid was run through a series of equations to tell us if the asteroid is a potential target. The constraints used were the distance between the asteroid and the Earth, the time it takes to reach the asteroid, and finally, the ΔV necessary for a mission. Missions were considered based on existing or developing NASA technologies, to include the Orion spacecraft and the SLS (Space Launch System). James Russell from Lockheed Martin developed a study to see the feasibility of using the Orion spacecraft, or multiple Orion spacecrafts, for a NEO mission. His study

primarily focused on the ability of the Orion spacecraft to house the necessary volume and mass required for an asteroid mission. Russell found that the vehicle can adequately host two astronauts for missions up to 4-6 months [9]. Being that the proposed missions to NEO asteroids are of much less time, we can conclude that the Orion spacecraft is sufficient.

A comparison that was thought of after the start of this research was the ability to model and predict the arrival of these asteroids compared to research that had already been done in this area. High fidelity models are used by JPL in order to predict the close approach times and distances of the aforementioned objects. The difference between the two classification groups determines how the data is stored. Giorgini et al., in their description of the models, state, "Major bodies are represented in pre-computed trajectory files which are interpolated very accurately to retrieve position and velocity at any instant." They continue, "Small-bodies have their position and velocity at one instant (only) compactly stored in a database and are then numerically integrated 'on-the-fly' by Horizons to other times of interest (also very accurately), using all known physics" [8]. Special perturbations are used to model the changes in the asteroid's (and the Earth's) orbit over time. Using this perturbation technique requires the use of numerical integration or integrating equations over periods of time to develop a solution as opposed to having a closed-form analytic solution, or an analytic approximation. The Horizon's system uses the asteroid's initial position and velocity vectors to integrate. Giorgini et al. explains that a, "variable order, variable step-size integrator is used to control error growth," to correctly model the effects of gravity and other perturbing forces [8]. This raises the question, though: can the same thing be accomplished with simpler, less computational methods? Restricted 2-body propagation uses orbital mechanics equations without perturbations or integration. These equations, used in this research, allow us to see how a 2-body propagation

technique compares to the numerical integration methods currently being used.

There are three main mission types that were investigated:

Mission Type 1: Uncrewed Fly-By

Mission Type 2: Uncrewed Arrival

Mission Type 3: Crewed Arrival

Each candidate asteroid was evaluated and a mission type was selected based on its characteristics.

1.4 Methodology and Resources

Modeling asteroid orbits and calculating when they cross the Earth's orbit requires the use of the classical orbital elements (COEs) used to describe each asteroid's orbit. These COEs include the semi-major axis, eccentricity, inclination, right ascension of the ascending node, argument of perigee, and true anomaly [10]. However, true anomaly is often converted to mean anomaly in order to propagate forward over a period of time. Due to the elliptical nature of orbits, the true anomaly does not change a constant amount over time. The missions proposed consist of a parabolic or hyperbolic escape trajectory toward the asteroid from a LEO parking orbit. With respect to the Earth, the orbit of the asteroid is modeled as a hyperbola because of the asteroid's close approach characteristics and the speed at which it passes by. A hyperbolic trajectory, with respect to the Earth, is again used to return to the Earth where reentry will occur. Missions are analyzed to compute the opportunities with the shortest duration and smallest ΔV. In terms of resources, this investigation was accomplished using computer programming and simulations through MATLAB® Version 8.1.0.605, R2013A, on a 64-bit Windows 7 Operating System, with 4.00 GB

of RAM and an Intel(R) Celeron(R) CPU E3400 @2.60 GHz processor. Therefore, AFIT computers are sufficient for completion.

1.5 Objectives

In order to continue to be a world power, the United States must be capable of leading the race to gain knowledge of space. President Obama stated that, "Our goal is the capacity for people to work and learn, and operate and live safely beyond the Earth for extended periods of time...And in fulfilling this task...we will strengthen America's leadership here on Earth" [3]. Through the research of potential targets and the optimization of the best way to reach those targets, the US Air Force could develop ways to maintain that leadership. The main goal of this research is to present NASA and other organizations with candidate asteroids for the three mission types that can be accomplished in a short period of time, less than 40 days. In addition, researching a topic allows for my growth and knowledge as an Air Force officer. Growing and learning came through the propagation of these 2BP equations over time using MATLAB® and other computer programs that strengthen my ability to comprehend the material and teach in necessary areas. Comparing my results to others while also analyzing the data to provide sufficient insight for mission opportunities can be directly related to a mission in the Air Force.

In order to accomplish these goals, the following objectives need to be met:

1. Search the NEO catalog and compute when asteroids cross the Earth's orbit
2. Determine where the Earth is at the crossing point and see if a mission is possible
3. Validate results versus other calculated results
4. Analyze that information to determine the best type of mission

5. Use that knowledge to recommend a mission plan

6. Identify mission candidates

2. Literature Review

The purpose of this chapter is to introduce the literature and ideas on the topic that have been proposed or are currently being worked on. The current NEO mission proposals that have been published are very similar in that they only propose one way of accomplishing a mission to an asteroid: a medium to long duration mission. This literature review will not only discuss the ideas on the topic but it will also describe why these proposals are not ideal and how this presented research will develop more efficient methods of accomplishing this goal. Beginning with a background of the 2BP is essential to discussing the orbital mechanics processes used. After the 2BP discussion, proposed NEO missions will be discussed and critiqued, providing a gateway into the methodology of solving the short duration mission problem.

2.1 2-Body Problem Background

First, a brief background is necessary to discuss how the 2BP was formulated. Johann Kepler and Sir Isaac Newton derived laws of orbital dynamics principles that we use today. Kepler developed three laws of planetary motion based off of Tycho Brahe's observation data that was the first to state the orbit of the planets is an ellipse, not a circle [11]. He developed the first two laws in 1609 and the last in 1619. The laws state [11]:

1. "The orbit of each planet is an ellipse, with the sun at a focus."
2. "The line joining the planet to the sun sweeps out equal areas in equal times."
3. "The square of the period of a planet is proportional to the cube of its mean distance from the sun."

While Kepler's laws describe planetary motion, Newton's laws provide an explanation of why the planets move as they do. Released in 1687 in his *Principia*, Newton described his three laws of motion as [11]:

1. "Every body continues in its state of rest or of uniform motion in a straight line unless it is compelled to change that state by forces impressed upon it."

2. "The rate of change of momentum is proportional to the force impressed and is in the same direction as that force."

3. "To every action there is always opposed an equal reaction."

Newton also developed an equation to express the Law of Universal Gravitation. This law has become the foundation of orbital dynamics, resulting in the development of the N-Body Problem. The equation corresponds to Figure 3 as well.

$$\boldsymbol{F_g} = -\frac{GMm}{r^2}\frac{\boldsymbol{r}}{r} \tag{1}$$

Where

$$\boldsymbol{F_g} = \text{force vector on mass } m \text{ due to mass } M$$

$$G = \text{universal gravitational constant}(6.67 \times 10^{-11}\text{N}^2\frac{\text{m}^2}{\text{kg}^2})$$

$$M = \text{mass of body one (kg)}$$

$$m = \text{mass of body two (kg)}$$

$$\boldsymbol{r} = \text{position vector from body one, } M\text{, to body two, } m$$

The N-Body Problem describes how an object travels through space and can be affected by numerous other bodies and forces. Each additional body adds a force to the traveling object that pulls it in a different direction. The use of all of these bodies

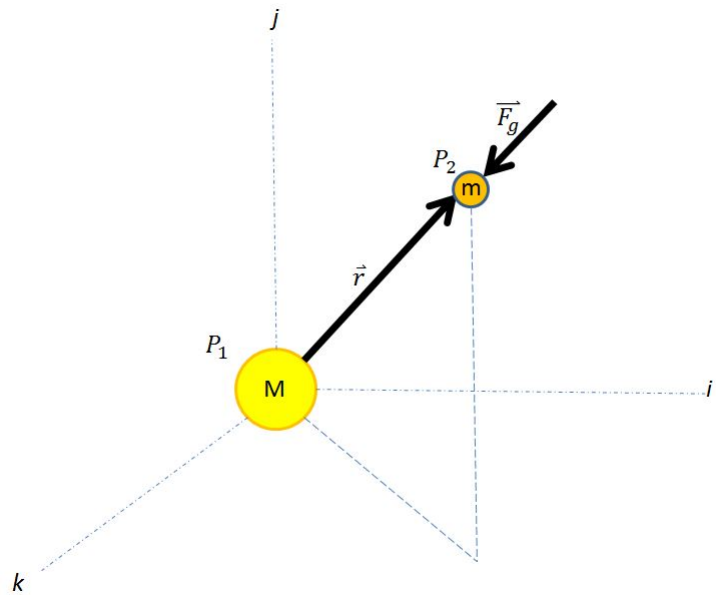

Figure 3. Depiction of the gravitational force F_g on mass m, where mass M is centered on the origin of the Inertial frame.

results in a higher fidelity model of the orbit of the object. However, the results presented in this research were developed using only 2 bodies. The 2BP uses three main assumptions to produce a closed-form analytical solution. These assumptions are:

1. The bodies are treated as point masses
2. Gravity along the line joining the masses is the only acting force [11]
3. The mass of the asteroid is negligible compared to the mass of the Earth or the Sun and the mass of the Earth is negligible compared to the Sun

The third assumption is the most important because stating $m << M$ allows us to work with the restricted 2BP. The restricted 2BP can be seen in Figure 3. Making it restricted allows us to say that the Sun, or in some cases the Earth, is located at the origin of the inertial plane. Without this assumption, Figure 3 does not hold.

The asteroid will be modeled in orbit around the Sun and upon close approach, the asteroid will be modeled around the Earth. When orbiting the Sun, the Earth effects will not be modeled and vice versa. Also, at times, the Earth will be propagated about the Sun without effects from the asteroid. The reason the research was performed using the 2BP is because it has a closed-form analytical solution to this problem. The solution is:

$$\ddot{\boldsymbol{r}} = -\frac{\mu}{r^3}\boldsymbol{r} \tag{2}$$

Where

$$\mu = \text{gravitational parameter}(\frac{\text{km}^3}{\text{s}^2})$$

This gravitational parameter comes from Equation (1) and the restricted 2BP assumptions.

$$\mu = G(M + m) = GM \tag{3}$$

Because

$$m << M, \text{in the restricted 2BP}$$

Using Equation (2) allows us to identify several constants of the motion, e.g., specific mechanical energy, ε, the angular momentum vector, $\vec{h}$, and the eccentricity vector, $\vec{e}$. A lengthy derivation can be used to develop a trajectory equation that we will also use in Chapter 3 to calculate the position of the asteroid. This derivation can be found in *Spaceflight Dynamics* by Dr. William Wiesel and *Fundamentals of Astrodynamics* by Roger Bate, Donald Mueller and Jerry White [10, 11].

$$r = \frac{p}{1 + e\cos\nu} = \frac{a(1 - e^2)}{1 + e\cos\nu} \tag{4}$$

Where

$$p = \text{semi-latus rectum of the orbit (km)}$$

$$e = \text{eccentricity of the orbit}$$

$$\nu = \text{true anomaly of the orbit}$$

Equation (4) computes the position of the body in orbit around the Sun using the parameters of the body's orbit. It is important to note, when p is used in Equation (4), the solution applies for all conic sections, but when a and e are used instead, the solution only applies to circles, ellipses, and hyperbolas. Equation (4) utilizes some new variables that have not yet been discussed but are of essential importance. From Equation (4), depending on the eccentricity value, four different conic sections can be produced. This means that using the restricted 2BP will produce four possible orbital paths for the asteroid. Table 1 describes the possible paths depending on eccentricity [11]. Not included in the table is the possibility of a degenerate ellipse, parabola, or hyperbola, if the eccentricity is equal to 1. These have a conic section of a line and can approach their degenerate cases depending on their respective ε.

Table 1. Describes the different results for orbital path based on eccentricity in Equation (4). Adapted from Bate, Mueller and White [11].

Eccentricity	**Conic Section**
$e = 0$	Circle
$0 < e < 1$	Ellipse
$e = 1$	Parabola
$e > 1$	Hyperbola

The definition of conic sections will not be discussed here but it is important to remember a few things regarding these orbital trajectories that stem from the 2-body assumptions and using the restricted 2BP. First, the body that the asteroid is orbiting around, whether the orbit is a hyperbola, parabola, ellipse, or circle, must be located

at one focus of the orbit. Secondly, the trajectory that the asteroid follows is in a plane that is fixed in inertial space. Finally, even though we have constants of the motion, the components of these constants are not necessarily always the same [11]. For instance, even though ε is a constant, the potential and kinetic energy components change throughout the orbit. This is a direct result of Kepler's Second Law and the conservation of angular momentum, which are the reason that as objects approach closer to their orbiting body, they move faster, and the farther away they are, they move slower. These trajectories also tell us another key fact about the orbit and that relates to the semi-major axis size. Table 2 relates semi-major axis values with the type of conic section.

Table 2. Describes the different results for semi-major axis based on orbital path. Adapted from Chobotov [12].

Semi-Major Axis	Conic Section
$a = r$	Circle
$a > 0$	Ellipse
$a \to \infty$	Parabola
$a < 0$	Hyperbola

Now that the orbital path has been discussed, we can begin to define the orbital parameters of the asteroid, the COEs. The COEs describe or define the orbit of one object around another. In this case, the orbiting object is the asteroid about the Sun, the asteroid about the Earth, or the Earth about the Sun. Figure 4 shows a very basic diagram of what each of these COEs look like for an asteroid orbiting the Sun. Starting with the aforementioned semi-major axis, a, the size of the orbit can be defined. The bigger the a value, the bigger the orbit. It has units of distance, usually measured in kilometers. The next element is eccentricity, which dictates the shape of the orbit. As shown above, the eccentricity, e, of the orbit, combined with a and ε, describes which orbital path the asteroid is following. Once we have the size and shape of the orbit, we can compute the inclination, i, of the orbit above the

ecliptic plane. The orbit of the Earth around the Sun defines the ecliptic plane which is the plane which always includes the center of the Sun and the Earth. Therefore, the inclination of the Earth's orbit around the Sun is always 0° in the ecliptic plane. The inclination of the asteroid's orbit now defines a new plane for the orbiting object, and since angular momentum is orthogonal to the the position and velocity vectors, it is orthogonal to this new plane. The next two elements determine where periapsis is located and at what location does the asteroid cross the ecliptic plane (provided the inclination is not 0°). The right ascension of the ascending node, Ω, is an angle that is defined from the vernal equinox (first point of Aries) to the point at which the ecliptic plane is crossed, the ascending node, or the point where the asteroid goes from the southern hemisphere to the northern hemisphere of the Sun. The vernal equinox direction is the vector from the Earth to the Sun on the first day of Spring for the Earth's northern hemisphere. This vector is shown as $\hat{\Upsilon}$ in Figure 4. The right ascension of the ascending node is a final swivel to pinpoint the exact plane the asteroid is in. Now, to see where periapsis is oriented in the orbit plane, another angle, the argument of perigee, ω, is measured from the ascending node to the location of periapsis. This tells us where in the orbit is the closest approach to the Sun. Finally, the true anomaly of the orbit, ν, is the angle measured from periapsis to the location of the asteroid in its orbit. A true anomaly of 0° means the asteroid is at periapsis. Much more in the area of 2-body propagation will be covered in Chapter 3.

Wiesel also uses the assumptions of the restricted 2BP to derive the fundamental equation of our calculations, the vis-viva equation [10]. Equation (5) will be used to calculate different orbital characteristics for the outbound, rendezvous, and divert maneuvers.

$$\varepsilon = \frac{V^2}{2} - \frac{\mu}{R} \tag{5}$$

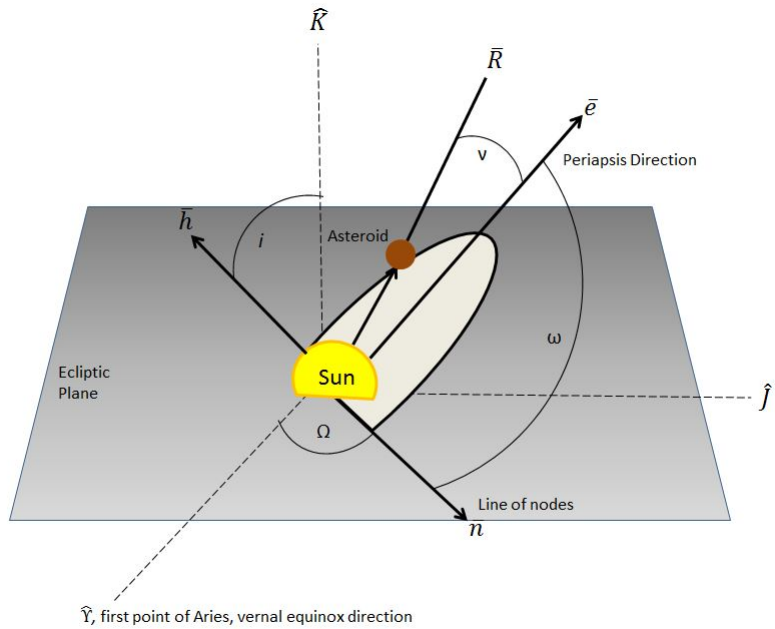

Figure 4. A diagram (not to scale) of an asteroid orbiting the Sun and how its classical orbital elements are defined.

Where

$$\varepsilon = \text{specific mechanical energy}(\frac{\text{km}^2}{\text{s}^2})$$

$$V = \text{magnitude of velocity of the orbiting body in the restricted 2BP}$$

$$R = \text{magnitude of position vector to the orbiting body in the restricted 2BP}$$

2.2 Proposed Missions in Development

As described in Section 1.2, numerous proposals have been published describing different missions to Earth crossing asteroids. Currently, NASA has decided on the Asteroid Redirect Mission (ARM) to focus its goals for the next decade [1]. The original ARM concept involves taking a piece of an asteroid, or an entire small asteroid itself, and redirecting it toward the moon. NASA developed two ways of accomplishing this. Option A is to deploy a container large enough to capture a free-flying asteroid of up to 8 m in diameter and then fly it into an orbit around the moon.

Option B is to have a vehicle land on a large asteroid, deploy its robotic arms, and pick up a boulder up to 4 m in diameter from the surface. This vehicle would then place this asteroid piece into a 'distant retrograde orbit' around the moon [1]. A 'distant retrograde orbit' was chosen because it is known to be more stable than a prograde orbit in the Earth-moon circular restricted 3-body problem [13]. More recently, NASA has decided to pursue option B.

The ARM is ideal in some ways and very limited in others. First, to bring a single piece of an asteroid back to the moon extremely limits the amount of the body we want to study. Rather than obtain an extravagant knowledge of multiple asteroids across the solar system, this plan would halt our study at only one. While it seems to overstep our bounds, thinking about numerous asteroids when nobody has ever landed on one, a plan that gives the ability to study multiple asteroids if desired is ideal. The proposed ARM also does not test crewed mission capabilities to distant objects but rather follows similar trajectories back toward the moon. Granted, the potential missions for asteroid visitations are by no means the same that were used for the Apollo program, and going to an asteroid in orbit around the moon is highly different than the moon itself, but to have a brand new crewed mission to an asteroid would be an extraordinary achievement and set us up much better for the future. A successful first mission would also allow for development of more missions to different asteroids, providing even more planetary defense capability. What the ARM presents though is short duration crewed missions to this asteroid. To bring the asteroid back is going to take years, but because the object is in orbit around the moon, multiple crewed missions to and from the moon can take place over a few days or weeks as opposed to months or years. While these short duration missions are what we are looking for, the lack of an original destination make the ARM not very beneficial for our space exploration goals when compared to a crewed mission to an asteroid itself.

Another mission currently being developed by NASA is the Origins Spectral Interpretation Resource Identification Security-Regolith Explorer, or OSIRIS-REx, mission. This uncrewed mission will simply arrive at a Near Earth Asteroid (NEA), Bennu, and return a sample to Earth for study. Projected to launch in 2016, arrival at the target is anticipated for 2018 and return to Earth in 2023 [14]. The mission itself is a great way to learn more about asteroids and their composition. It is a representation of *Mission Type 2* with an uncrewed visit to Bennu. In terms of safety, not having any humans on board makes the mission much safer but still very expensive and time consuming. Unfortunately, a 7 year mission, even if it is successful, is what this research is looking to avoid.

Mars Direct is a proposal developed by Mr. Robert Zubrin that is an in-depth look at a mission to Mars. He details a launch of supplies early on that will bring the "Earth Return Vehicle" [15]. An early "prescursor mission," as it is often called, has been debated by many in missions to asteroids. It would only be needed with *Mission Type 3* where a crewed visit would follow. Zubrin, and most other Mars mission proposers, include precursor missions because without them it will be much more difficult to accomplish their goals in a reasonable amount of time. The Sabatier reaction will be used for in-situ resource production combining supplied hydrogen with carbon dioxide from the Martian atmosphere to create methane and oxygen, giving the future crewed mission enough fuel to get home [15]. This future crewed mission will include four astronauts setting out on a low-energy trajectory. These low-energy trajectories take advantage of the gravitational pull of the planets and allow for very low ΔV cost, saving a lot of fuel. This trajectory however leads to a 6-month trip to the planet and therefore will not be referenced for a NEA mission. Another method that Zubrin presents to save fuel is the concept of aerobraking. Aerobraking is a method of slowing the spacecraft down enough to enter an orbit about the object

using its atmosphere [15]. An initial burn is used to enter the atmosphere upon arrival, then using the atmosphere to slow down, the spacecraft will enter an elliptical orbit. Upon each orbit around the object, the spacecraft will enter the atmosphere at its orbital periapsis, which is the desired eventual orbital altitude, and slow down enough to lower its elliptical apoapsis, or high point in the orbit. This process will continue over the next many weeks or months until an apoapsis of the orbit drops to the desired circular orbit altitude. Once this occurs, a burn will be performed at the apoapsis point to raise periapsis so as to not enter the atmosphere again. This process saves a great amount of fuel but leads to high heat build-up on the spacecraft that will need to be dissipated. The method of aerocapture is very similar but involves going deeper into the atmosphere to achieve the desired apoapsis altitude in one shot, taking only a few hours or days as opposed to weeks and months [16]. This method would be ideal for the hyperbolic orbits that we are considering, however the lack of atmosphere around asteroids and NEOs make this method extremely difficult. Therefore, a rendezvous fuel burn slowing us down to the desired orbital velocity, will be required.

2.3 Uncrewed Mission Proposals

The following proposed missions are *Mission Type 2* missions where probes or uncrewed spacecraft are sent to study the asteroids on their own. While safety is a bit less of a concern, the ability to automate a mission to an asteroid is extremely difficult and time consuming. The Discovery program instituted by NASA accepts proposals to unknown space bodies for future research and exploration. Numerous proposals have been submitted but the major ones to asteroids are summarized below. First, the Advanced Jovian Asteroid Explorer (AJAX) mission involves visiting a Trojan asteroid using orbital and landed elements. The mission will characterize

geological properties and chemistry of a Trojan asteroid for the first time [17]. The Binary Asteroid in-situ Explorer (BASiX) is another mission with a goal of turning a NEO "into a geophysical laboratory to study rubble-pile asteroids" [18]. Mission designers desire to create craters in the asteroid by setting off small explosions and then sensing those blasts to measure seismic shaking. The proposed spacecraft would have minimum ΔV capability so the time to reach the asteroid is upwards of 19 months, making BASiX a long duration mission [18]. Finally, the Dark Asteroid Rendezvous (DARe) mission involves visiting numerous asteroids at different altitudes to study "planetary migration and plantesimal scattering" [19]. In regards to scientific study, these missions will be greatly helpful in exploring different asteroids and developing potential planetary defense techniques. In regards to *Mission Type 1* missions, really any of these probes could be used to perform a fly-by of an asteroid. The incredible cost and time required to develop a fly-by mission does raise questions as to the value of such a mission. If we are going to expend the time and money to make it to an asteroid, it makes sense to slow down and visit it, rather than simply pass by it. However, should a probe or automated part of the mission become unusable after launch, a fly-by makes complete sense as to not waste the resources and time it took to get up into space. In terms of successful completed missions, the Japanese Hayabusa mission made it to the asteroid Itokawa on a 7 year mission from May 2003 until return in June 2010 [20]. A lengthy two year outbound leg was completed with continuous ion engine acceleration. Measurements and data taken from orbit helped lead to two landings on the asteroid in late 2005, and the return trip began in April 2007 [20].

