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The First Human Asteroid Mission: Target Selection and Conceptual Mission Design

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President Obama has recently declared that NASA will pursue a crewed mission to an asteroid by 2025. This paper identifies the optimum target candidates of near-Earth objects (NEOs) for a first crewed mission between 2018 and 2030. Target asteroids in the NEO database with orbital elements that meet predetermined requirements are analyzed for mission design. System architectures are then proposed to meet the requirements for five designated NEO candidates. Additional technology which can be applied to crewed NEO missions in the future is also discussed, and a previous NEO target selection study is analyzed. Although some development of heavy launch vehicles is still required for the first NEO mission, design examples show that a first mission can be achieved by 2025. The Ares V and the Orion CEV are used as the baseline system architecture for this study.

Nomenclature

a	Semi-major Axis (AU)
AU	Astronomical Unit
BOM	Burnout Mass
CEV	Crew Exploration Vehicle
e	Eccentricity
EDS	Earth Departure Stage
EOR	Earth Orbit Rendezvous
H	Absolute Magnitude
i	Inclination (deg)
I_{sp}	Specific Impulse (s)
ISS	International Space Station
JPL	Jet Propulsion Laboratory
LEO	Low-Earth Orbit
LH_2	Liquid Hydrogen
LOX	Liquid Oxygen
MW_e	Megawatt Electrical
MT	Metric Ton
$NASA$	National Aeronautics and Space Administration
NEO	Near-Earth Object
OTV	Orbital Transfer Vehicle
$RSRM$	Reusable Solid Rocket Motor
$SSME$	Space Shuttle Main Engine
TLI	Trans Lunar Injection
$VASIMR$	Variable Specific Impulse Magnetoplasma Rocket
ΔV	Velocity Change (km/s)

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I. Introduction

There is a nationally growing interest in a crewed mission to a near-Earth object (NEO) as part of a flexible path option of the U.S. human space exploration program. When referring to the future of manned spaceflight, President Obama announced during a recent trip to Kennedy Space Center on April 15, 2010, "We'll start by sending astronauts to an asteroid for the first time in history." Such a new space exploration initiative by NASA for achieving a first human asteroid mission by 2025 has been reassured by President Obama's U.S. National Space Policy on June 28, 2010. Although returning to the Moon and a mission to Mars are two suggested options for the next crewed mission, it would be more logical to first visit an NEO. A crewed mission to a designated NEO presents new challenges to human space exploration, which could help establish a new era of human spaceflight. To plan for the first crewed mission to an NEO, target asteroids must be evaluated, various mission lengths must be tested, and multiple system architectures must be analyzed. Only missions between 2018 and 2030, and mission lengths of 45 to 200 days will be considered in this paper. Current Orion CEV and Ares V designs will serve as the baseline system architecture for the study. Although these designs are used, it is important to note that the system architectures which utilize the designs are provided to represent the performance capabilities of a moderately sized crew vehicle and an Ares V-class launch vehicle.

A. NEO Background

Comets and asteroids whose perihelion is less than or equal to 1.3 AU, and whose orbits approach and/or cross Earth's orbit are known as near-Earth objects. For this study no comets will be considered because of the low eccentricity and semi-major axis mission constraints (to be discussed in detail in following sections). NEOs typically range in size from several meters to tens of kilometers. However, the smaller objects drastically outnumber the larger asteroids. Many NEOs have orbits very similar to Earth's orbit, which makes them more desirable targets for crewed missions. Some asteroids can even be reached with lower ΔV requirements than previous lunar missions. This is mostly because ascent and descent burn ΔVs for a lunar landing can be higher than those required for asteroid arrival and departure burns. In 1998, NASA accepted a mandate to detect and