2.4 Crewed Mission Proposals

Since the era of the Apollo missions, scientists and engineers have been developing proposals to NEOs in hopes of creating the next great NASA program. The discussion of these missions will be analyzed based on their mission length in the following subsections. Similar to the objectives discussed in Section 1.5, the proposals found potential targets and then developed missions to said targets based on possible trajectories.

2.4.1 Long Duration Missions (More than 8 Months)

In 1977, John Niehoff discussed three requirements that he created to describe an ideal NEO. These requirements were for the asteroid to have an orbital period of close to 1 year, be in an almost circular orbit, and be in an orbit close to coplanar with the ecliptic [21]. The thinking here is that the more similar the asteroid is to the Earth, the easier and quicker a mission would be. There are two ways to select a possible NEO when it comes to crossing the Earth's orbit. Either the two have similar orbits and very low inclination differences, presenting a possible mission to aphelion or perihelion, or the asteroid will cross the Earth's orbit at its ascending or descending node. Oftentimes, the missions to a node, though higher in ΔV, present a faster option than a mission to aphelion or perihelion. For Niehoff, using these criteria he was able to choose two different NEOs as potential candidates, though he did specify that neither met all of his criteria but rather that these were the two closest options. The objects chosen were 1976 AA and 1973 EC. Missions to 1976 AA were analyzed for a potential early 1990s mission, but time for the mission crept up to about 1 year in duration. A five month outbound transfer was decided in order to arrive at 1976 AA at its aphelion. This would minimize the amount of ΔV necessary for the mission. After a 30 day stay on the asteroid, a 6 month return trajectory

would take place, returning the astronauts exactly 1 year after departure [21]. For the object 1973 EC, another minimum energy trajectory was planned leading to a potential mission duration of approximately 3 years. Due to the high length of these missions, the flight profiles and trajectories are not ideal for a short duration mission.

Shoemaker and Helin delved into a similar area a year later, describing why and how to study asteroids. They, like Niehoff, felt that a low eccentricity and low inclination orbit is ideal to allow for low ΔV trajectories in somewhat short periods of time. 1976 AA, though its inclination is high (19 degrees), presents a good candidate for exploration because of its proximity to the Earth. The authors discussed whether it is ideal to rendezvous at perihelion or at aphelion [22]. Due to higher velocities at perihelion though, it is clear that for a minimum ΔV, rendezvous should occur at aphelion. The mission lengths and trajectories were very similar to Niehoff's studies and in turn were too long for missions presented in this research. Shoemaker and Helin focused on ballistic trajectories with no desire to minimize time. Therefore, again their results do not lend themselves to a short duration mission. These missions were proposed before the use of more recent technology, leading the authors to reference Niehoff's estimate of 28 Shuttle launches needed to reach 1976 AA. This, of course, is not necessary in today's day and age but is an interesting side note [22].

Many proposals are subject to constraints developed by the design team or modern technology. A group of Chinese scientists, Rui Xu, Pingyuan Cui, Dong Qiao, and Enjie Luan developed a mission using the following list of requirements [23]:

1. The interval 2009-2010 should be used for launch dates

2. Orbit of target must be well determined

3. Less than 3 and a half AU is necessary for target aphelion

4. ΔV upon departure cannot exceed 2 km/s

5. Total ΔV upon rendezvous cannot exceed 6 km/s

Meeting these constraints, an asteroid (1943 Anteros) was chosen. Optimization algorithms were used in order to determine the minimum ΔV trajectories to and from the asteroid, which produced long mission durations. The proposal did however talk about the useful topic of gravity assist. This technique involves using a planet's gravity to increase spacecraft velocity and use less energy on its way to a farther planet or object [23]. The same authors also developed a multi-year mission to the asteroid Ivar. Their mission included the use of a 2:1 ΔV-EGA (developed from ΔV, Earth Gravity Assist [10]) trajectory that utilizes a deep space maneuver in order to approach the Earth and gain a gravity assist, seen in Figure 5, reproduced from Dong Qiao et al., Beijing Institute of Technology. Gravity assist, though interesting, will not be considered for the missions presented as there is no need for a gravity assist from Earth or another planet.

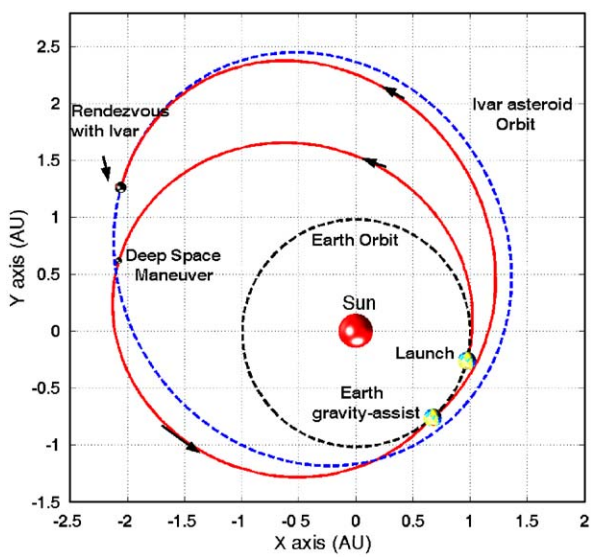

Figure 5. Possible trajectory for multi-year mission to Ivar. Reproduced from Dong Qiao et al., Beijing Institute of Technology [24].

2.4.2 Medium Duration Missions (1-8 Months)

Perhaps the most similar mission to what is being presented in this research was developed by a team of scientists, led by Dave Korsmeyer from the NASA Ames Research Center. They describe an initial precursor mission to a NEO in order to obtain simple reconnaissance, gain information of the target surface, and even help the crewed mission navigate to the NEO through a transponder on the surface [4]. As described in Section 2.4.1, many authors consider the best targets for a mission to be low eccentricity, low inclined orbits similar to Earth. An interesting statistic pointed out is that every degree of inclination adds another 0.5 km/s to the post-escape ΔV, which makes selection of the right NEO very important. However, should the mission plan on utilizing the crossing point of the asteroid at one of its nodes, this inclination difference should all but be eliminated. The trajectories developed create 90 day missions that involve either departure from Earth while the asteroid is closest or arrival back to Earth when the asteroid is closest, making the proposed mission one long leg and one short leg [4]. The target asteroid, 2000 SG344, projected mission can be seen in Figure 6 and Figure 7, reproduced from David Korsmeyer et al., NASA Ames Research Center.

A similar mission was described by Lockheed Martin to the NEO 2008 EA9. The mission differs in that it involves an outbound transfer orbit that takes three months but arrives at the asteroid while it is closest to the Earth. Then, after a five day stay at the object, a roughly three month return transfer trajectory is used to return to Earth. This does not constitute a short mission through arrival or departure when the asteroid is closest but rather uses this close point to visit, thus lengthening the trajectories to and from the asteroid [6]. The projected mission plan can be seen in Figure 8, reproduced from Josh Hopkins et al., Lockheed Martin. Compared to the Korsemeyer missions above, the ΔV in this mission is lower, but as stated in Section

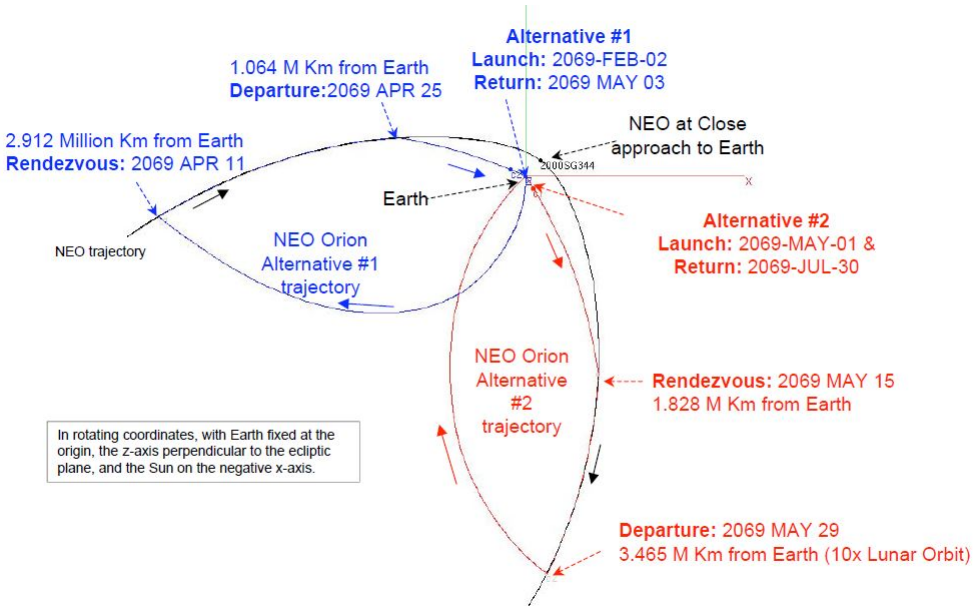

Figure 6. With Earth in the center, blue details a trajectory with arrival close to Earth. Red details a trajectory with departure close to Earth. Reproduced from David Korsmeyer [4].

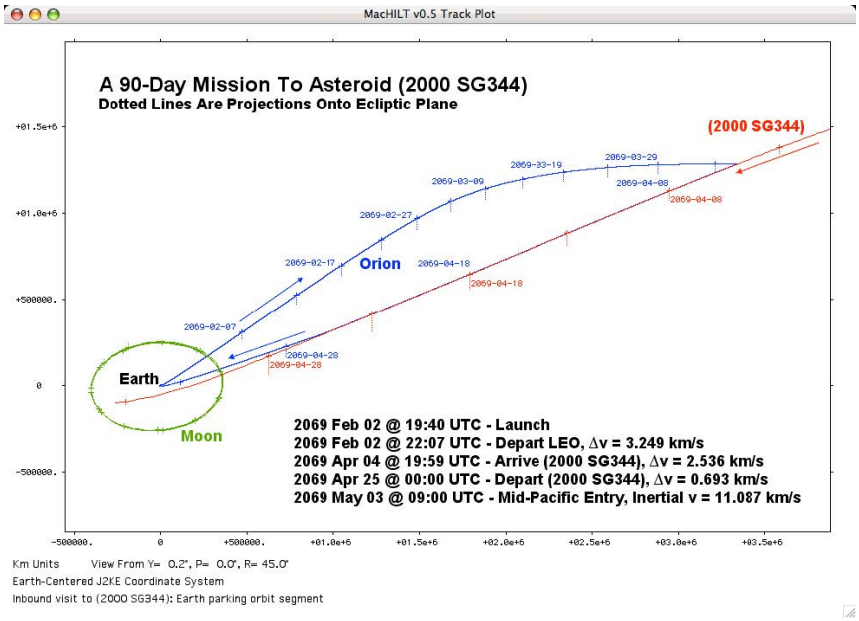

Figure 7. Possible trajectory for 90 day mission to 2000 SG344. Reproduced from David Korsmeyer et al., NASA Ames Research Center [4].

1.1, duration of the mission is the most important factor. A very similar mission trajectory was developed by a group of Spanish scientists, Jesus Gil-Fernandez, Raul Cadenas-Gorgojo, and Diego Escorial-Olmos, who developed an 8 month mission that includes long outbound and return legs with minimal stay on the asteroid itself [25].

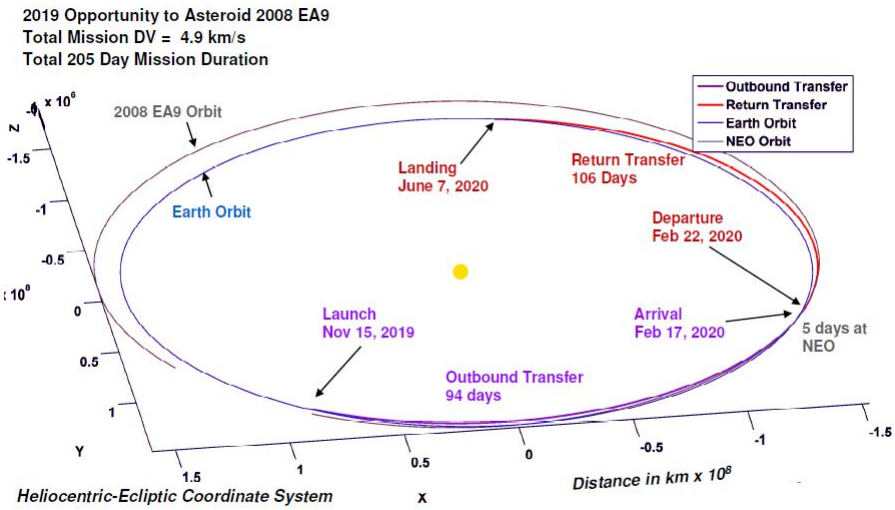

Figure 8. Possible trajectory for 205 day mission to 2008 EA9. Reproduced from Josh Hopkins et al., Lockheed Martin [6].

In an attempt to demonstrate the simplicity of a two month mission plan to a NEO, Jones et al. developed a 60 day mission that included a 30 day stay on the asteroid 1991 VG. The authors decided on a 15 day outbound and return trip time and used that to calculate transfer hyperbolas. The calculated ΔV for a projected 60 day mission from a LEO starting point is only 6.1 km/sec [26]. Unlike Lockheed Martin, the authors developed a mission plan that arrives prior to the close approach point (15 days) and departs the same amount of time after this point, seen in Figure 9, reproduced from Jones et al. [26]. This proposal relates to the presented research in that rather than launching a mission to a NEO from Earth, like the Apollo missions, they decide to start from LEO in the same orbit as the ISS. They also use aerobraking

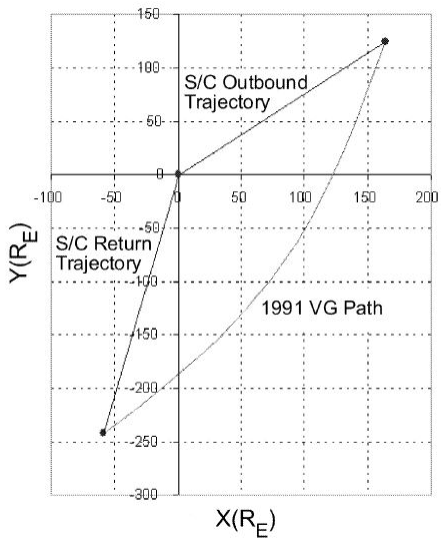

Figure 9. Possible trajectory for mission to asteroid 1991 VG as the asteroid travels in a retrograde orbit as seen from Earth. Reproduced from T.D. Jones et al. [26].

techniques to return to LEO, rather than arriving back on Earth. Surprisingly, though the mission trajectories are discussed from LEO, the authors recommend full assembly on the ground as "We have learned from ISS experience that it is cheaper and less risky to integrate and test a spacecraft on the ground before launch than to perform major assembly in orbit..." [26]. Finally, they even discuss the possible use of the L2 Lagrange point on the far side of the Earth from the sun, for the use of stability while potentially rendezvousing mission pieces together [26].

Using MATLAB® algorithms, two Stanford students, Cyrus Foster and Matthew Daniels, computed ΔV values and trajectories for 90-day, 180-day, and 365-day missions to different NEOs. These results provided a table describing how many rendezvous mission opportunities are available for each mission length and how much ΔV each would cost [27]. While the idea of proposed mission times helps to develop the trajectories, we are looking to minimize those mission times at the cost of a somewhat increased ΔV. Daniel Zimmerman, Sam Wagner, and Bong Wie had a similar method in setting proposed mission durations and basing their ΔV research on that.

Their ideas focused on target selection and which asteroid would be the most beneficial to go visit [28]. This is very important in the mission development process. If the selected asteroid does not have a conducive trajectory to a short duration mission, the ΔV costs will be much too high. Wagner and Wie, in another study, developed the following basic list of requirements for choosing an NEO to visit [29]:

1. No comets will be considered, only asteroids
2. Minimum ΔV must happen near a close approach to Earth (They used 0.2 AU away)
3. Low eccentricity and inclination
4. Must be slow rotating and a single object

A similar approach discussed in Section 1.3 will be used to minimize the number of NEOs in the catalog.

These same authors developed a set of mission designs to the asteroid Apophis in 2029 [30]. Like other missions, their designs considered mission lengths of 180 and 365 days. Setting mission duration times allows for the consideration of many ΔV possibilities. They also discussed early and late mission trajectories. Early involved departing Earth while still very distant from the asteroid and using the close approach for the return trip (as seen in Figure 10), while the late trajectory arrives at the asteroid at the close approach date and has a longer return trip, which can be seen in Figure 11, both reproduced from Sam Wagner et al., Iowa State.

Though intensely studied and developed, only Jones et al. presented possibilities that relate to the missions proposed in this research. No missions have been created that are on the order of a few days or weeks. A major reason for this is the desire to save ΔV by approaching at aphelion of the orbit as opposed to a quick out and back trip when the asteroid is close to the Earth. For many of the missions proposed in

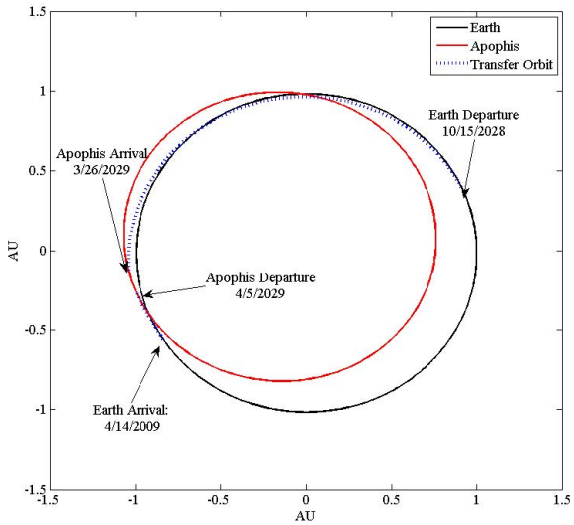

Figure 10. Early departure trajectory to Apophis. Reproduced from Sam Wagner et al., Iowa State [30].

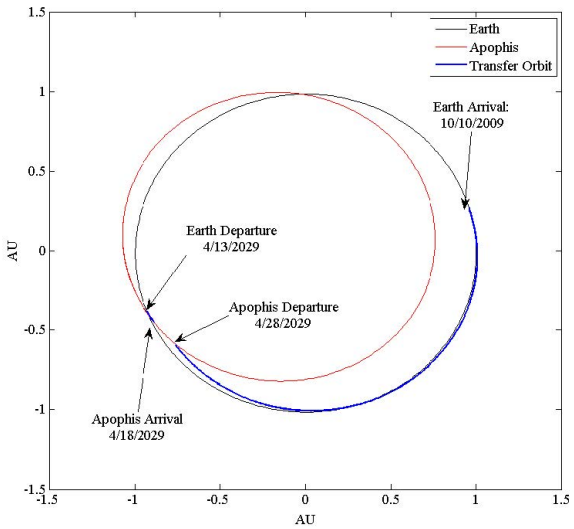

Figure 11. Late departure trajectory to Apophis. Reproduced from Sam Wagner et al., Iowa State [30].

this chapter, time was not the number one factor. However, the research presented in this thesis utilizes a minimum time approach to identify the best candidates for a short duration mission. The development of these short duration missions could open up countless capabilities in the realm of space for crewed or uncrewed missions.

3. Methodology

In this presented research, we will identify and analyze short duration mission candidates. This chapter will present the necessary equations and discuss how to use them in order to propagate the Earth and the asteroid and figure out when and where they come close. From here, mission ΔV and mission duration will be evaluated. All of these topics and their corresponding calculations are covered in the following sections.

3.1 Asteroid Determination

In general, there are two major determinations that need to take place in order to develop a mission to an asteroid. The first is the position of the asteroid when it crosses into the Earth's orbit. The second part is to find the position of the Earth while the asteroid is at that crossing point. This involved finding close approach dates when the asteroid and Earth were within certain tolerances at the crossing point. We will start by computing when and where the asteroid crosses the orbit of the Earth.

3.1.1 NEO Catalog

NASA's Jet Propulsion Laboratory has a "JPL Small-Body Database Search Engine" available to the public [31]. The search engine allows users to create outputs based on certain constraints. For this presented research, the search was limited to "NEOs" as the object group and "Asteroids" as the object kind. This research is concerned with missions to Earth-crossing asteroids, and there are no limitations to what this means. Therefore it was decided to keep the constraints and limitations very basic as to allow for exploration of the largest set of missions. After choosing what object was desired, the output profile was chosen to yield the necessary orbital

and object parameters. The output fields for each NEO asteroid were:

1. Object Full Name/Designation
2. JD, Epoch of Osculation (Julian Day)
3. MJD, Epoch of Osculation (Mean Julian Day)
4. e, Eccentricity
5. a, Semi-major Axis (AU)
6. i, Inclination (deg)
7. Ω, Longitude of the Ascending Node (deg)
8. ω, Argument of Perihelion (deg)
9. r_p, Perihelion Distance (AU)
10. r_a, Aphelion Distance (AU)
11. M, Mean Anomaly (deg)
12. n, Mean Motion (deg/day)
13. P, Orbital Period (days)
14. P, Orbital Period (years)
15. Data Arc Span (days)
16. OCC (Orbital Condition Code) (How well the orbit is defined from 0-9)
17. Number of Observations Used
18. H, Absolute Magnitude Parameter

19. d, Diameter (km)

20. P_{rot}, Rotational Period (hrs)

The most important of these values are the 6 orbital elements (COEs) that define the orbit of the asteroid (mean anomaly is considered a COE because it can be converted to true anomaly using eccentricity). From the COEs, one can estimate the crossing point of the asteroid's orbit with the Earth's orbit. In order to properly use this data, it was necessary to transfer it into MATLAB® from the Excel `.csv` file that was created from the JPL website. The "Import Data" command in MATLAB® was used to generate a script that pulled in each output field described above and create its own column vector storing that information. The standard gravitational parameter of the Sun, $\mu_{\odot}$, is also output by this script for later calculations.

3.1.2 Asteroid Crossing Point

Now that all of the asteroid orbital data is input, the next step is to find where each asteroid crosses the Earth's orbit. Figure 12 shows us that there are generally two possibilities where the asteroid will intercept the orbit of the Earth. Though Figure 12 illustrates more of an Aten type asteroid, an Apollo type asteroid crossing point follows the same calculation algorithm. In determining these two possible true anomaly values, we can use the polar form of a conic section with the origin at one focus, Equation (4) [10].

We need to solve Equation (4) for ν and therefore need to define r. Since the distance of the asteroid at the crossing point is from the Sun to the Earth's orbit, r can be defined as 1 AU. This will output two values and in order to find which ν is correct, a rotation to Inertial coordinates must take place. If we know r and ν, which we do, we can write an expression for $\boldsymbol{r}$ in terms of $\boldsymbol{P}$, $\boldsymbol{Q}$, $\boldsymbol{W}$ or Perifocal coordinates. Eventually, we will have two values for $\boldsymbol{R_{cross}}$ because we currently have two values

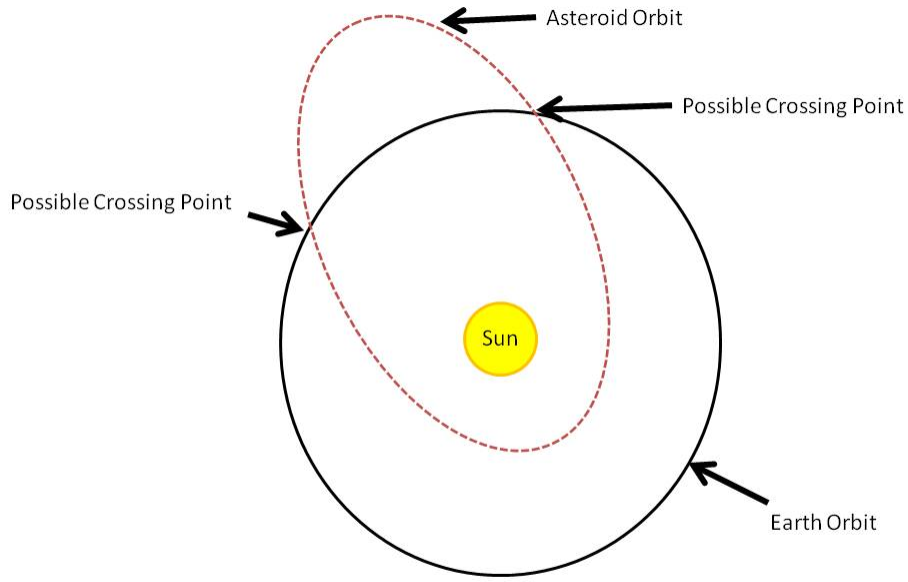

Figure 12. Simple diagram (not to scale) of the crossing point possibilities of an asteroid.

for ν.

$$\boldsymbol{R_{pqw}} = r\cos\nu\boldsymbol{P} + r\sin\nu\boldsymbol{Q} \tag{6}$$

The best way to determine which true anomaly value to use is to rotate these two Perifocal vectors to the Inertial $\boldsymbol{I}$, $\boldsymbol{J}$, $\boldsymbol{K}$ frame. The Perifocal frame is the Inertial frame rotated about the right ascension of the ascending node (Ω), the inclination (i), and the argument of perihelion (ω) of the orbit of the asteroid. This rotation can be seen in Equation (7).

$$\boldsymbol{R_{cross}} = R(\Omega, i, \omega)\begin{pmatrix} r\cos\nu \\ r\sin\nu \\ 0 \end{pmatrix} \tag{7}$$

Where

$$R(\Omega, i, \omega) = \begin{bmatrix} \cos\omega\cos\Omega - \cos i\sin\omega\sin\Omega & -\cos\Omega\sin\omega - \cos\omega\cos i\sin\Omega & \sin i\sin\Omega \\ \cos\omega\sin\Omega + \cos i\cos\Omega\sin\omega & \cos\omega\cos i\cos\Omega - \sin\omega\sin\Omega & -\sin i\cos\Omega \\ \sin\omega\sin i & \cos\omega\sin i & \cos i \end{bmatrix}$$

Now that we have our two position vectors in the Inertial frame, we can compare the $\boldsymbol{K}$ values. Whichever value is smallest (closest to zero) corresponds to the true anomaly value that we want to use. By computing the crossing point closer to the ecliptic, we minimize the $\boldsymbol{\Delta}V$ necessary to reach the asteroid.

After the location is known where the asteroid is going to cross the Earth's orbit, the next step is to compute the velocity vector of the asteroid. Here, we will need to use the Perifocal reference frame again to solve for $\boldsymbol{V}$ and then rotate back to the Inertial frame. The equation to define position in the Perifocal frame is already defined in Equation (6), and the rotation back to the Inertial frame can be seen in Equation (7). The process is the same for the velocity vector, but we will instead use Equation (8) to move to the Perifocal frame.

$$\boldsymbol{V_{pqw}} = \begin{bmatrix} -\sqrt{\frac{\mu_\odot}{p}} sin\nu \\ \sqrt{\frac{\mu_\odot}{p}}(e + cos\nu) \\ 0 \end{bmatrix} \tag{8}$$

Where

$$p = a(1 - e^2)$$

Since a is defined in AU, we must convert it to kilometers to calculate a p value in kilometers and eventually a velocity vector in kilometers per second. The value used for an AU was 1 AU $= 149,597,870.700$ km. Finally, the same rotation matrix

from Equation (7) is multiplied by the vector from Equation (8) to obtain $\boldsymbol{V_{cross}}$. To recap, we now have 20 pieces of information: the asteroid's orbital elements, the orbital elements of the Earth, the position of the asteroid in its orbit where it crosses paths with the Earth's orbit and its velocity vector in the inertial frame when it crosses.

The next step is to estimate the dates when the asteroid will be at the crossing point. Due to the nature of elliptical orbits, propagating the true anomaly value linearly over a given time will not yield the correct true anomaly value later in time. This is because of Kepler's 2nd Law which states that the radius vector sweeps out equal areas over equal times [10]. Knowing this, Kepler developed a way to convert an elliptical orbit to a particular auxiliary circle. The key to the circle is knowing that a given area in the ellipse is smaller than the same given area in the auxiliary circle by the ratio of minor to major axes. Using this circle, we can go from a true anomaly to an eccentric anomaly and then to the mean anomaly. The mean anomaly can be calculated from the true anomaly and propagated properly over a period of time. To move from the true anomaly of the crossing point to the eccentric anomaly, E, we use Equation (9).

$$E = 2\tan^{-1}(\sqrt{\frac{1-e}{1+e}}\tan\frac{\nu}{2}) \tag{9}$$

This equation eliminates any potential quadrant errors. Once the eccentric anomaly is computed, the mean anomaly can be calculated.

$$M = E - e\sin E \tag{10}$$

Where

$$E = \text{eccentric anomaly (in radians)}$$

Now, after converting radians to degrees, we have the mean anomaly at the crossing point. This can be compared to the mean anomaly value that was pulled from the NEO catalog on the given Julian day. Taking the crossing point mean anomaly and subtracting the given mean anomaly, the distance that needs to be traveled in the orbit can be calculated. To estimate how long this will take, we use the mean motion of the asteroid and Equation (11).

$$\Delta t = \frac{M_{dist}}{n} \tag{11}$$

Where

M_{dist} = difference between crossing point mean anomaly and given mean anomaly (in deg)

Δt = time from epoch until first crossing date (in days)

This Δt value is equal to the number of days from the epoch time until the first crossing date. Adding Δt to the original Julian day value will give the Julian day for the first crossing of the Earth's orbit. This Julian day can be converted back to a year:month:day:hour:minute:second format to produce an official date of the crossing. In this research, we compute every crossing date up until the year 2080. Allowing this many crossing dates to be given yields many opportunities and potentially lets us plan for missions many years down the line. A loop was developed to add the period of the asteroid (in days) to the crossing Julian day value for every crossing point up until 1 Jan 2080, creating a matrix of dates when the asteroid crosses the Earth's orbit.

3.2 Earth Determination

We will now look at the position of the Earth on these asteroid crossing dates and output another matrix containing only the dates where the Earth is in close range when the asteroid passes by. Since we already have the asteroid crossing dates, we can compute where the Earth is in inertial space of the Sun-Earth 2BP. First, a file was written to extract the following orbital elements for the Earth.

1. $JD_{\oplus}$, Epoch of Osculation (Julian Day)
2. $e_{\oplus}$, Eccentricity
3. $a_{\oplus}$, Semi-major Axis (AU)
4. $a_{\oplus km}$, Semi-major Axis (km)
5. $i_{\oplus}$, Inclination (deg)
6. $\omega_{\oplus}$, Argument of Perihelion (deg)
7. $rp_{\oplus}$, Perihelion Distance (AU)
8. $ra_{\oplus}$, Aphelion Distance (AU)
9. $M_{\oplus}$, Mean Anomaly (deg)
10. $n_{\oplus}$, Mean Motion (deg/day)
11. $P_{\oplus}$, Orbital Period (yrs)
12. $Rot_{\oplus}$, Rotational Period (hrs)

It is important to recognize that the Julian day reference for the Earth is different than that of the asteroid so in order to determine mean anomalies and $\Delta t's$, we must estimate the mean anomaly of the Earth on the asteroid's Julian day. Computing the

difference between the two Julian day values yields a Δt value in days that can be multiplied by the mean motion of the Earth and added to its given mean anomaly. This provides the mean anomaly of the Earth for the asteroid's Julian day but this is not the same day that the asteroid crosses the Earth's orbit. Therefore we can take the Δt value from Equation (11), multiply that by the mean motion of the Earth and add that to the mean anomaly we just computed. This will yield the mean anomaly of the Earth at the first asteroid crossing date.

To calculate the Earth's mean anomaly at each crossing date, we can take the period of the asteroid, P_{ast}, multiply it by the mean motion of the Earth, $n_{\oplus}$, and add to the previous mean anomaly. This will occur for every crossing date prior to 2080. For comparison purposes, we now need to calculate the mean anomaly of the Earth at the crossing point to see when that value and the mean anomaly on the crossing date are within a certain tolerance, 10°. This process is presented in Equation (12).

$$M_{i+1} = P_{ast} n_{\oplus} + M_i \tag{12}$$

Where

M_i = mean anomaly of the Earth on the date the asteroid's orbit crosses the Earth

Looking at Figure 13 we see that the position vector of the asteroid at the crossing point can also be used for the Earth since the crossing point is a position on both bodies' orbits. The $i - j$ plane defines the ecliptic plane. The inverse tangent of the j component over the i component yields the angle between the first point of Aries and the crossing point, or the true longitude, $\ell_{\oplus}$ of the orbit when the Earth is at the crossing point. This angle is equal to the right ascension of the ascending node

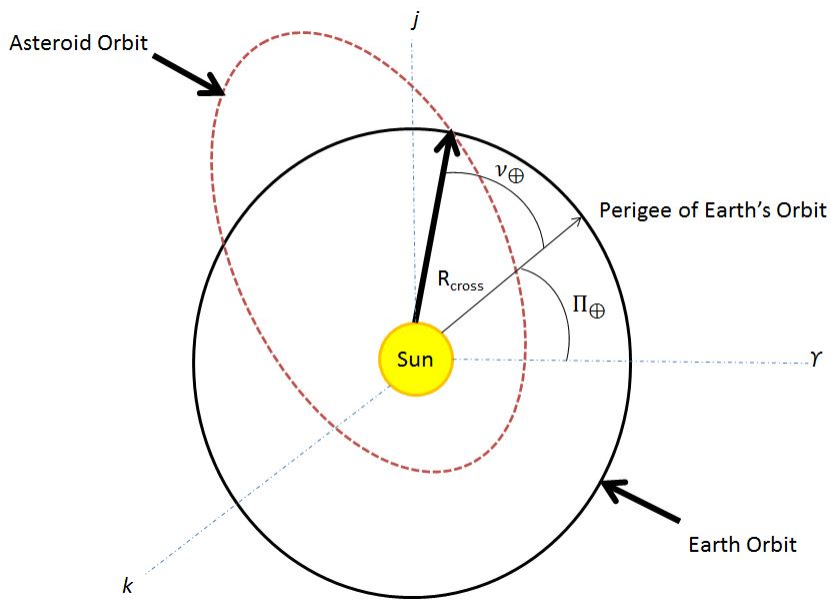

Figure 13. Simple diagram (not to scale) of the way to calculate true anomaly for the Earth at the crossing point.

of the Earth's orbit, plus the argument of perigee of the Earth's orbit, plus the true anomaly. Since the Earth's orbit inclination is 0°, $\Omega_\oplus$ is undefined. Therefore we use longitude of periapsis, $\Pi_\oplus$, which is the angle measured from $\hat{\boldsymbol{I}}$ to periapsis. This value is defined as $\omega_\oplus$ above, and now we can compute $\nu_\oplus$ at the crossing point.

$$\ell_\oplus = \Omega_\oplus + \omega_\oplus + \nu_\oplus = \Pi_\oplus + \nu_\oplus \tag{13}$$

We know the argument of perihelion for Earth, allowing a simple subtraction to yield the true anomaly of the crossing point in the Earth's orbit. From here we can use Equations (9) and (10) to compute the mean anomaly of the Earth at the crossing point. By defining a simple if/then loop, we can see if the difference between this value and any of the M_i values is less than 10 degrees on each Julian day that the asteroid crosses the Earth's orbit. If so, another matrix is constructed with the more restrictive crossing dates included. From here we can calculate the position vector of the Earth on the crossing date using Equations (6) and (7) to go along with our

position vector of the asteroid at the crossing date. Finally, another constraint is in place to determine the dates that lead to very close approaches. Another if/then loop is used after the first one that requires the magnitude of the difference between the two position vectors to be less than 7 million km. Finally, we end up with a matrix of close approach dates that has met two rounds of constraints: mean anomaly and distance.

These close approach dates, though accurate, do not tell the full story of a possible close approach. Due to perturbations that have not been accounted for and possible errors in COEs of the asteroids, the close approach date may not be the closest position of the asteroid to the Earth. There are essentially two ways to account for the possibility of a different closest approach date; both start by utilizing a close approach date estimated through both rounds of constraints discussed above, and propagating that date 30 days forward and backward in time in three day increments. The first method is to propagate the Earth and the asteroid forward and backward in time to compute the closest approach. The second method is to keep the Earth fixed and propagate the asteroid a month forward and backward to account for multiple errors in the period of the orbit. The first method assumes correct propagation and positions of the Earth and asteroid, and the movement is only to compute the closest point. The second method assumes an error in the propagation or orbital data of the asteroid. We can do the first method by taking the mean anomaly of the Earth and the asteroid (at the crossing point) and add or subtract 30 times the mean motion value, or 27 times, then 24 times, etc. With this new mean anomaly, a new true anomaly can be calculated using Newton's method to solve for eccentric anomaly and from there using Equation (9) solved for ν to move to true anomaly. Finally, using the new true anomalies, a matrix of position and velocity vectors can be created using Equations (6), (7), and (8). Now, by simply taking the difference between the two

matrices we can see the closest point of the asteroid to the Earth over a 60 day period. The second method involves the same calculations but only for the asteroid, while the Earth remains fixed in the same position as it would be based on the solution found in method one. More about these methods will be discussed in Section 4.2.

The final task we need to complete with the Earth is use its position, $\boldsymbol{R}_{\oplus}$, and velocity, $\boldsymbol{V}_{\oplus}$, as well as the position and velocity of the asteroid in reference to the sun in order to go from a Heliocentric frame to a Geocentric frame, for our trajectory planning. Using these position and velocity values, we can create a rotation matrix, R_{HtoG}.

$$R_{HtoG} = \begin{bmatrix} \hat{\boldsymbol{R}}_{\oplus} & \hat{\boldsymbol{x}} & \hat{\boldsymbol{h}}_{\oplus} \end{bmatrix} \tag{14}$$

Where

$$\boldsymbol{h}_{\oplus} = \boldsymbol{R}_{\oplus} \times \boldsymbol{V}_{\oplus}$$

$$\boldsymbol{x} = \boldsymbol{h}_{\oplus} \times \boldsymbol{R}_{\oplus}$$

$$\boldsymbol{h}_{\oplus} = \text{angular momentum vector of the Earth's orbit}$$

3.3 Delta V Computations

We will analyze each asteroid mission in 4 segments, including three main fuel burns and a reentry leg. Each segment is represented by the letters A, B, C, and D in Figures 14 and 15. The letter A corresponds to the outbound ΔV, ΔV_{out}, which will be used to send us to the asteroid. Letter B represents the rendezvous ΔV, ΔV_{rend}, that will be used to match velocities with the asteroid and allow us to perform our research mission. The divert ΔV, ΔV_{divert} is represented by the letter C. This burn

is the smallest of the three burns, and puts us on a trajectory back to Earth. Finally, the letter D shows the Earth reentry leg. For this section, since we are referencing the asteroid from the frame of the Earth, we will use equations with the Earth as the central body, as opposed to a Heliocentric sun frame.

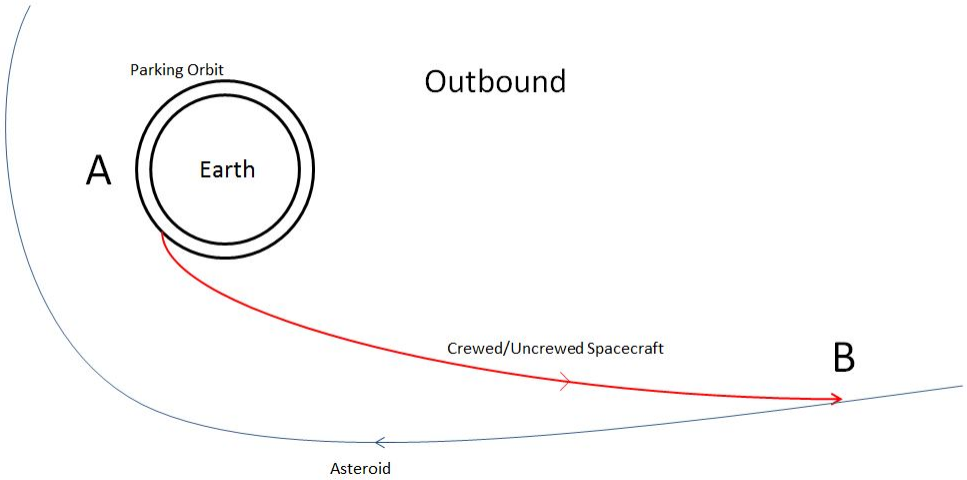

Figure 14. The approximated outbound trajectory from a parking orbit to the asteroid.

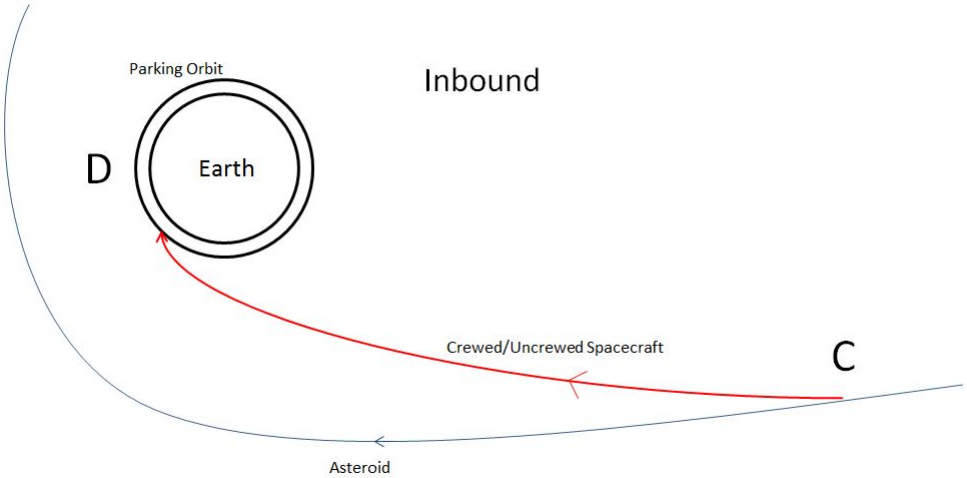

Figure 15. The approximated inbound trajectory from the asteroid back to Earth reentry.

3.3.1 Outbound ΔV

Determining the outbound ΔV is a simple process once some parameters are set. The first thing to estimate is what altitude is desired for our parking orbit in LEO. For our mission, 250 km was chosen as our altitude. Adding this to the common value for radius of the Earth ($r_{\oplus} = 6378.137$ km) we calculate the radius of the parking orbit, r_{park}, to be 6628.137 km. Now that we have r_{park}, we can introduce a more specific form of Equation (5), the vis-viva equation to be used for the departure trajectory.

$$\varepsilon_{ob} = \frac{V_{esc}^2}{2} - \frac{\mu_{\oplus}}{r_{park}} \tag{15}$$

On the outbound trip we are leaving the Earth's sphere of influence and approaching, for the purpose of these calculations, an infinite distance away from the Earth. V_{esc} is the velocity needed to reach 'just beyond' this sphere of influence and allow the spacecraft to continue on a parabolic trajectory. A hyberbolic trajectory is developed when there is more velocity than just the necessary escape velocity and the velocity at 'infinity' is greater than zero. At this 'infinite' distance away from Earth, ε is zero, leaving us with the following equation [11]:

$$V_{esc} = \sqrt{\frac{2\mu_{\oplus}}{r_{park}}} \tag{16}$$

Equation (16) allows us to compute an escape velocity necessary to escape the Earth's sphere of influence. The next step is to calculate a burnout velocity, V_{BO}. This velocity combines V_{esc} with V_{extra}, where V_{extra} is a value dictated by the mission. If we want to reach the asteroid faster, we add ΔV at departure. However, if the parabolic trajectory gets us to the asteroid in a reasonable amount of time, then V_{extra} can remain equal to 0 km/s and is not necessary. Our speed at 'infinity', or when we leave the sphere of influence, is V_{∞}. Knowing that ε remains constant along an orbit,

ε at the departure point can be compared to ε at 'infinity' [11].

$$\varepsilon_{ob} = \frac{V_{BO}^2}{2} - \frac{\mu_\oplus}{r_{park}} = \frac{V_\infty^2}{2} - \frac{\mu_\oplus}{r_\infty} \tag{17}$$

Since r_∞ is so large, that term can cancel to zero. Now rearranging to solve for V_∞, Equation (18) is used [11]. The equation simplifies because of the definition of escape speed from Equation (16).

$$V_\infty^2 = V_{BO}^2 - \frac{2\mu_\oplus}{r_{park}} = V_{BO}^2 - V_{esc}^2 \tag{18}$$

Equation (18) shows that as V_{BO} increases with an added boost at departure, the speed at infinity will increase as well. Therefore, ideally we do not want to add any extra ΔV to our outbound trajectory. The greater V_∞, the more rendezvous ΔV is necessary to match velocities with the asteroid. This will be covered more when we talk about ΔV_{rend} in Section 3.3.2. Now that we have V_{BO} though, we can determine the COEs of our parabolic trajectory to the asteroid. If V_{extra} is 0 km/s, then V_{esc} is the same as V_{BO}. Going back to the vis-viva equation, Equation (5), we can see that this would just mean ε is still equal to zero. Using Equations (19), (20), and (21) we can calculate the semi-major axis, a_{ob}, eccentricity, e_{ob}, and angular momentum value, h_{ob}, of the outbound trajectory.

$$a_{ob} = -\frac{\mu_\oplus}{2\varepsilon_{ob}} \tag{19}$$

$$e_{ob} = 1 - \frac{r_{park}}{a_{ob}} \tag{20}$$

$$h_{ob} = r_{park} V_{BO} \tag{21}$$

In a parabolic trajectory, we should have a semi-major axis value of infinity and an eccentricity value of 1, which is exactly what these equations yield. Simply by looking at Equation (15) we can see that if the burnout velocity is greater than the escape velocity, ε would be positive, indicating a hyperbolic trajectory. If ε is positive, then Equations (19) and (20) show a_{ob} is negative and e_{ob} is greater than 1 which is to be expected for a hyperbolic trajectory.

We now have the parameters of a parabolic or hyperbolic trajectory to the asteroid. The process of computing the outbound ΔV involves the necessary burn to increase speed from our parking orbit to our required trajectory. Equation (22) displays this value and what our outbound ΔV needs to be in order to place the spacecraft on the proper trajectory.

$$\Delta V_{out} = V_{BO} - V_{park} \tag{22}$$

Where

$$V_{park} = \sqrt{\frac{\mu_{\oplus}}{r_{park}}}$$

Knowing the shape of the trajectory also allows us to compute the time of travel for the outbound leg. Using Equations (23) and (24) and multiplying the resulting vectors by the matrix in Equation (14) we can calculate the position and velocity of the asteroid, on the closest approach date, in the Geocentric frame.

$$\boldsymbol{R_{diff}} = \begin{bmatrix} R_{ast,i} - R_{\oplus,i} \\ R_{ast,j} - R_{\oplus,j} \\ R_{ast,k} - R_{\oplus,k} \end{bmatrix} \tag{23}$$

$$\boldsymbol{V_{diff}} = \begin{bmatrix} V_{ast,i} - V_{\oplus,i} \\ V_{ast,j} - V_{\oplus,j} \\ V_{ast,k} - V_{\oplus,k} \end{bmatrix} \tag{24}$$

Using the position vector of the asteroid and by solving Equation (4) for ν we can compute the true anomaly, $\nu_{arrival}$ of the outbound orbit at the arrival. Once the true anomaly value is obtained, there are two options to calculate the eccentric anomaly of the trajectory and it depends on if V_{extra} is zero or not. Parabolic and hyperbolic trajectories have different methods of calculation. If V_{extra} is zero and the trajectory is parabolic, then we use D as the eccentric anomaly value and it is calculated in Equation (25). Authors Bate, Mueller, and White provided Equations (25) through (30).

$$D = \sqrt{p_{ob}} \tan \frac{\nu_{arrival}}{2} \tag{25}$$

Where

$$p_{ob} = \text{semi-latus rectum of a parabolic orbit (km)}$$

The equation for p_{ob} for a parabolic orbit is simply 2 times the radius at periapsis, or the closest point. This would just be the starting point of the parabolic orbit, more simply, the parking orbit. Thus we have:

$$p_{ob} = 2r_{park} \tag{26}$$

Now that we have the parabolic eccentric anomaly, this can be converted to a time of flight value in seconds, which we can easily convert to days by dividing by 86,400 seconds. Equation (27) shows the time of flight value, $\Delta t_{arrival}$, for a parabolic trajectory from a parking orbit to the asteroid at its closest point. Equation (27)

is a slightly altered form of Barker's equation [10, 11]. The change comes from the definition of D, where Barker's equation does not contain the $\sqrt{p_{ob}}$ in its D term, and rather includes this in a slightly different version of Equation (27).

$$\Delta t_{arrival} = \frac{1}{2\sqrt{\mu_{\oplus}}}[p_{ob}D + \frac{1}{3}D^3] \tag{27}$$

Above, the common parabolic case is covered where no extra ΔV is used, thus keeping ΔV_{rend} relatively low, but we still have the possibility of adding some extra ΔV and ending up on a hyperbolic trajectory to the asteroid. The same true anomaly value, $\nu_{arrival}$, is used, so the only thing that changes is the equations. In the equations below, F is the eccentric anomaly value. It is important to remember that, should $\nu_{arrival}$ be between 0 and π, F is positive, and therefore, if $\nu_{arrival}$ is between π and 2π, F should be taken as negative.

$$\cosh F = \frac{e_{ob} + \cos\nu_{arrival}}{1 + e_{ob}\cos\nu_{arrival}} \tag{28}$$

$$F = \ln[\cosh F + \sqrt{\cosh F^2 - 1}] \tag{29}$$

Now with F, we can say that Equation (30), which is really just the hyperbolic version of Kepler's elliptical orbit equation, shows the time of flight value, $\Delta t_{arrival}$, for a hyperbolic trajectory from a parking orbit to the close approach point of an asteroid. Like the value from Equation (27), this value must also be converted to days.

$$\Delta t_{arrival} = \sqrt{\frac{(-a_{ob})^3}{\mu_{\oplus}}}(e_{ob}\sinh F - F) \tag{30}$$

The outbound leg now has a calculated ΔV value based on desire for extra speed, along with a time of flight to the asteroid.

3.3.2 Rendezvous ΔV

As discussed in Section 3.3.1, the ΔV required for rendezvous is very dependent on whether any extra ΔV is added on the outbound leg. Normally it would make sense for any added speed to have a linear impact on V_∞, but as we can see in Equation (18), the added speed will have a quadratic effect. If V_∞ increases, it will take more ΔV in order to match speeds with the asteroid going in a different direction. At first, V_∞ was going to be our velocity of the spacecraft upon arrival. However, since these asteroids have such close approaches and do not allow for the spacecraft to really reach an 'infinite' distance away, another approach was sought. Therefore, to achieve a better worst case scenario, we will use the vis-viva equation for a parabolic or hyperbolic orbit to calculate the velocity upon arrival. This will always be greater than V_∞ and should give us a better idea of the rendezvous ΔV necessary, but it still follows the same characteristics described above with V_∞. It did, however, bring about some questions as to the actual effects of the added ΔV at departure and what that will do to our time until arrival and the speed at arrival. In order to see, 2013 FU13, an asteroid that met all the necessary criteria (this will be covered in Chapter 4), was chosen and the spacecraft arrival characteristics of time (from Section 3.3.1) and velocity (soon to be shown in Equation (31)) were compared against an increasing ΔV at departure, shown in Figures 16 and 17.

These plots are exactly what we expected. Figure 16 shows a great increase to our speed at arrival early on, but the trend becomes more linear as more ΔV is added. The impact this will have on rendezvous ΔV is extreme and will be discussed in the next chapter. As for Figure 17, the time until arrival already starts out small but dramatically decreases as more ΔV is added. As the extra ΔV goes from zero to one km/s, the time until arrival drops roughly 3 days. Therefore, if we have some extra ΔV to play around with in mission planning, this is something to consider. Chapter

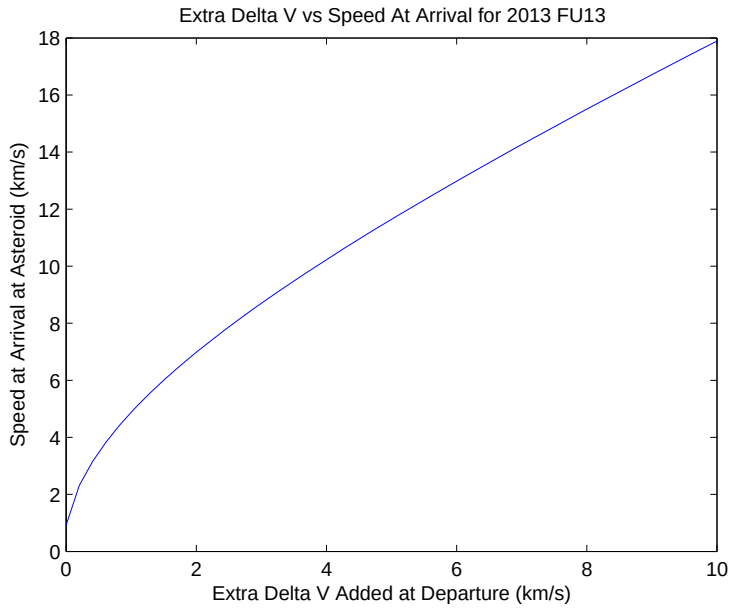

Figure 16. The speed of the spacecraft at arrival at asteroid 2013 FU13 versus the added ΔV at departure.

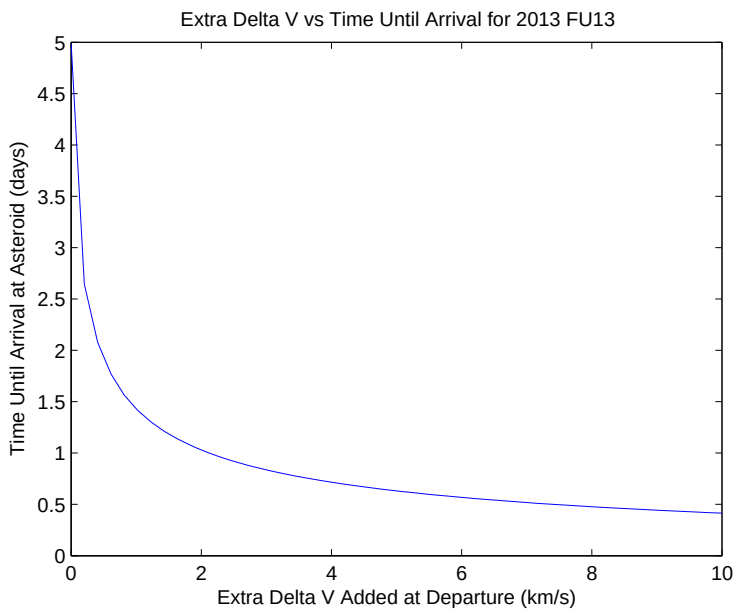

Figure 17. The time it takes for the spacecraft to reach the asteroid 2013 FU13 versus the added ΔV at departure.

4 will discuss the characteristics of other asteroids in regards to added ΔV, but we can assume at this point that each asteroid will have pretty much a similar plot.

Now, for actual calculations of ΔV_{rend}, we already know the velocity of the asteroid and therefore we need to compute the flight path angle of the asteroid, as well as the speed of the spacecraft at the close approach point and its flight path angle. Solving for velocity of the spacecraft uses the vis-viva equation, Equation (5), and the relationship $\varepsilon = -\frac{\mu_\oplus}{2a_{ob}}$. Some rearranging yields:

$$V_{arrival} = \sqrt{\mu_\oplus(\frac{2}{R_{ast}} - \frac{1}{a_{ob}})} \tag{31}$$

Where

$$R_{ast} = \text{magnitude of position vector of asteroid in Geocentric frame}$$

With the speed of the spacecraft we can use the angular momentum of the trajectory in order to determine the flight path angle of the spacecraft upon arrival, $\phi_{arrival}$.

$$\phi_{arrival} = cos^{-1}(\frac{h_{ob}}{R_{ast}V_{arrival}}) \tag{32}$$

This value now needs to be compared to the flight path angle of the asteroid, ϕ_{ast}. To calculate the asteroid flight path angle the equation is the same as Equation (32) but instead uses V_{ast} and h_{ast}.

$$\phi_{ast} = cos^{-1}(\frac{h_{ast}}{R_{ast}V_{ast}}) \tag{33}$$

Where

$$h_{ast} = \mid R_{ast} \times V_{ast} \mid$$

V_{ast} = magnitude of velocity vector of asteroid in Geocentric frame

With these flight path angles, the law of cosines can be used to determine the ΔV_{rend} value.

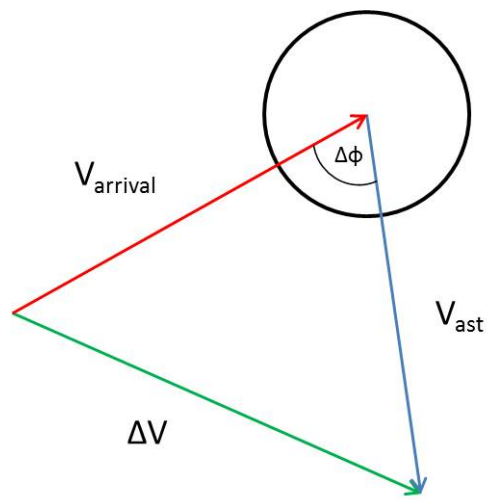

Figure 18. Visualization of rendezvous ΔV maneuver and use of the law of cosines.

Figure 18 shows how the law is used for Equation (34). ΔV_{rend} is the amount of ΔV needed to match speeds with the asteroid to make a safe landing.

$$\Delta V_{rend} = \sqrt{V_{ast}^2 + V_{arrival}^2 - 2V_{ast}V_{arrival}cos\Delta\phi} \tag{34}$$

Where

$$\Delta\phi = \phi_{arrival} + \phi_{ast}$$

3.3.3 Divert ΔV

With the outbound and rendezvous legs accounted for, we now have to see how expensive it will be to return to Earth. Figures 19 and 20 show that we do not need a lot of fuel to alter our trajectory and get us returning back toward Earth if we do the burn laterally. A lateral burn changes our trajectory to bring us back to Earth

but only costs minimal ΔV. If the burn is not lateral than we end up spending unnecessary fuel in order to accomplish the same goal.

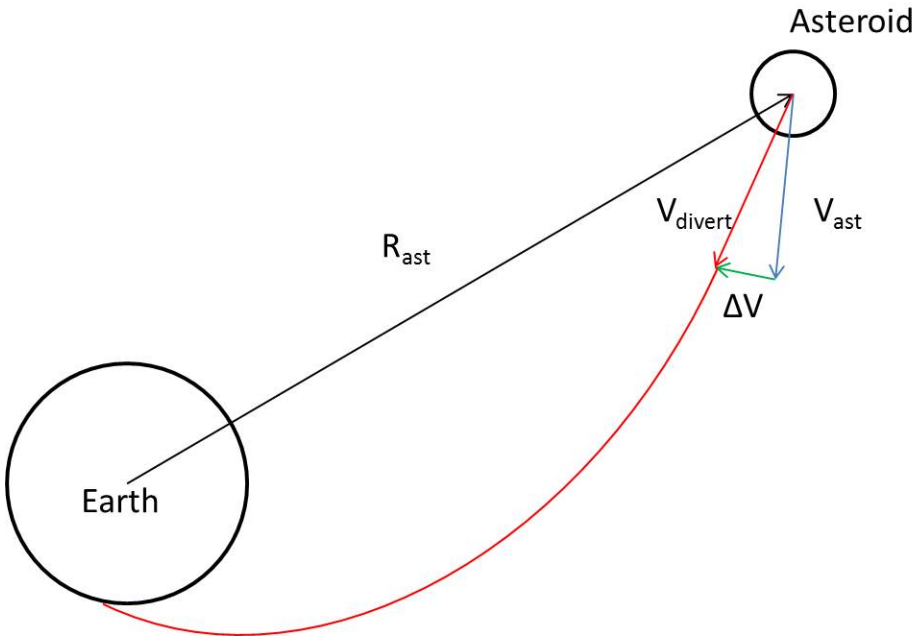

Figure 19. This diagram demonstrates the advantages of a lateral divert maneuver to achieve a proper return trajectory.

The calculations are similar to that of the outbound leg because we utilize the fact that ε and angular momentum are both constants of the motion for the trajectory and do not change from the asteroid to the entry of the Earth's atmosphere. Like in other burns, we start by defining some simple parameters as to how we would like to return to Earth. In order to best utilize the fuel we are bringing, using the atmosphere to slow us down on reentry is ideal and leads us to choose a reentry altitude of 70 km and flight path angle, ϕ_{entry}, of -7 degrees. For later calculations we will need the magnitude of the position vector of reentry, R_{entry}, which will just be our reentry altitude added to $r_{\oplus}$.

To compute the ε value of the divert trajectory, we recognize that the velocity and position vectors when we start is the same as that of the asteroid because our change is so small. With ε, we can also calculate semi-major axis.

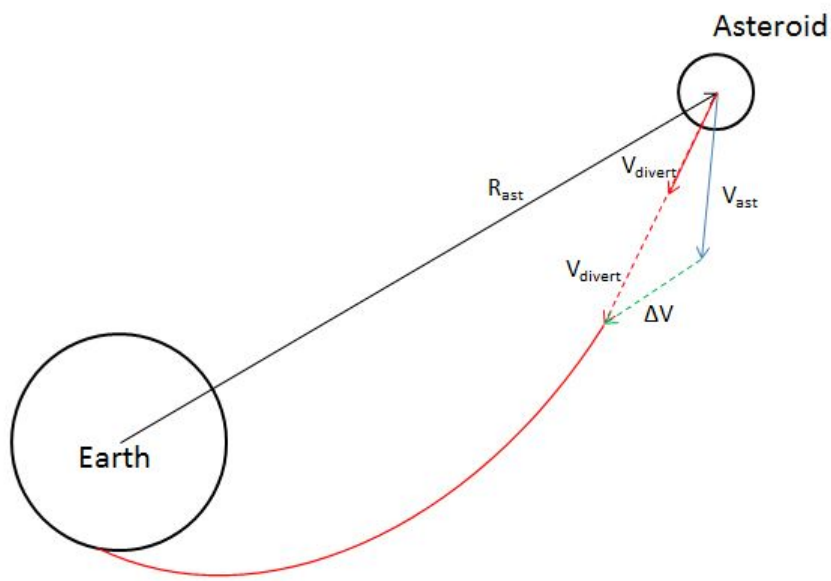

Figure 20. This diagram shows the disadvantages of a non-lateral divert maneuver.

$$\varepsilon_{divert} = \frac{V_{ast}^2}{2} - \frac{\mu_\oplus}{R_{ast}} \tag{35}$$

$$a_{divert} = -\frac{\mu_\oplus}{2\varepsilon_{divert}} \tag{36}$$

Now, using R_{entry} and ε_{divert} we can calculate the speed upon reentry by specifying the general vis-viva equation.

$$V_{entry} = \sqrt{2(\frac{\mu_\oplus}{R_{entry}} + \varepsilon_{divert})} \tag{37}$$

The other orbital elements of the trajectory can also be calculated, to include angular momentum, which remains constant from the asteroid to the Earth, eccentricity, and flight path angle.

$$h_{divert} = R_{entry} V_{entry} \cos\phi_{entry} \tag{38}$$

$$e_{divert} = \sqrt{1 - \frac{h_{divert}^2}{\mu_{\oplus} a_{divert}}} \tag{39}$$

$$\phi_{divert} = cos^{-1}(\frac{h_{divert}}{R_{ast} V_{ast}}) \tag{40}$$

Finally we have enough to calculate the ΔV_{divert} burn. Figure 19 shows that we can use the equation for a simple plane change where the change in direction is the difference in the flight path angles.

$$\Delta V_{divert} = 2V_{ast}\sin(\frac{\phi_{divert} - \phi_{ast}}{2}) \tag{41}$$

After the calculation of ΔV_{divert}, we have the three burns calculated that will impact the mission. To compute return time, the same equations and methodology that were used for the outbound ΔV can be used. However, since the speed of the asteroid is so great, we will be on a hyperbolic trajectory so only Equations (28), (29), and (30) need to be used.

The code is now complete to determine: the orbital elements of the asteroid, the date it is closest to the Earth, how close it approaches, how much fuel it would cost for a crewed or uncrewed mission to the asteroid, and how long the mission will take. Now, of course, this does not include potential fuel used for an actual landing on the asteroid, which will not be very high given the minimal atmospheric conditions on an asteroid and the ability to match velocities. The next chapter will discuss how the code was run and what asteroids were actually found for potential missions.

3.4 Selection Criteria

Now that the major math is completed, the next step is to estimate which asteroids we want to retain for potential missions and which ones we can exclude. There were

three major selection criteria that were used to weed through the asteroids and only concern ourselves with the candidates that have mission potential. These criterion are summarized in Table 3.

Table 3. Details the selection criteria to determine each asteroid close approach opportunity.

Criteria	Limitation
Distance	Within 1.5 Million km
Time	Outbound Trip Less Than 15 days
Fuel Cost	Mission ΔV Under 15 km/s

The first thing to consider was distance. How far the asteroid is from the Earth at their closest point is paramount to a successful mission. With limited ΔV available, the farther the spacecraft has to travel, the more dangerous the mission. Therefore the first restriction was to not consider any asteroid that did not approach within 1.5 million kilometers. The moon's orbit is approximately 384,400 kilometers from the Earth and moon missions were roughly 8-12 days, depending on the stay time on the lunar surface [32]. With this in mind, a mission to an asteroid at 1.5 million kilometers away is a bit extreme but this is a good baseline eliminator and a necessary broad requirement because it is the first criteria that needs to be met. These selection criteria do not make final decisions on which asteroids to choose, but rather to just give a list of ones that would be the best candidates for a crewed or uncrewed mission.

Once the distance to the asteroid was considered, the next requirement was the time to the asteroid. With consideration of an 8-12 day lunar mission and the distance to the moon from the Earth, a limit of 15 days was selected. As discussed in Chapter 2, there are no short duration missions currently being proposed or developed. Throughout my research and discussions, though, the idea of two to three weeks was often thought of as a ideal mission length to an asteroid. It is short enough to excite the public and allow for current technology to be used but also long enough

to explore the asteroid and obtain the information we have been longing for. This period of time would also allow the mission to follow a lunar mission plan for guidance and assistance. Therefore, 15 days to the asteroid is on the upper limit of that plan. A 15 day outbound trip, assuming the same time coming back and a day or two on the asteroid, would lead to a mission duration of about a month.

The final criterion is the amount of ΔV needed for the mission. Oftentimes, with the basic calculations performed and no further analysis, the shorter missions have the highest ΔV requirements, which is what we would expect. So to compute an asteroid that comes within 1.5 million kilometers and has a projected mission length of under a month, while also keeping ΔV relatively low presents an ideal short duration mission candidate. For this reason, a required ΔV of 15 km/s was selected as the final requirement for an asteroid to be considered. This number, though very high, will still be very restrictive and allow for mission planning within the scope of our current technological bounds. Further analysis will be performed on this list of asteroid candidates.

4. Results and Analysis

Now that the equations and code have been developed to calculate how close the asteroid comes to the Earth, when it does, and how costly it will be to get there, we need to look at the results that came out of this. This chapter will not only cover all of the results that were obtained following the steps in Chapter 3 but also analyze what these results mean and how we can apply them to what we are trying to accomplish. In addition to following the methods from the previous chapter, any additional steps that were taken will be covered and described in detail.

4.1 Asteroid Selection

All of the equations described in Chapter 3 were coded into MATLAB® scripts in order to automate the calculation process for each asteroid. The JPL catalog, described in Section 3.1.1, yielded 13,065 asteroids that were considered NEOs. Once the main code was run, an output file with characteristics of a potential mission was output, but only if the asteroid met at least the first constraint, no more than 1.5 million kilometers away at its closest point. The code ran for every Earth orbit crossing date of each asteroid. Therefore every file that was output represented a close approach opportunity, not a different asteroid, because it is possible that an asteroid crossing date meets all of the requirements in one year and fails to meet them in another, based on the movement of the Earth. So the numbers in Table 4 are representative of the number of opportunities for missions, not the number of asteroids that meet the requirements. The opportunity did not have to meet all three requirements to be output, just the distance limitation. After the code was first run the following data was output.

Table 4 tells us that 27 asteroid opportunities met all three requirements, one of

Table 4. Details the number of asteroid close approach opportunities meeting each component of the selection criteria.

Requirement	Number of Opportunities
Within 1.5 Million km	145
And Outbound Trip Less Than 15 days	115
And Mission ΔV Under 15 km/s	27

them being 2013 FU13, which was used in Chapter 3. So out of the 13,065 cataloged NEO asteroids that were reviewed, each with numerous Earth orbit crossing dates, only 145 opportunities meet at the very least the distance requirement. If that is the case, some might wonder why are they even called NEOs? Missions to these other asteroids would take months if not years and really not do not come too close to the Earth anyway. One thing to keep in mind is the time used for crossing dates. The missions were believed to be before the year 2080 and therefore some of these asteroids could pass close by after that and not be included. However, this cannot be the only thing impacting the results.

4.1.1 Close Approach Comparisons

NASA and JPL keep a 'Small-Body Database' that provides orbital data, close approach data, and even orbital diagrams [31]. This browser has been an excellent tool to validate not only the orbital data that was pulled into MATLAB® from the NEO catalog but also to check the close approach dates, perhaps the most important aspect of correctly modeling the orbit. These dates can be inspected in STK as well, as long as the orbit of the asteroid is modeled correctly with a correct true anomaly value. In order to demonstrate the validity of my calculations, 5 asteroids were selected that passed all three criteria and compared their next close approach date in the NASA/JPL database.

While these calculations are correct, there were 12 of the 27 opportunity candi-

Table 5. Asteroid calculated close approach dates versus the NASA/JPL database close approach dates.

Asteroid	Calculated Date	NASA/JPL Date
2001 GP2	5 October 2020	3 October 2020
2001 AV43	5 November 2029	11 November 2029
2008 LG2	29 June 2049	16 June 2049
2012 PB20	30 January 2025	9 February 2025
2012 UX136	1 November 2037	4 November 2037

dates that did not match up close approach dates with the NASA/JPL database. There are a number of things that could lead to the slight difference in the close approach dates shown in Table 5, with perturbations most likely being the biggest factor. As the asteroid travels around the Sun, solar radiation pressure and the gravity pull of the planets and other celestial bodies can have a major impact on the asteroid and affect how it travels through the solar system. As discussed in Chapter 1, perturbations were not taken into account in the calculations and therefore could be the main cause of this difference. However, more analysis needed to be done to estimate why some calculations are showing correct and others not. Consider the fact that NEOs are only tracked at the Earth orbit crossing point. Therefore, at this one point of the orbit, it is very well determined. However, as the asteroid continues to orbit the Sun, the confidence in the orbital propagation drops. For further analysis, there were two major aspects that were looked at, STK and OCC.

Discussed in Chapter 3, one of the pieces of information that the NEO catalog produces is the Orbital Condition Code, OCC, that describes how well the orbit is defined with 0 meaning the most defined, and 9 meaning very poorly defined. If an asteroid has an OCC of 0, then there should be matching close approach dates, and if an asteroid has a 9, then it makes sense if the dates do not match because the orbital elements are changing with each new data set that NASA/JPL inputs. Table 6 displays the OCCs from the five asteroids in Table 5 and 5 whose close approach

dates did not match with the NASA/JPL database to see if OCC played a role in some of the differing close approach dates.

Table 6. The OCC for 10 different asteroids that either match or do not match the calculated NASA/JPL close approach dates.

Asteroid	OCC	Matching Close Approach Dates?
2001 AV43	0	Yes
2008 LG2	1	Yes
1998 KY26	2	No
2010 TE55	3	No
2012 PB20	4	Yes
2010 CK19	4	No
2012 UX136	6	Yes
2001 GP2	6	Yes
2009 QR	6	No
2007 DC	8	No

The results show that OCC does not give any insight as to whether or not the dates will match up. The asteroids with OCCs of 0 and 1, as predicted, match with their close approach dates, but the asteroids with 2 and 3 do not. On top of that, asteroids with an OCC of 4 both match and do not match. The same is true for an OCC of 6. This raises the question of a potential cutoff. Are all well defined asteroids (OCC of 0 or 1) matching up with the given database information? There were only two other asteroids with their orbits well defined that met all three criteria. Those asteroids were 2013 XK22 (OCC of 1) and 2000 PH5 (OCC of 0), also called 54509 YORP. The close approach date for 2013 XK22 was calculated as 17 December 2044, but its reported date is rather 22 September 2044. The difference could be affected by perturbations, but 2008 LG2, used above, has the same OCC and was only off by a couple weeks in its calculation. 2000 PH5 was, unfortunately, off by years. Therefore, it is safe to say that the OCC does not have any impact on whether or not the close approach dates match up. From here, the next step is to use the propagation in STK and model the asteroid versus the Earth and go to the predicted close approach date

to see how close they are.

4.1.2 STK Modeling

The first scenario was modeled for asteroid 2013 FU13, the same one from Figures 16 and 17, whose results did not match up with the database. Knowing that 2013 FU13 (will be referred to as FU13) was an asteroid and not a satellite, it was important to utilize the Component Browser in STK rather than creating a satellite object. The simplest way to input an asteroid is to duplicate a current planet and change its characteristics, such as size, gravitational parameter and orbital elements. However, STK 10 does not allow for planet duplication unless you are using a 'testPlanet' file. Therefore, after contacting Analytical Graphics Inc.(AGI), the 'testPlanet' file was obtained and used. The 'testPlanet' file was duplicated and FU13's orbital elements were input to be propagated to the calculated close approach date. We see in Figure 21 where FU13 is on the close approach date of 19 March 2038 compared to the Earth with the moon also included for reference. Though the asteroid actually approaches closest a few days prior, this figure validates the calculated 2-body approach discussed in Chapter 3. Figure 22 is also shown to demonstrate the Apollo characteristics of the asteroid. Though some asteroids have bigger inclinations, FU13 only has a roughly 0.75° inclination, making the problem virtually 2-D. The green grid in Figures 21 to 26 defines the ecliptic plane.

Now that an asteroid that did not match up was validated through modeling, it made sense to model one that did match up. The same process of duplicating a 'testPlanet' was used to model the orbit of 2001 GP2. The asteroid chosen had a calculated close approach date of 5 October 2020. Although Figure 23 makes it seem that the inclination of 2001 GP2 is much greater than zero (that of the ecliptic plane), the inclination is only about 1.3°, which is even smaller at the crossing point, as we

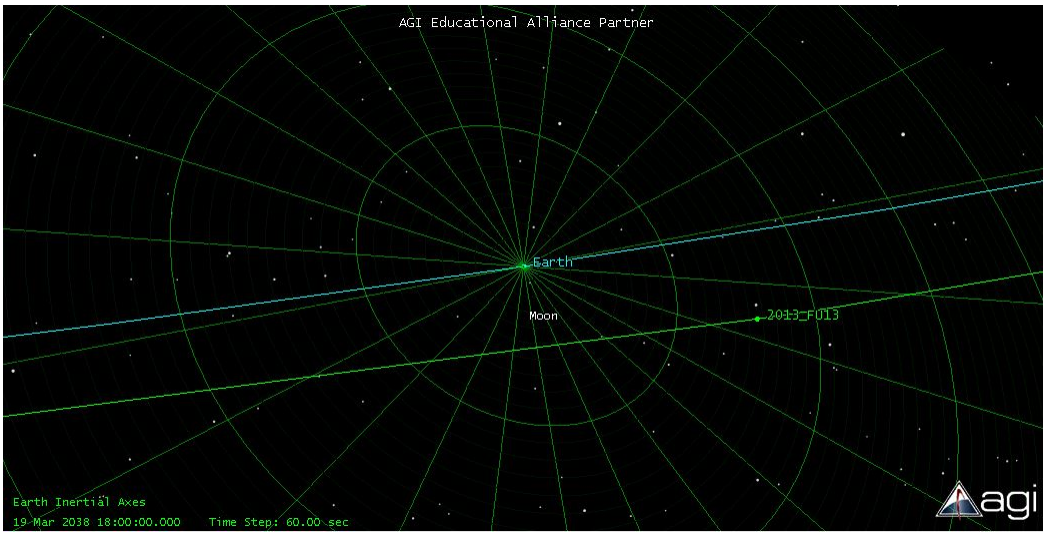

Figure 21. A 3-D, geocentric STK screenshot of 2013 FU13 (green) as it passes by the Earth (blue) on 19 March 2038.

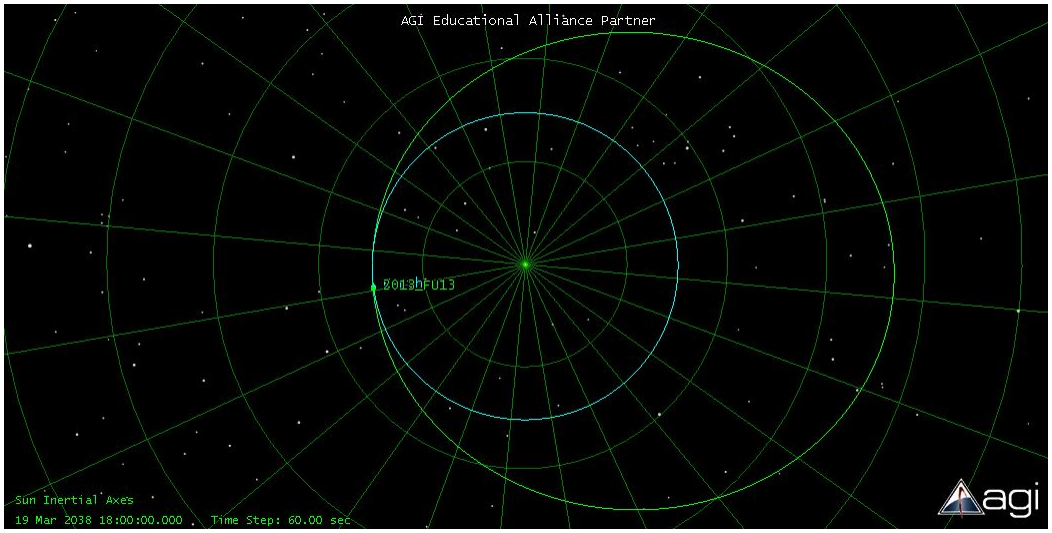

Figure 22. A 2-D, heliocentric STK screenshot of 2013 FU13 (green) as it passes by the Earth (blue) on 19 March 2038.

learned in Section 3.1.2. Therefore the problem can be simplified to two dimensions. The 2-D orbital view in Figure 24 gives perspective as to the size of 2013 FU13's orbit see in Figure 22. 2001 GP2 has roughly the same size orbit as the Earth while 2013 FU13's is much bigger. Both, though, are still considered Apollo asteroids.

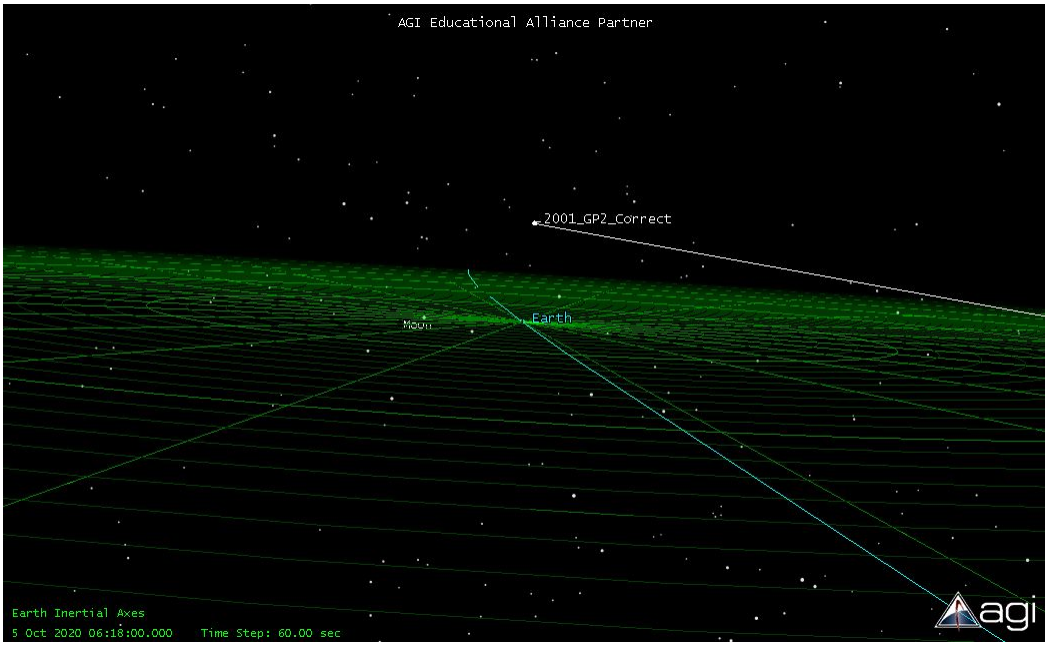

Figure 23. A 3-D, geocentric STK screenshot of 2001 GP2 (white) as it passes by the Earth (blue) on 5 October 2020.

Continuing the validation through modeling, it was decided to use one more asteroid to give us a general consensus that the calculations were correct. It made the most sense to choose one that had close approach dates that did not match up with the NASA/JPL database. The asteroid 2009 QR, also used in Table 6, was modeled next using the same approach in STK 10, and its calculated close approach date was 30 August 2023. Figure 25 shows 2009 QR at the point when its roughly 3.41° inclined orbit crosses the ecliptic plane, making the problem 2-D. The date when this happens is actually 24 August 2023, so the date is very close and the change can be attributed to the perturbation effects described above. The asteroid approaches roughly 2 lunar distances away, providing a great mission opportunity. Again, this was an asteroid

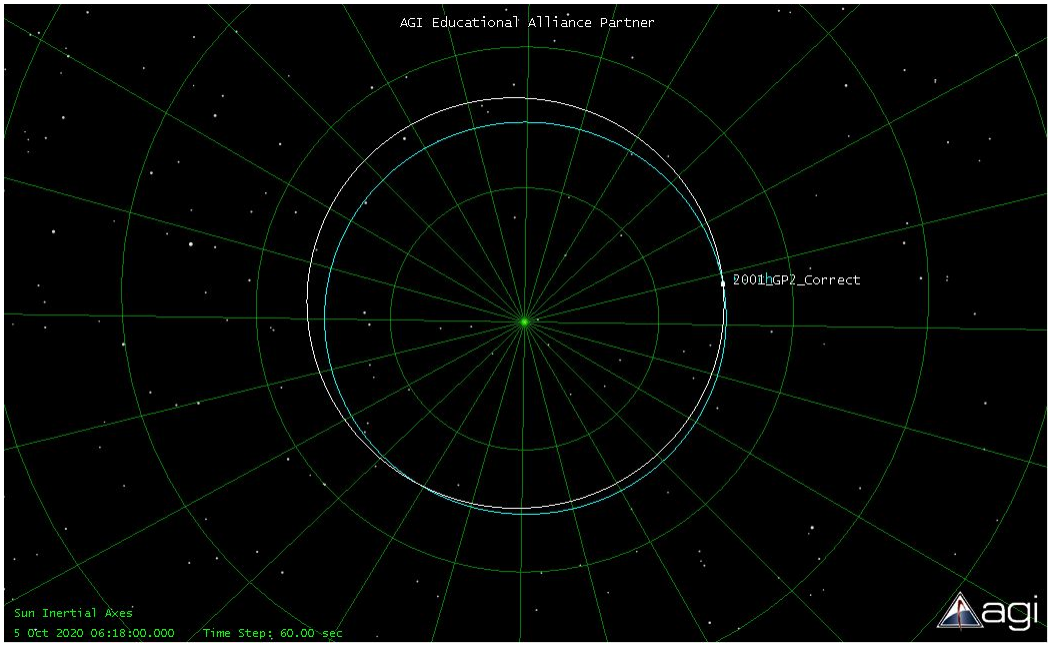

Figure 24. A 2-D, heliocentric STK screenshot of 2001 GP2 (white) as it passes by the Earth (blue) on 5 October 2020.

with a close approach date that did not match up with the database, but the orbital propagation in STK provided validation for the 30 August 2023 close approach. The view from the Sun as the central body shows that 2009 QR is another Apollo asteroid. These are ideal for close approaches due to their Earth-like orbits that allow for low relative approach speeds, thus leading to missions with smaller $\Delta V's$.

Figures 23 to 26 clearly demonstrate that the propagation used in Chapter 3 was correct. This is because the propagated asteroid orbits in STK show the same close approach distance on the same close approach date that was calculated. As discussed in Section 1.2, the different techniques that we used are sufficient for calculating general close approach dates and modeling them. One thing to consider for why 12 opportunities are different in the database is what constitutes a close approach for NASA/JPL. Perhaps the constraints and restrictions used in Chapter 3 were stricter than what the database uses, but that seems unlikely. The documentation of the tool states "Output is only produced when the selected object reaches a minimum distance

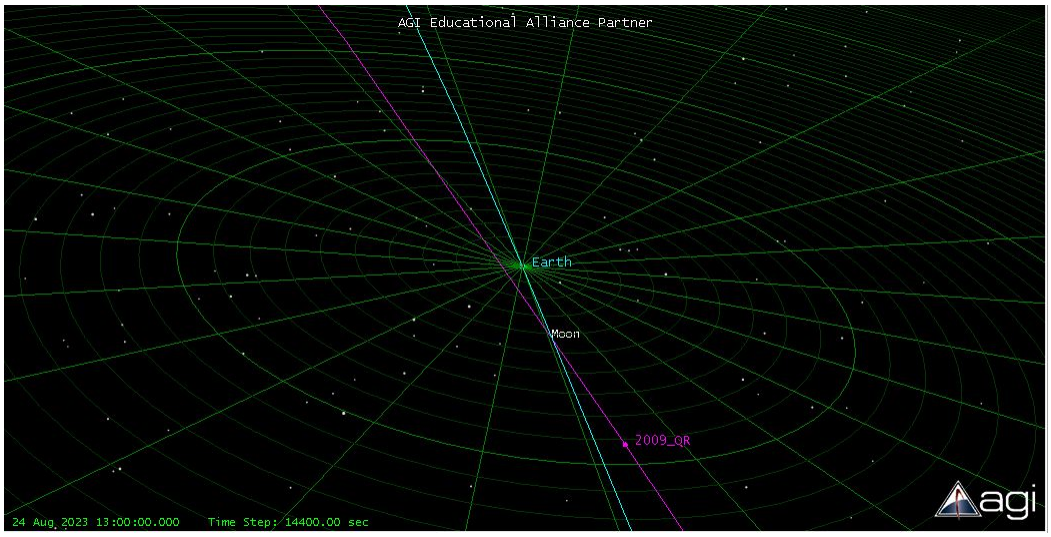

Figure 25. A 3-D, geocentric STK screenshot of 2009 QR (purple) as it passes by the Earth (blue) on 24 August 2023.

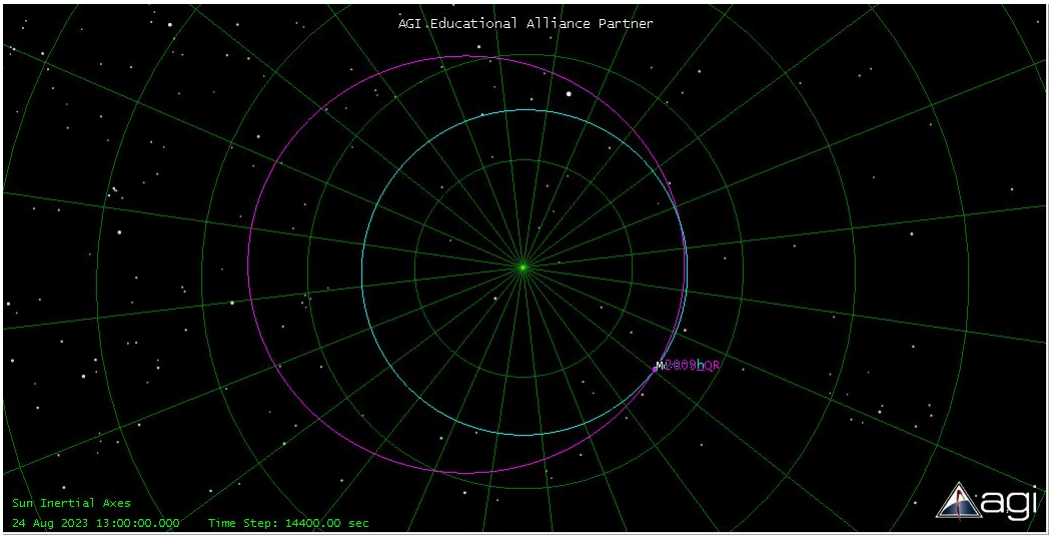

Figure 26. A 2-D, heliocentric STK screenshot of 2009 QR (purple) as it passes by the Earth (blue) on 24 August 2023.

within a set spherical radius from a planet..." [8]. However, we do not know what that minimum distance is. If the constraints aren't causing the difference, it is possible that the higher fidelity NASA models, which used numerical integration to propagate with special perturbations, show that the distance between the asteroid and the Earth is farther than calculated. It is strange, though, that these models would still have some opportunities that matched up while others did not. The perturbational effect on each asteroid will be different because every asteroid has different COEs, but the effects should not alter close approach data by years. This is potential future work that can be looked into.

4.2 Final Opportunities

Now that we know there are some differences in comparing the 2-body results to NASA/JPL's, the opportunities that match up are going to be the ones we focus on for the rest of the analysis. Sometimes, the database does not extend as far as the first calculated close approach date for the asteroid. However, the results presented in this research include these asteroids and all 15 opportunities can be seen in Table 7.

We will try to determine opportunities with the lowest ΔV and time to arrival to choose for our missions. ΔV is somewhat misleading in this context because we are not accounting for mission planning. The total ΔV number could be one thing, but a rendezvous burn cannot be accomplished the same day as the outbound burn. Therefore, ΔV, at this point, is just used as a marker to show the absolute lowest possibility. A simple scatter plot was produced for each mission opportunity to see how the cost of the mission compares to the time until arrival at the asteroid. Ideally, an opportunity would have low values of both and be in the bottom left quadrant of Figure 27.

Table 7. Final list of close approach opportunities to be analyzed.

Asteroid	Calculated Date
1999 VX25*	11 November 2034
2001 GP2	5 October 2020
2001 GP2*	6 October 2057
2001 AV43	5 November 2029
2006 HE2*	29 May 2040
2006 XY*	1 December 2061
2006 XY*	1 December 2072
2008 LG2	29 June 2049
2009 QR*	30 August 2037
2009 QR*	31 August 2051
2012 PB20	30 January 2025
2012 UX136	1 November 2037
2013 ED68	15 March 2022
2014 AA*	28 December 2077
2015 EG	6 March 2019

* Unvalidated, database does not extend to this date

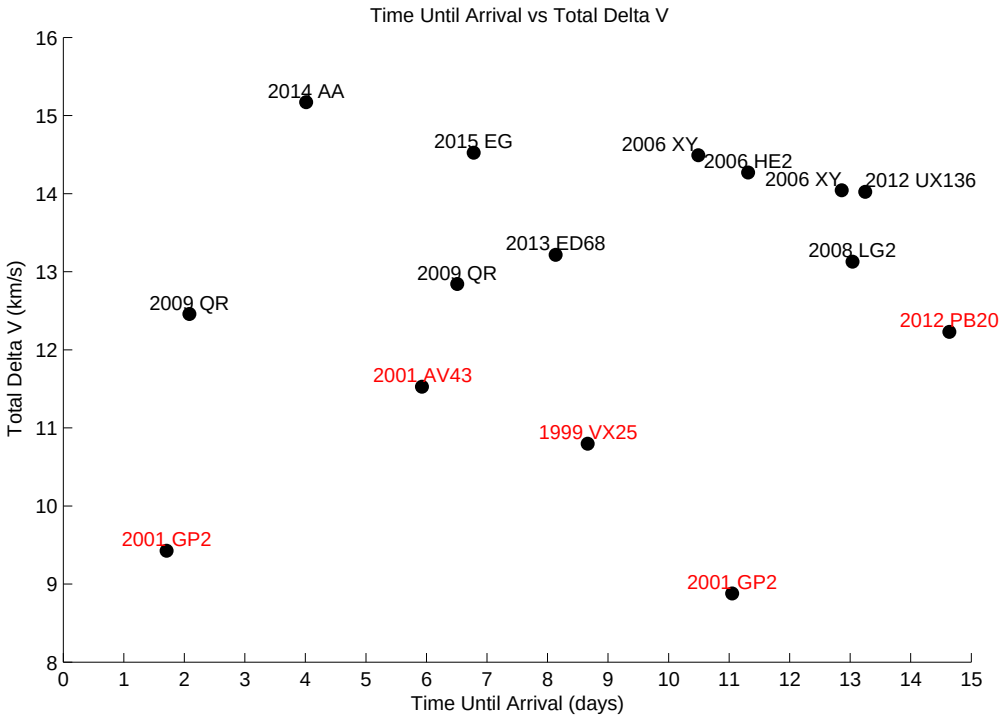

Figure 27. A scatter plot of opportunities from Table 7 and time versus ΔV comparison.

The five red labels show the five opportunities that provide the lowest cost and length of time for a mission. These five include four different asteroids and two opportunities for 2001 GP2, with its first close approach date having a lower cost of the two options. From here, we want to see the details of a mission for each one of these opportunities. Then, we can analyze how much more extra ΔV can be added to the departure and what length of stay can be expected on the asteroid. When the code was run to loop through the catalog and obtain results for each asteroid, a text file was created that contained orbital parameters of the asteroid as well as mission information. Using these output files, Table 8 was developed (see Table 7 for close approach dates). Outbound ΔV for every case was the same value, 3.21 km/s, because that is the difference between the escape velocity and the current parking orbit velocity the spacecraft would have. Therefore, that value was left out of the table but needs to be considered in total ΔV calculations. As discussed earlier, ΔV here is not the exact cost because of the lack of mission planning. For instance, divert ΔV will go down the farther away the bodies are from each other but currently they are at the closest point, so the value is higher.

Table 8. The cost values for each of the calculated five asteroid close approach opportunities. Outbound cost was left out because it is constant for each mission depending on the added ΔV at departure.

Asteroid	Closest Distance (km)	Rendezvous ΔV (km/s)	Divert ΔV (km/s)
1999 VX25	934,981	3.55	4.03
2001 AV43	724,655	3.80	4.52
2001 GP2	1,100,872	2.57	3.09
2001 GP2	312,884	2.89	3.32
2012 PB20	1,328,937	4.50	4.52

4.3 Time Analysis and Mission Planning for Minimum Time Approach

Something that has not been discussed thoroughly is the time element to these missions. For instance, how long would a mission to each one of these asteroids take and does that fit what we are trying to accomplish? Table 9 displays the time to arrival for each of the five opportunities we have discussed with no ΔV added at departure. The data below represents the results from the calculated close approach date, with the Earth and asteroid both propagated.

Table 9. The distance and time values for each of the calculated five asteroid close approach opportunities.

Asteroid	Closest Distance (km)	Time Until Arrival At Closest Point (days)
1999 VX25	934,981	8.66
2001 AV43	724,655	5.93
2001 GP2	1,100,872	11.05
2001 GP2	312,884	1.71
2012 PB20	1,328,937	14.64

Each opportunity has a close approach that meets our constraints. However, the speed of each asteroid as it passes by is so fast that the time to arrival on days before or after the calculated close approach date begins to grow rapidly. For example, Figure 28 shows the increase in time to arrival depending on when the mission begins. For comparison, the distance is also displayed in the figure to show how rapidly the distance increases between Earth and the asteroid.

This brings everything back to the initial motivation of a short duration mission. Can we accomplish visiting these bodies in weeks rather than months? From here it seems that there is only a small window to accomplish this mission. There are ways to get around this, though. First, the return leg does not have to be the same time as the outbound leg. Section 2.4.2 discusses longer outbound trips with shorter return legs and vice versa. This is something we can consider; adding ΔV at the start to arrive within a week or two and then taking a longer return trip back to Earth.

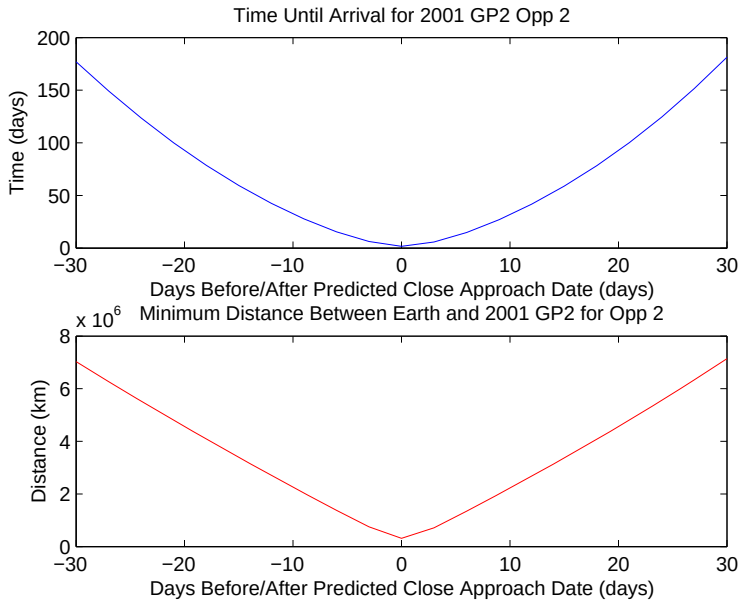

Figure 28. The time until arrival at asteroid 2001 GP2 shown with the minimum distance between Earth and 2001 GP2.

This can get dangerous though as we learned above, that adding ΔV at departure can greatly increase the necessary ΔV for the mission. For reference, Lance Benner from JPL at California Institute of Technology (CalTech) performed numerous ΔV calculations for different asteroids following the same method used by Shoemaker and Helin back in Section 2.4.1 [33]. His analysis used ballistic trajectories with no ΔV added at departure and no date to reference for actual mission planning, and for this reason, the comparison between his results and the results above cannot be made. In addition, following Shoemaker and Helin's method, the divert ΔV value is not accounted for. However, Benner does state "For comparison, delta-v for transferring from low-Earth orbit to rendezvous with the Moon [is] 6.0 km/s" [33]. We now have a value for reference for a moon mission that does not include returning to Earth but this value can simply be added on to come up with a total ΔV. The next step is to determine how much ΔV can be added at departure to minimize the time until arrival at the asteroid while also keeping ΔV low enough to have a potential mission.

There will be four assumptions of mission planning that will be used for analysis, shown in Figure 29. They are:

1. An arbitrary value of 3 days will be spent on the asteroid itself.
2. The outbound leg time used will be to the asteroid on the calculated close approach date with minimum time.
3. Rendezvous maneuver will occur after arrival and divert maneuver will occur 3 days after rendezvous.
4. Return leg time will be hyperbolic with no added ΔV and will be added to the outbound leg and a 3 day stay on the asteroid to get total time.

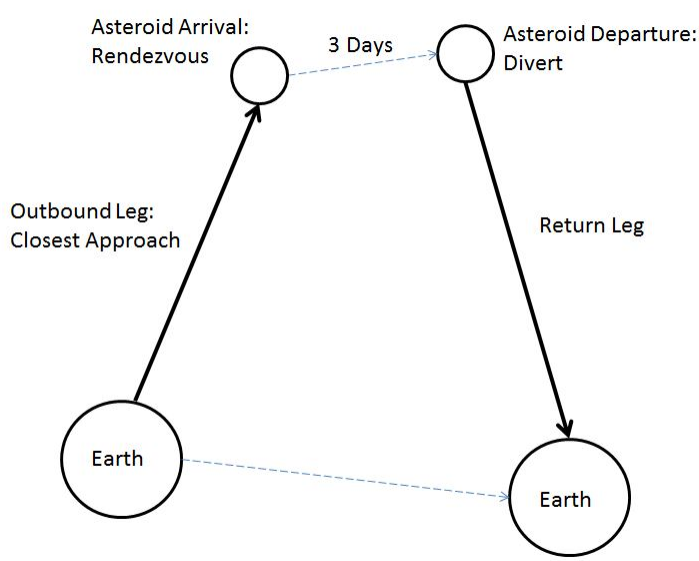

Figure 29. The basic parts of analysis used for mission planning for different asteroid close approach opportunities.

With 1999 VX25, the minimum time until arrival is 8.66 days with no added departure boost, and this would require a departure from LEO on 30 November 2034. With 9 days through space until arrival, the rendezvous ΔV on 9 December

2034 is 3.83 km/s, which is on the low end. After a three day stay on 1999 VX25, the divert ΔV is 0.86 km/s, which is quite low. From here, let's see what added boost at departure will do to the mission. With a boost of 1 km/s, our time to 1999 VX25 drops to 2.12 days. Now, arriving on 3 December 2034 (about a week earlier), the rendezvous ΔV jumps because of our increased velocity at arrival to 7.65 km/s. This all but eliminates this mission possibility, but after a three day stay, a return trip on 6 December requires about 1.44 km/s and returns us to Earth in about a week. The added boost effects are incredible, but as shown in Table 10 below, an added boost may not be entirely necessary. We are looking at a 24 day mission with only 7.90 km/s of ΔV, which is very plausible.

Table 10. The effects of added boost at departure on mission planning for 1999 VX25.

Added Boost (km/s)	Mission Length (days)	Total ΔV (km/s)
0	24	7.90
0.5	13	11.45
1	13	13.30

What is also very interesting about these calculations is the trend of the divert, rendezvous, and outbound $\Delta V's$. As Figure 30 shows, the divert ΔV is small the farther the two bodies are from each other and only escalates the closer they get. The trend of the rendezvous burn is tougher to explain. The closer the asteroid is to the Earth, the faster the spacecraft is moving and therefore the value would be expected to increase at the minimum distance, not decrease. Further analysis to conclude why the drop occurs at this point is something to consider.

2001 AV43 presents a more logical opportunity on 14 November 2029. A 6 day outbound trip leads to a 4.17 km/s rendezvous leg on 20 November 2029, a three day stay, and finally a return trip on 23 November of about 9 days using 0.87 km/s. Adding 1 km/s of boost from LEO will have a different effect on each opportunity, some being more beneficial than others. In this case, our outbound leg drops to

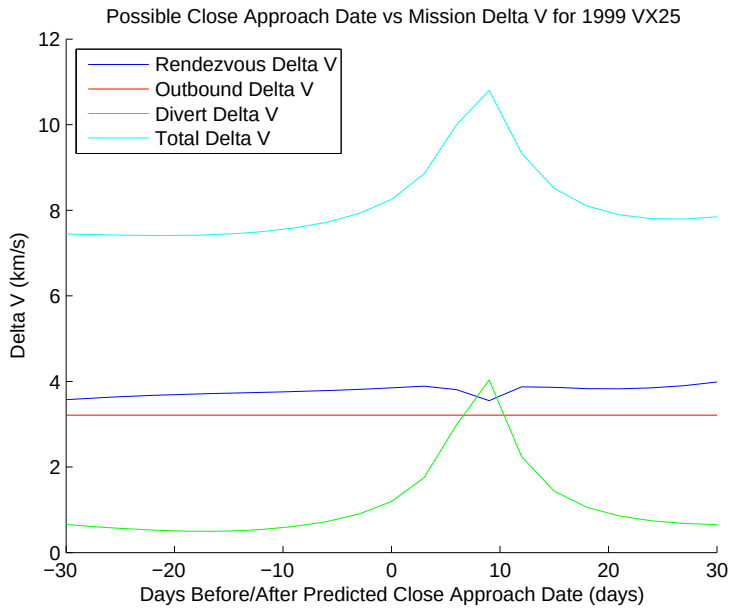

Figure 30. Each component of mission ΔV compared to the predicted close approach date of 1999 VX25 with no ΔV added at departure.

about 1 and a half days, putting our arrival on 16 November. The rendezvous leg now becomes about 7 km/s and finally a return trip of roughly 4 days utilizing about 2 km/s of ΔV.

Table 11. The effects of added boost at departure on mission planning for 2001 AV43.

Added Boost (km/s)	Mission Length (days)	Total ΔV (km/s)
0	18	8.25
0.5	12	11.09
1	10	13.21

2001 AV43 presents another excellent opportunity with no added boost at departure. A mission duration of 18 days is adequate and a mission ΔV of 8.25 km/s, though higher than 1999 VX25, is still quite low. Not surprisingly, the plot for total mission ΔV looks very similar to that of 1999 VX25.

Taking a look at the last three opportunities in Table 12. The ΔV plots for opportunities for 2001 GP2 can be seen in Appendix A. While the time seems to work for all of the missions in the table, the ΔV increases greatly once a boost is

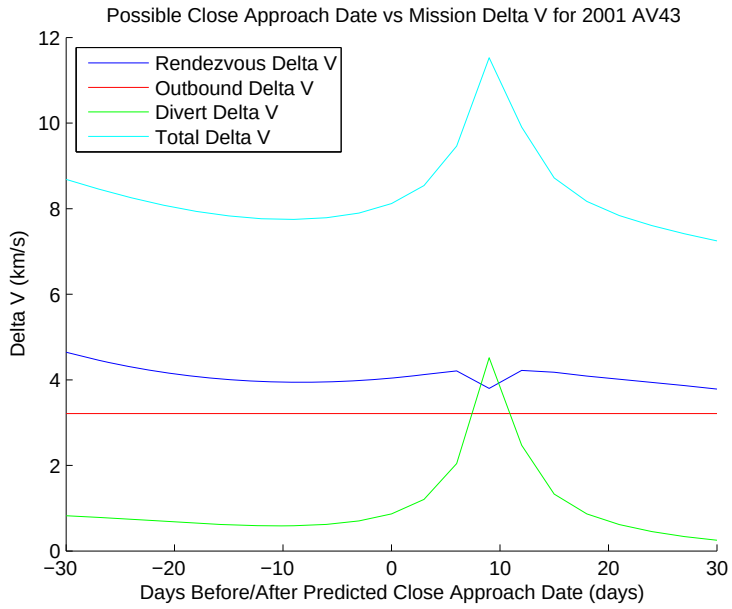

Figure 31. Each component of mission ΔV compared to the predicted close approach date of 2001 AV43 with no ΔV added at departure.

Table 12. The effects of added boost at departure on mission planning for 2001 GP2 Opp 1, Opp 2, and 2012 PB20.

Asteroid	Added Boost (km/s)	Mission Length (days)	Total ΔV (km/s)
2001 GP2 Opp 1	0	30	6.97
	0.5	13	10.43
	1	13	12.26
2001 GP2 Opp 2	0	9	7.26
	0.5	8	9.51
	1	8	11.01
2012 PB20	0	36	8.58
	0.5	15	12.05
	1	14	13.97

added at departure. It also seems that this boost does not seem to have a great impact on arrival time; only shrinking it by a day or two. Therefore, it seems no boost, or at least a very small one, should be added at departure. The cost figure for 2012 PB20 is shown to follow a similar path as the other figures.

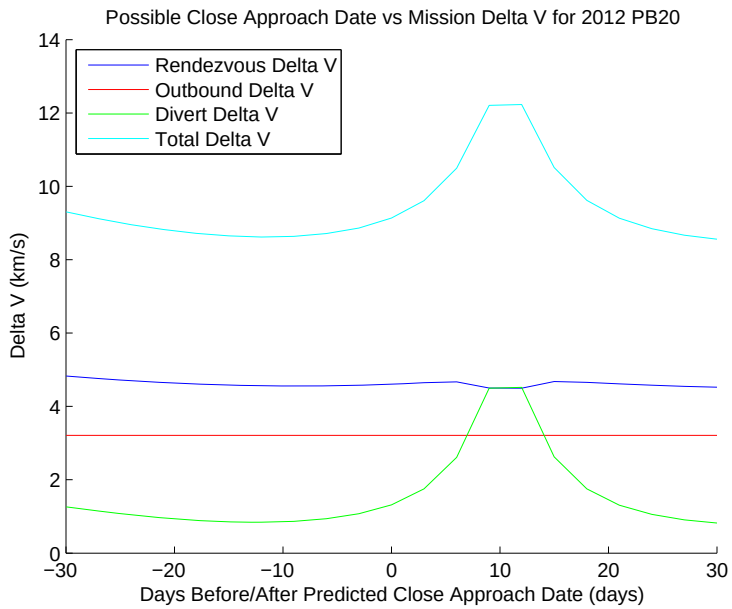

Figure 32. Each component of mission ΔV compared to the predicted close approach date of 2012 PB20 with no ΔV added at departure.

4.4 Developing a Minimum ΔV Approach by Accounting for Potential Errors

The following section presents an analysis to determine a minimum ΔV mission to compare to the minimum time mission presented earlier. The analysis also allows us to account for potential propagation errors. The thought came about in Chapter 3 that if there are slight potential errors in the close approach date (due to perturbations and period errors), is it possible that a different date will yield lower ΔV values? Previously it was discussed how the closest distance was calculated propagating the Earth and the asteroid to see when the closest approach would occur. This gave us

a new close approach date to use, and the date for each of the five opportunities presented utilizes the new close approach date. Figure 33 shows how both bodies are being propagated in order to estimate the best close approach date.

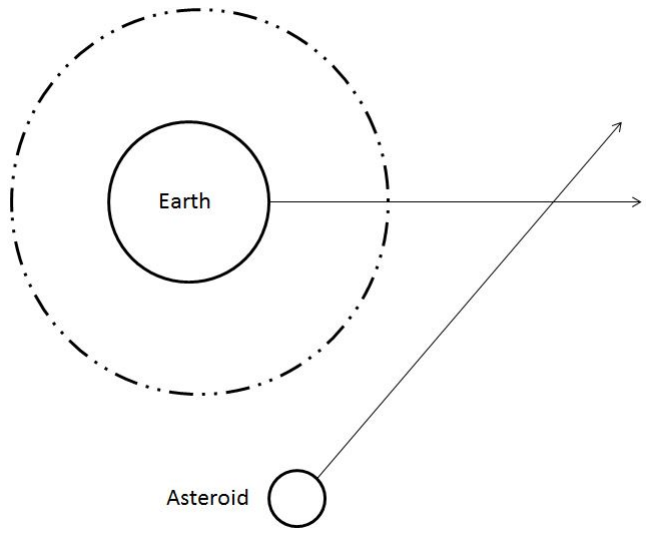

Figure 33. Both bodies are being propagated in order to figure out the minimum ΔV.

Taking a different approach, if we do not propagate the Earth, and keep it fixed in its orbit, while still propagating the asteroid, we can cover an array of different close approach dates and try to account for these small calculation errors. Not propagating the Earth means the period errors that come from the asteroid will change where the Earth is when the asteroid comes by, as shown in Figure 34. Since we do not know what the approach will look like, keeping the Earth fixed in its orbit will help us account for more of those different approaches. The purpose of this analysis, though impossible, was to see how the ΔV was affected by propagation of purely the asteroid. The position of the Earth that was used was that of the closest approach. In other words, the minimum distance between the two bodies was found, then the asteroid was moved 30 days back and 30 days forward and compared to that same, fixed position of the Earth. Essentially, there are now three different dates to work with. We have a

predicted close approach date (described in Section 3.2) that we get from adding the period to the initial close approach date. There is a calculated close approach date (method one also described in Section 3.2) that describes the date where both bodies are propagated to the closest point. And finally a minimum ΔV date (method two from Section 3.2) that describes the minimum fuel cost for a mission accounting for close approach errors.

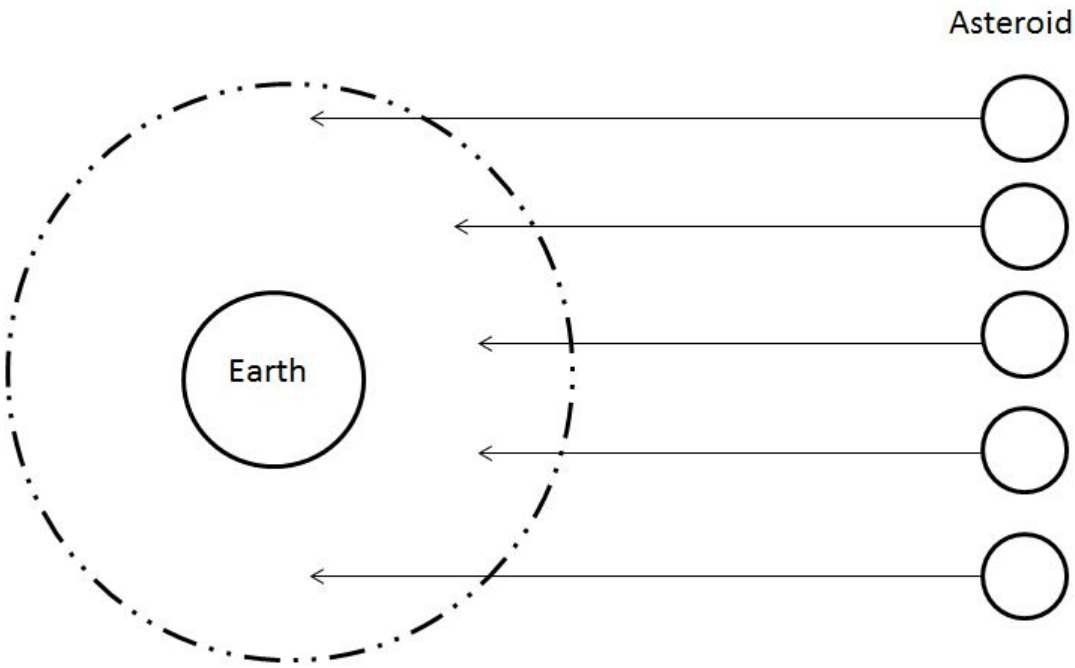

Figure 34. The Earth remains fixed in its orbit while the asteroid is being propagated in order to figure out the minimum ΔV and closest approach date.

Every different option from 30 days prior to the calculated close approach to 30 days after was considered, in 3 day increments. The results surprised because they showed that the minimum ΔV case does not always occur on the calculated or predicted close approach date but usually rather on a date within a few days. Though this does not clear up the differences in the database and calculated values, it does allow us to recognize the possibilities of different missions should errors be found in the close approach dates. Something to keep in mind for the total ΔV is that the

mission planning we saw used before is not taken into account. This means that the minimum ΔV date is simply the date with the lowest combination of outbound, rendezvous, and divert $\Delta V's$ and does not consider how long the outbound trip is prior to the rendezvous leg, like we did in Section 4.3. That being said, the data below gives more of a general overview than a definitive solution. Figure 35 shows each component of ΔV for a mission to 1999 VX25 from one month prior to 30 November 2034 to one month after.

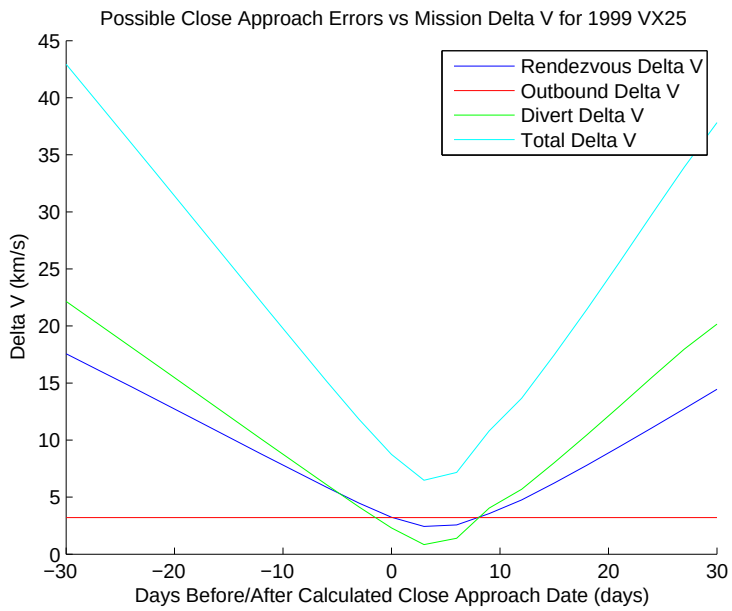

Figure 35. Each component of mission ΔV compared to the calculated close approach date of 1999 VX25 using method 2.

From the plot we can see that the total ΔV for the mission can be much less than the projected 7.90 km/s. The matrix of total ΔV values tells us that a 6.48 km/s cost is achievable. Our minimum ΔV comes about 3 days after the calculated close approach date when the divert value becomes very small. The total ΔV line on the plot seems to clearly follow the divert ΔV line. As discussed above, outbound ΔV is going to remain constant because of the constant parking orbit and escape velocity. Not surprisingly, rendezvous ΔV is great the farther away the asteroid is

due to the high speed difference between the asteroid and the spacecraft. The divert ΔV trend was unexpected because we would expect the farther away we are, the lower the value because only a slight change in direction is necessary. However, we see that the farther we are, the much greater divert ΔV becomes and the divert line couple with the rendezvous line drives the total line. This unexpected divert cost can be attributed to the extreme changes in distance between the asteroid and the Earth. Since the Earth is no longer being propagated, the distances between the bodies gets much larger much faster, leading to a very high divert cost.

Table 13. The predicted, calculated, and minimum ΔV dates for the asteroid 1999 VX25.

Predicted Date	Calculated Date	Minimum ΔV Date
21 November 2034	30 November 2034	3 December 2034

The next opportunity we will consider is 2001 AV43 and its predicted close approach date of November 5th, 2029. For this opportunity the total fuel cost would be 8.25 km/s. Though the total ΔV number is greater than that of 1999 VX25, the time to arrival is just short of 6 days and is based on a calculated close approach date of 14 November 2029. Including a return trip, this adds up to about a week difference between the two trips, putting 2001 AV43 at the top of the short duration list. Again, like 1999 VX25, we have the option of adding ΔV at the start but with the already high estimate of fuel cost, there is not a need to add any ΔV to the mission currently. The time it takes to arrive at 2001 AV43 qualifies as short duration; the only issue is the high ΔV value required.

The best way to approach this candidate and to try to minimize the ΔV is to do the same as we did with 1999 VX25 and vary the close approach possibilities to see, if errors were present, the minimum cost potential. As Figure 36 shows us, the same pattern that we saw in Figure 35 occurs with ΔV possibilities. The divert cost once again drives the total cost and dictates when the best approach would be.

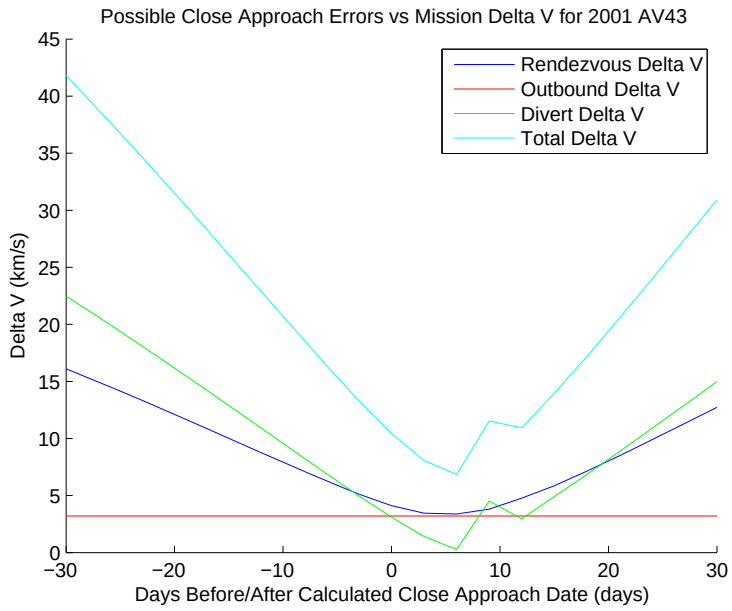

Figure 36. Each component of mission ΔV compared to the calculated close approach date of 2001 AV43 using method 2.

The results show that a minimum ΔV of 6.85 km/s is possible should the mission occur 6 days after 14 November 2029. At this time, divert ΔV drops to only 0.26 km/s. Combining that with a constant outbound cost and a rendezvous cost of 3.38 km/s, 20 November 2029 seems like the cheapest cost date for a visit to 2001 AV43.

Table 14. The predicted, calculated, and minimum ΔV dates for the asteroid 2001 AV43.

Predicted Date	Calculated Date	Minimum ΔV Date
5 November 2029	14 November 2029	20 November 2029

The next opportunities shown in Table 8 come from the same asteroid 2001 GP2. For the purposes of the analysis, however, each opportunity will be treated separately as two totally different approaches. First, we will focus on the first predicted approach date of 5 October 2020. Just a note, since the period of the orbit of 2001 GP2 around the Sun is 386.15 days (very close to that of Earth), we would expect the other close approach date to be around 5 October, just some number of years in the future.

This turns out to be the case as Table 7 shows the other close approach date to be 6 October 2057. From Table 12 for 2001 GP2 Opp 1 (we will refer to the first opportunity as Opp 1), the total ΔV is 6.97 km/s. However, the time to 2001 GP2 is the longest we have seen at 11.05 days. Different than the first two options is that the calculated close approach date is the same as the predicted date. Once again, we see a potential trade-off as to the time it takes to arrive versus the amount of ΔV. Since the time to arrival is longer than we would like, let us take a look at what increasing departure, or outbound, ΔV would do to the time and speed values.

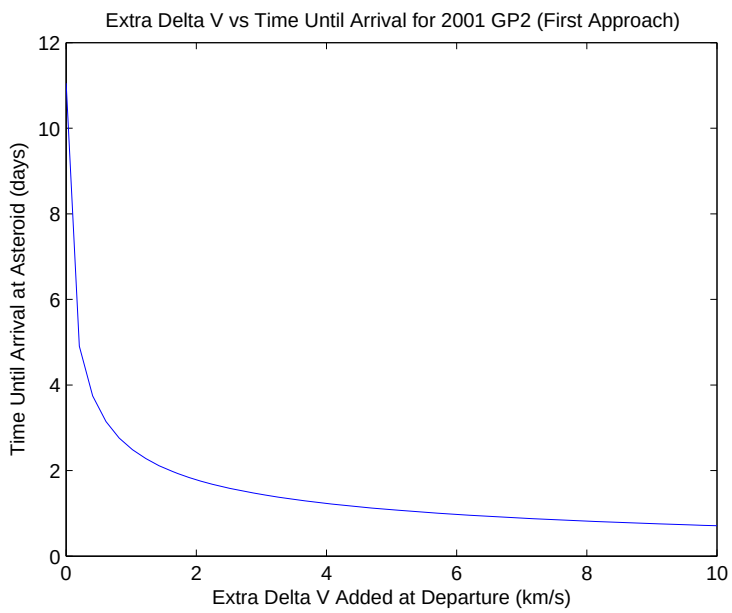

Figure 37. The time until arrival of the spacecraft to asteroid 2001 GP2 versus the added ΔV at departure.

Looking at Figures 37 and 38 we can see that a slight increase of about 0.5 km/s at departure could have a large impact on the arrival time to the asteroid; possibly cutting it down to 4 days or less. However, the speed at arrival will increase by approximately 1 km/s, causing potential increases in rendezvous ΔV. To analyze both options, we can compute the minimum ΔV for a mission with 0 km/s added ΔV at departure and also do it for 0.5 km/s added at departure and see what impact

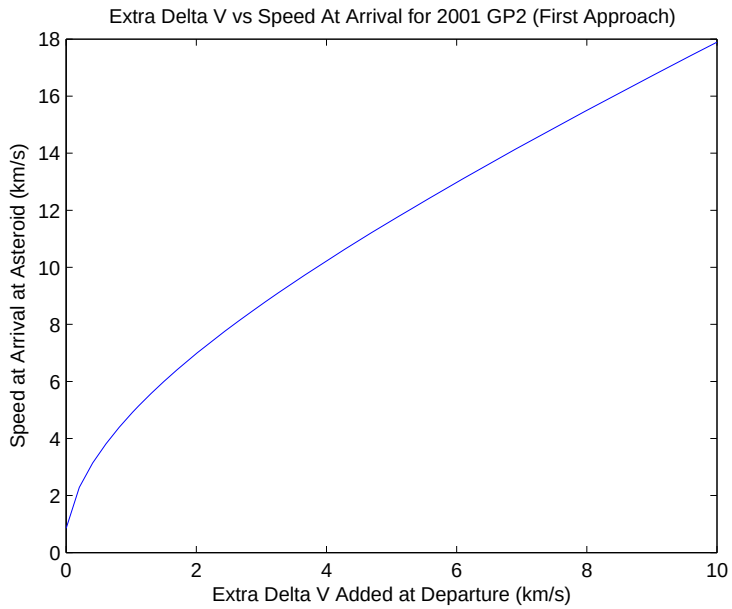

Figure 38. The speed of the spacecraft at arrival at asteroid 2001 GP2 versus the added ΔV at departure.

that would have on ΔV for the mission.

The minimum ΔV for this mission occurs after the calculated close approach date, about 3 days after, and yields a total ΔV of 5.57 km/s. This minimum ΔV value is a bit less than what was computed for a mission to 2001 AV43, so if 0.5 km/s added at departure does not increase ΔV too much, this mission might be the better cost option.

Figure 40 shows that rendezvous ΔV, with an added boost at departure, does not get as cheap as shown in Figure 39. The rendezvous ΔV gets as low as about 4 km/s and the total only comes down to about 8.8 km/s. This is a dramatic increase from the 5.57 km/s we achieved before. Therefore, it is not smart to add that much ΔV at departure in order to speed up the arrival when options such as 2001 AV43 seem more logical and easier to perform.

The next opportunity for 2001 GP2 is on 6 October 2057. This opportunity at first glance looks like a much better prospect than the previous journey of 2001 GP2

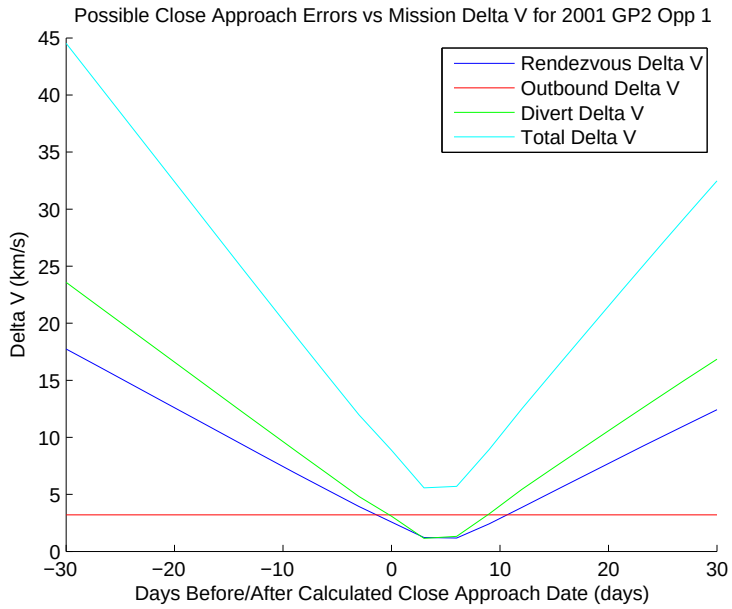

Figure 39. Each component of mission ΔV compared to the calculated close approach date of 2001 GP2 Opp 1 using method 2.

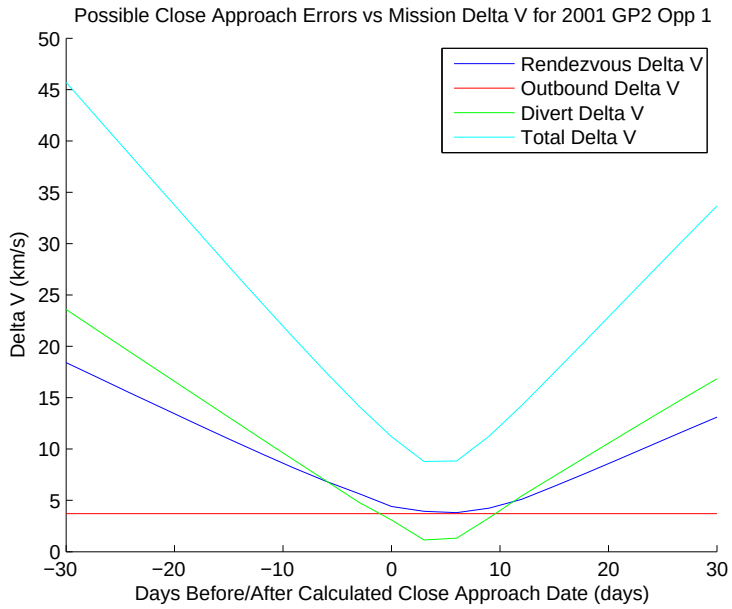

Figure 40. Each component of mission ΔV compared to the calculated close approach date of 2001 GP2 Opp 1 with an added ΔV of 0.5 km/s at departure using method 2.

Table 15. The predicted, calculated, and minimum ΔV dates for the asteroid 2001 GP2 Opportunity 1.

Predicted Date	Calculated Date	Minimum ΔV Date
5 October 2020	5 October 2020	8 October 2020

toward Earth. 2001 GP2 Opp 2 is projected to come within the moon's orbital radius around the Earth at approximately 313,000 km. Because it comes so close, the time until arrival is between 1 and 2 days, and the total ΔV is 7.26 km/s. The fact that the time to the asteroid is so small, any added ΔV at departure is unnecessary and would only create a much more costly mission. With only a 2 day trip time outbound and roughly 2 day return trip time, more time can be spent on 2001 GP2 exploring and researching the area which is something that has not been discussed much yet. The minimum ΔV that can be obtained on a mission to 2001 GP2 is 5.41 km/s, the smallest of any of the previous opportunities. Figure 41 shows similar ΔV trends where rendezvous ΔV drops down to 1.06 km/s leaving the smallest total ΔV at about 6 days after the calculated close approach date. With the lowest ΔV thus far, it is also worth seeing what some added boost at departure would do. Figure 42 shows that total only drops as low as 9 km/s. Again, an additional ΔV cost at departure could have much costlier effects in the long run.

Table 16. The predicted, calculated, and minimum ΔV dates for the asteroid 2001 GP2 Opportunity 2.

Predicted Date	Calculated Date	Minimum ΔV Date
6 October 2057	6 October 2057	12 October 2057

Finally, the last opportunity to look at from Table 8 is 2012 PB20. Compared to the other candidates, at first glance, 2012 PB20 does not compare in regards to time to the asteroid, and total ΔV necessary. With no ΔV added, it would take just over two weeks, about 14.6 days, to arrive at the asteroid with a total ΔV of 8.58 km/s. These values push the boundaries of a short duration mission. The calculated

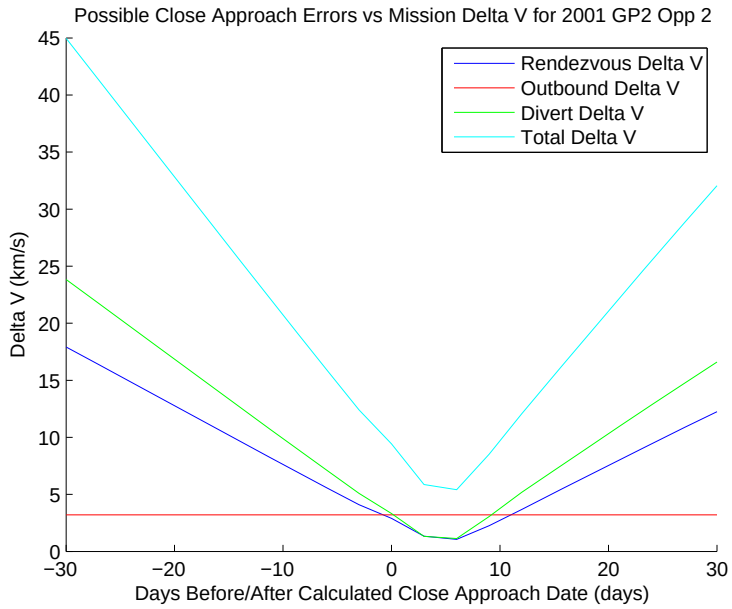

Figure 41. Each component of mission ΔV compared to the calculated close approach date of 2001 GP2 Opp 2 using method 2.

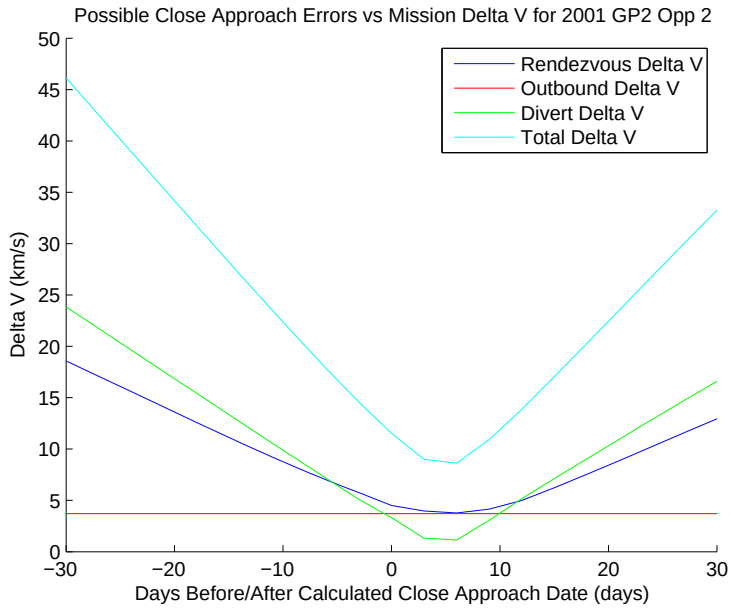

Figure 42. Each component of mission ΔV compared to the calculated close approach date of 2001 GP2 Opp 2 with an added ΔV of 0.5 km/s at departure using method 2.

close approach date is 12 days after 30 January and that is when the asteroid passes within approximately 1.33 million km. When computing the minimum ΔV date, Figure 43 shows more of the same in high ΔV values. The divert ΔV, like previous opportunities, dictates the total but it never drops below about 4 km/s, leaving a high total ΔV. The lowest combined ΔV occurs 6 days after the calculated closest approach and is the only asteroid to have a higher minimum ΔV when the Earth is held fixed. A mission to 2012 PB20 has a mission ΔV of roughly 10.40 km/s. At last we have our final table of differing close approach dates, Table 17.

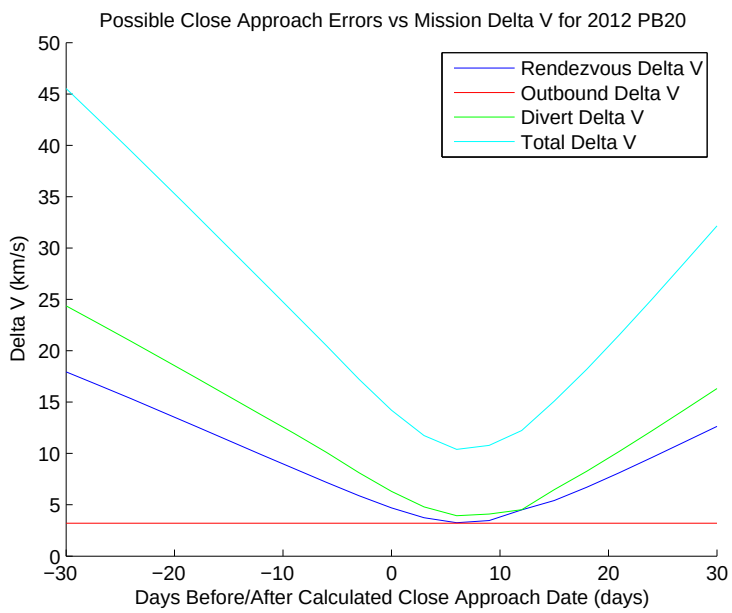

Figure 43. Each component of mission ΔV compared to the calculated close approach date of 2012 PB20 using method 2.

Table 17. The predicted, calculated, and minimum ΔV dates for the asteroid 2012 PB20.

Predicted Date	Calculated Date	Minimum ΔV Date
30 January 2025	11 February 2025	17 February 2025

Finally, we can ask, why does the minimum ΔV date occur when it does in relation to the calculated close approach date? When comparing all of these plots to those

from Section 4.3 and Appendix A, we see that each opportunity has a minimum ΔV date after the calculated close approach date. This means that the minimum ΔV date occurs after the asteroid and Earth are closest together. This is not immediately explainable and is something that is tough to model in STK because of the lack of Earth propagation. However, if we consider Figures 35 through 43, we can try to see why this would occur. First off, divert ΔV is a major factor that we can look at. We know that when both bodies are propagated, the divert ΔV increases as the close approach date approaches, which is what we would expect. The distances between the two are also small enough that the divert value is very small. However, when only the asteroid is propagated, the distance between the two gets so great that the divert ΔV shoots up too.

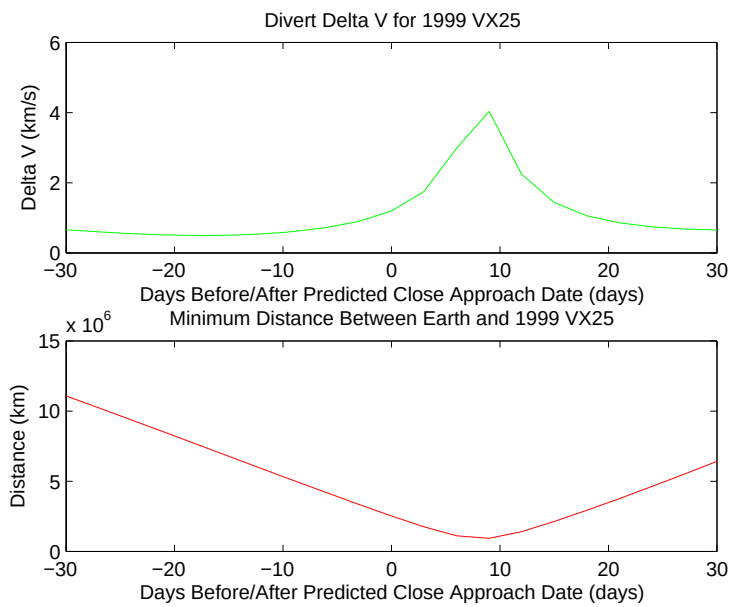

Figure 44. The distance and Divert ΔV component when both bodies are being propagated.

As Figure 44 shows, the divert ΔV component is highest at the calculated close approach date, so it makes sense that in Figure 45 we see the minimum divert cost occur after the calculated close approach date. The same thing is seen with the

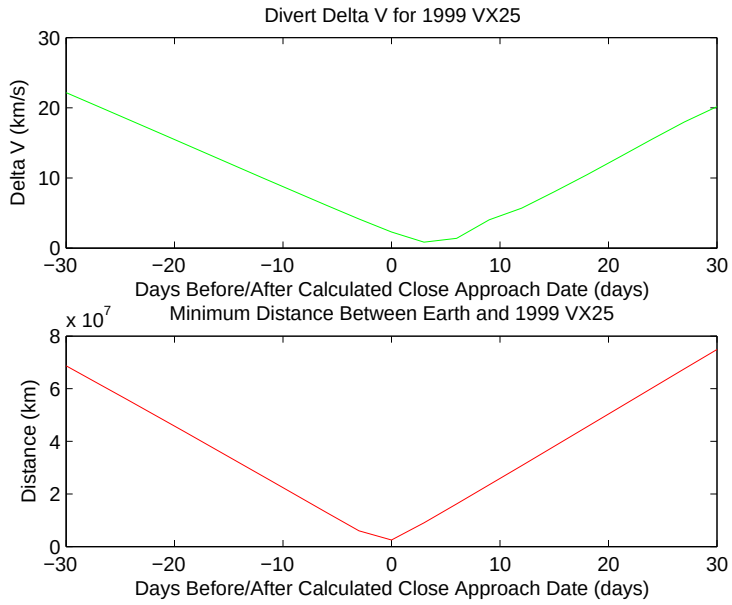

Figure 45. The distance and Divert ΔV component when only the asteroid is being propagated.

rendezvous ΔV and we know that the outbound leg remains constant. Therefore, we would expect the minimum ΔV date to be before or after the calculated close approach date, not right on it. Considering the fact that the asteroid is closer to the Earth after the calculated close approach date rather than before, it makes sense that the minimum ΔV date would fall slightly after. Perhaps the most important factor of all of this is the extra order of magnitude used when plotting distance between the two bodies for a fixed Earth as opposed to a propagated Earth. When both bodies are propagated, the distance between the two of them remains relatively small, on the order of millions (or hundreds of thousands) of kilometers. But when only the asteroid is propagated, it moves so fast that the distance increases greatly and makes mission planning difficult. While the information presented in Section 4.4 provides great insight to optimizing a mission, the conclusions will be developed using the minimum time approach.

5. Conclusions

Now that the results have been presented and analyzed, it is finally time to identify candidates for our three mission possiblities; uncrewed arrival, uncrewed fly-by and crewed arrival. The information presented below is based on the data from Chapter 4 and gives the final conclusions of the thesis. Prior to discussing the possible opportunities for a crewed or uncrewed short duration asteroid mission, the conclusions developed through this research must be presented. Initially, before beginning this research, the thought of numerous short duration mission opportunities was hard to fathom. For that reason, the fact that so many opportunities were developed in this research was a surprise. Secondly, as discussed in Chapter 1, the use of 2-body propagation versus NASA's higher fidelity propagation models was investigated. As Chapter 4 discusses, some results could not be validated through the database; however, enough opportunities were validated to conclude that the 2BP techniques used are sufficient for an initial estimation of short duration mission candidates. Should candidates be chosen and mission trajectories developed, higher fidelity models should be instituted to ensure a correct mission plan, but for an initial study, 2-body techniques do suffice. Utilization of the 2BP with no perturbations could also be seen as a limitation or weakness of this research; however, this limitation still presents an adequate solution with further analysis required in the future. Another major conclusion is that these proposed missions can have an impact on planetary defense. Enough missions are possible that improving orbit determination can be tested and the ability to operate around an asteroid can be optimized. These advancements could be used to detect a potentially Earth-impacting asteroid and help to deter it from impact. Being able to implement multiple short duration missions can give numerous opportunities for practice of planetary defense techniques. Many attempts and opportunities can lead to a working solution to protect the planet.

5.1 Candidates for Uncrewed Missions

There are two types of possible uncrewed missions. They include an arrival on the asteroid or simply a fly-by where the spacecraft approaches and passes by but does not actually touch down on the asteroid. Both options have their pros and cons and candidates will be covered in the following sections.

5.1.1 Uncrewed Fly-Bys

An uncrewed fly-by is a difficult mission to consider because of the relative lack of excitement by the public. To have an asteroid come so close and prepare a mission to visit it without actually stopping to visit and gather samples and information seems a bit unnecessary. However, there is a perfect opportunity that presents itself for a mission like this and that is the first pass of 2001 GP2. This asteroid is unique in that two of the best opportunities came from the same asteroid on its close approaches. Table 12 shows that for 2001 GP2. The first opportunity with no added boost at departure, has a mission length of 30 days with a total ΔV of only 6.97 km/s. This makes for an absolutely ideal mission. When comparing an uncrewed fly-by to an uncrewed arrival, the major difference is ΔV and the need to rendezvous with the asteroid and match its speed to eventually approach the surface and touch down. In other words, the total ΔV can actually be less than the 6.97 km/s that was calculated before. However, should the rendezvous leg be dropped and the spacecraft simply just turned around the asteroid, time for the return leg will be longer than anticipated because the spacecraft is no longer moving as fast as the asteroid or in the same direction and will simply loop around it. The amount of time projected for the mission is only 30 days. People will still be able to be motivated about the mission but the cost is not excessive or outrageous.

The first pass is an excellent option for an uncrewed fly-by mission because of the

ability to learn from the asteroid and prepare for the second pass. The time between passes is very lengthy (37 years) but not too long to think that a mission to both asteroids would be unreasonable. A fly-by of 2001 GP2 at its first close approach would give us adequate amounts of information about the size, shape, terrain, and overall body characteristics of the asteroid. The information could lead to optimal trajectories for the second mission, helping to minimize time and cost. In a more advanced look, seeing and mapping the terrain on the fly-by could lead to potential new technology developed for the crewed return. With all of the time between the approaches a new lander or spacecraft can be developed purely for the trip back to 2001 GP2 some 30 years later. Often times, an uncrewed arrival will be used to explore and gather more information than one on a fly-by mission. However, 2001 GP2 is a rare instance in that it has another close approach somewhat soon after its first approach. So why not save ΔV on the first trip by simply flying by and gathering information and putting ourselves in the best position for a crewed mission 37 years down the road?

5.1.2 Uncrewed Arrivals

The first asteroid that comes to mind for an uncrewed rendezvous mission is 2012 PB20. This mission fits the requirements for an uncrewed arrival perfectly. The key aspect is length of the mission, which Table 12 tells us is 36 days. This is greater than the short duration missions we are looking for in terms of a crewed mission, but the length of time lends itself to an uncrewed mission. Much like the 30 day mission to the first opportunity of 2001 GP2, the 36 days will still keep a motivated public while not exceeding our capabilities for a mission. In addition, the total ΔV is 8.58 km/s, but this can also be decreased in exchange for an increase in time. Another consideration could be an added burn at departure, but this is unnecessary. Though

time drops substantially, the new rendezvous ΔV causes the total ΔV to jump to approximately 12 km/s, and this value might be too extreme for an uncrewed mission.

Something to be considered with uncrewed arrivals on an asteroid is the possibility of using one of the opportunities outside of the five selected in Table 8. With an uncrewed mission, time is not of nearly as much importance. Sure, we do not want a decade long mission when an asteroid is coming very close in the first place, but if we can save some ΔV by extending the mission by a week or two, that is something to take advantage of. Also, a month or two long mission would keep the public very involved as this is the first United States spacecraft to land on an asteroid. Looking at Figure 27 we can look at the asteroids with longer time until arrival and lower ΔV values. For these reasons, 2008 LG2 and 2012 UX136 can also be considered for an uncrewed arrival mission. The mission data for these two asteroids can be seen in Appendices A and B.

2008 LG2 represents another excellent uncrewed arrival mission, very similar to 2012 PB20. We have another mission of about a month in duration with a minimum time ΔV of 9.17 km/s. Another thought when considering multiple candidates like this is the time over which the mission would take place. 2012 PB20 would occur in 2025 while a mission to 2008 LG2 would not occur until 2049. Splitting the two is 2012 UX136 in the year 2037. Once again we see a mission with a duration of 29 days and about 9.68 km/s of ΔV. With these three missions being very similar in length and cost, all three can be recommended as candidates for an uncrewed rendezvous mission. The only thing to distinguish them may be the timing of the trip. Because of this, 2012 PB20 might be the best option.

5.2 Candidates for Crewed Missions

A crewed mission is under the highest scrutiy because of the possibility of not only loss of technology, but loss of life. Therefore, a crewed mission should undergo more scrutiny than an uncrewed mission or fly-by. For this reason, only the best opportunities should qualify for crewed missions. An uncrewed mission that arrives on the planet is not a bad mission option, especially for an asteroid that does not have a lot of information available, like most of these asteroids. For well observed asteroids, though, a crewed mission makes sense to get the most out of the mission and get the most information possible. They are the most dangerous due to the possibility of loss of life. Not only has the United States never put humans on an asteroid but never has it even put a probe on an asteroid. So, while the rewards are sky high, the risks are very high as well.

The three opportunities that stick out for a crewed mission are the aforementioned second approach to 2001 GP2, 1999 VX25, and 2001 AV43. All of these asteroids present perfect opportunities for a crewed mission due to their short duration mission length and relatively low fuel cost. 1999 VX25 has a close approach opportunity in 2034 with a projected length of 24 days and only 7.90 km/s of ΔV required. This length is a bit on the high side for a crewed mission but definitely something that is achievable. The problem with this mission may be the amount of time before the mission is required. Although there is a wait time of about 16 years until the mission would take place, planning a mission of this magnitude could take even more time. Perhaps the toughest part is government approval and getting funding for a mission that has never taken place before. Looking at 2001 AV43, the cost is a bit higher at 8.25 km/s but the duration is only 2 and a half weeks. This creates another excellent opportunity for a mission. The close approach date is 2029, though, and if 2034 was going to cause timing problems, this asteroid may cause even more. The six day

difference in duration is not very extreme, and in order to save over half a km/s of fuel, the 1999 VX25 option seems to make more sense.

As discussed in Section 5.1.1, a crewed mission to 2001 GP2 on its second close approach opportunity with a precursor fly-by mission to the first close approach opportunity makes the most sense. The required ΔV is the lowest of the three crewed missions at 7.26 km/s, and the duration is by far the shortest at 9 days. This is due to the fact that the asteroid passes within the moon's orbit. This mission, because of its precursor mission redundancy, stands out above the rest. Though 2057 is a long ways down the road, adequate preparation and proper use of the fly-by data will only help lead to a flawless mission. In addition, the cost could be decreased based on results from the earlier fly-by mission. Though not absolutely necessary for a crewed mission (see Apollo 13 [34]), perfection is paramount and any mistakes can lead to a loss of life. Therefore a precise mission with a mapped asteroid and technology developed for 2001 GP2 and 2001 GP2 only is an excellent option. Another benefit is the fact that lives could be saved from the precursor mission should the fly-by not go as planned. For example, if the first mission does not reveal a plausible landing site, then no time or money is wasted by planning a second crewed mission. Or perhaps the composition of the asteroid is learned to be not very dense and is not conducive to a human visit. Once again the time and money put into a crewed mission can be saved and put toward another mission to a different asteroid.

In addition to the three options presented above, 2009 QR is another close approach opportunity in the year 2037 that presents a very short mission duration. This opportunity can be looked at as an alternative to 2001 GP2 while also keeping mission duration under 2 weeks. As Table 19 in Appendix B.1 shows, the mission would be on the order of 8-12 days. When no boost is added at departure, the total ΔV is around 9.32 km/s which is definitely possible. This asteroid was not originally

considered because of the high ΔV total presented in the Figure 27, but review of its short duration capabilities led to the addition of the asteroid as a possible candidate. Due to the fact that the close approach opportunity is in 2037, 20 years prior to 2001 GP2's close approach, this opportunity may present a stronger candidate. More information is included in Table 19 and Figure 50 in the Appendices.

5.3 Future Work

Though an extensive amount of time and effort was put into this research, there is always more that can be done. The first thing to focus on would be to perform ΔV analysis, time analysis, and mission planning for all 13,065 asteroids in the catalog. The constraints do an excellent job of limiting the bad opportunities that come through, but it is possible that some missions present a short duration or ΔV values that still do not meet the criteria. This is highly unlikely, but definitely possible and therefore should be considered in future work. Another recommendation is to implement more than 2-body dynamics into the model, complete with perturbations, to help estimate higher accuracy close approaches. The development of actual trajectories and mission planning is also recommended with the use of these more accurate dynamics models to expand the projected short duration missions.

Something that was covered for every asteroid in the catalog was determining close approach dates. Many of these dates fell within the limit of the year 2080 but many also didn't. Extending this date to account for more opportunities is something that can be done in the future. Before that though, it is essential to figure out the problems between the calculations of close approaches and why all of them did not match up with the NASA/JPL database. This has already been discussed in detail and maybe is as simple as the lack of orbital data beyond the Earth crossing point leads to poorly defined orbits. However, some major differences lead me to think differently. Perhaps

the major culprit is the asteroid Apophis that has been studied extensively for a close approach in April 2029. The calculations produce the exact same close approach date; however, the details of the close approach are very different. NASA predicts the asteroid to come within the orbit of the moon, but the calculations from Chapter 3 did not even come close to one lunar distance away [35]. This is something that needs to continue to be looked into to figure out if there are errors in either model.

Finally, we developed a minimum ΔV date that was based on potential errors with the close approach date. In order to estimate this date, we kept the Earth fixed in its orbit throughout the asteroid propagation before and after the close approach date instead of propagating it with the asteroid. However, something to look into could be combining both approaches to estimate the minimum ΔV date when the Earth is propagated. For that matter, adding in mission planning to develop a more accurate minimum ΔV date as well could have a great impact on the model. This means calculating a minimum ΔV date based on the shortest time to arrival, arbitrary length of stay on the asteroid, and diverting back to Earth. Once again, while this mission planning was done with a few asteroids, being able to do this with the entire catalog would be very helpful.

5.4 Review of Objectives

The objectives for this research were to:

1. Search the NEO catalog and compute when asteroids cross the Earth's orbit
2. Determine where the Earth is at the crossing point and see if a mission is possible
3. Analyze that information to determine the best type of mission
4. Use that knowledge to recommend a mission plan

5. Validate results versus other calculated results

6. Identify close approach candidates

The NEO catalog was investigated and every asteroid's crossing date was researched. After that, the Earth was propagated and close approach distances were developed, allowing each asteroid to go through numerous restrictions until possible opportunities were left. From here, a mission plan was developed and these results were checked with the NASA/JPL database. Finally, several candidates for asteroid close approach missions were identified. Clearly, each one of these objectives were met in order to develop mission plans and identify close approach candidates.

In conclusion, numerous asteroids have been identified as candidates for crewed and uncrewed missions. The background of the research and the motivation were set in place and the methods to identify these opportunities have been thoroughly discussed and developed. To revisit our quote from President Obama from Chapter 1, "Our goal is the capacity for people to work and learn, and operate and live safely beyond the Earth for extended periods of time...And in fulfilling this task...we will strengthen America's leadership here on Earth" [3]. This thesis has presented ample opportunities to begin working toward living beyond Earth and understanding the space environment around us. This education will only help strengthen America's ability to lead both domestically and internationally.

Appendix A.

This appendix contains extra figures not shown in the main text.

A.1 Mission ΔV

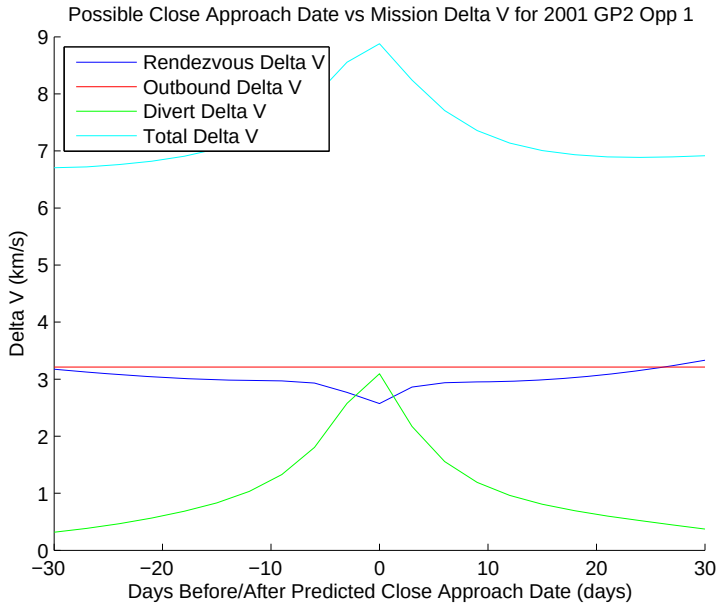

Figure 46. Each component of mission ΔV compared to the predicted close approach date of 2001 GP2 Opp 1 with no ΔV added at departure.

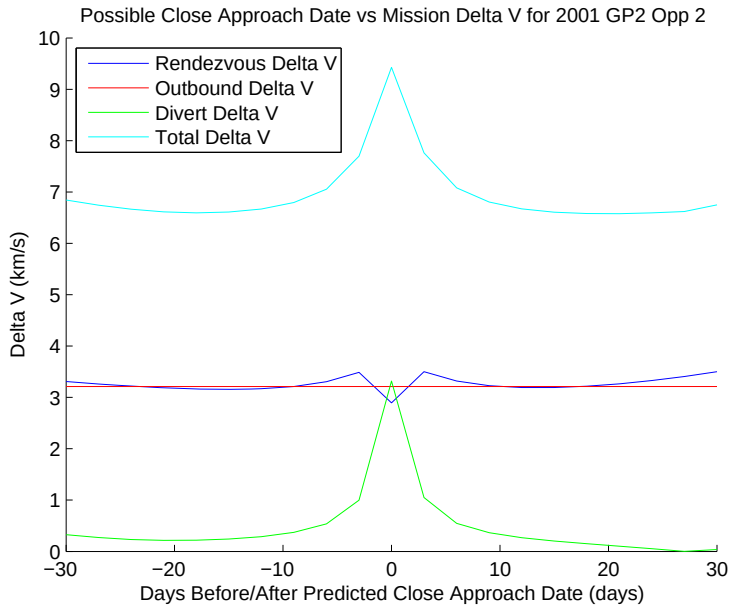

Figure 47. Each component of mission ΔV compared to the predicted close approach date of 2001 GP2 Opp 2 with no ΔV added at departure.

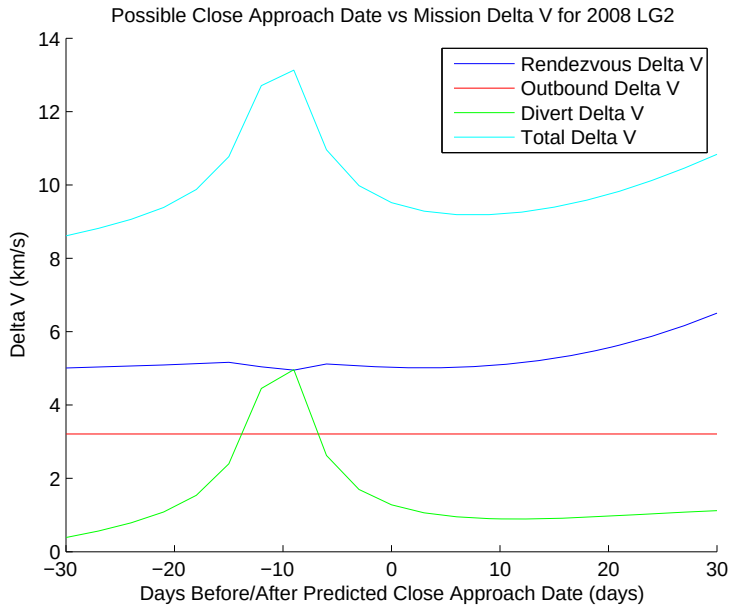

Figure 48. Each component of mission ΔV compared to the predicted close approach date of 2008 LG2 with no ΔV added at departure.

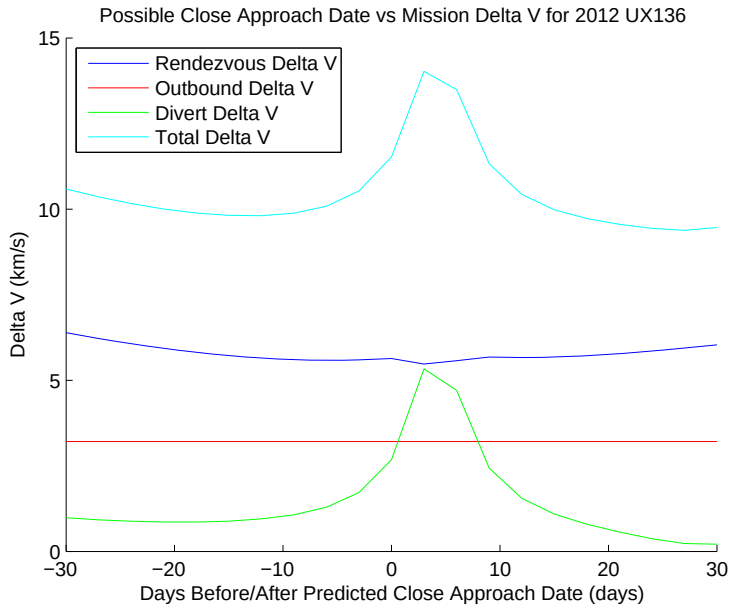

Figure 49. Each component of mission ΔV compared to the predicted close approach date of 2012 UX136 with no ΔV added at departure.

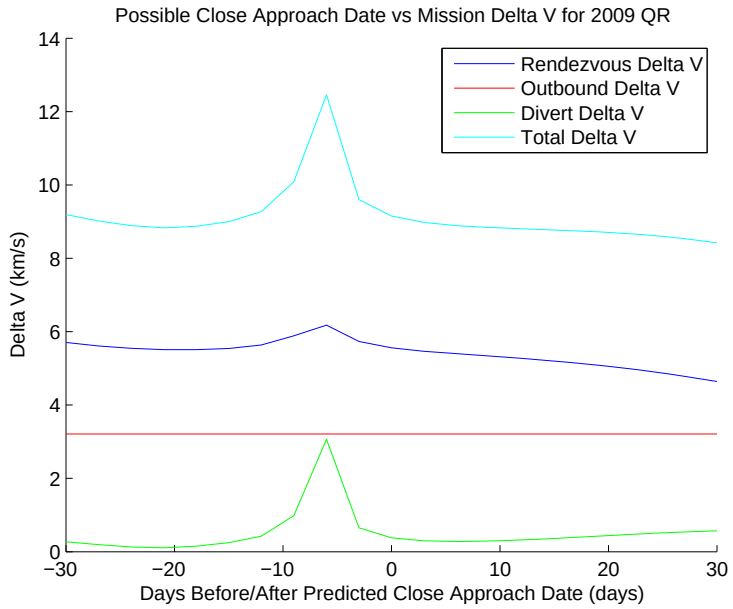

Figure 50. Each component of mission ΔV compared to the predicted close approach date of 2009 QR with no ΔV added at departure.

Appendix B.

This appendix contains extra tables not shown in the main text.

B.1 Mission Planning

Table 18. The effects of added boost at departure on mission planning for 2008 LG2 and 2012 UX136.

Asteroid	Added Boost (km/s)	Mission Length (days)	Total ΔV (km/s)
2008 LG2	0	32	9.17
	0.5	13	13.08
	1	13	14.93
2012 UX136	0	29	9.68
	0.5	11	13.75
	1	11	15.48

Table 19. The effects of added boost at departure on mission planning for 2009 QR.

Added Boost (km/s)	Mission Length (days)	Total ΔV (km/s)
0	12	9.32
0.5	8	12.41
1	8	14.21

B.2 Close Approach Dates

Table 20. The predicted and calculated ΔV dates for the asteroid 2008 LG2.

Predicted Date	Calculated Date
29 June 2049	20 June 2049

Table 21. The predicted and calculated ΔV dates for the asteroid 2012 UX136.

Predicted Date	Calculated Date
1 November 2037	4 November 2037

Table 22. The predicted and calculated ΔV dates for the asteroid 2009 QR.

Predicted Date	Calculated Date
30 August 2037	24 August 2037

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14. ABSTRACT

My investigation of the Near Earth Object (NEO) catalog has led to identification of numerous short duration, under 40 days, mission opportunities in the future for three different mission types: uncrewed fly-by, uncrewed arrival, and crewed arrival. 2-body propagation techniques were used to model the orbits of various asteroid candidates and the Earth to determine when a close approach would occur. Once the dates were calculated, distance between the bodies was computed to estimate the ΔV to complete the mission. From the mission ΔV values, a possible mission duration was also computed. The values were analyzed to determine the best options for the mission types described above. One candidate is presented for the uncrewed fly-by opportunity, three for the uncrewed arrival mission, and four more for a potential crewed mission. The results show that a short duration mission is not only possible but should be strongly considered in the near future. These short duration missions are in sharp contrast to the common multi-month or year long duration proposals. Among the other wealth and resource benefits, short duration asteroid missions are of supreme importance for planetary defense and maintaining a powerful US space presence.

15. SUBJECT TERMS

Asteroids, NEO, Crewed Missions

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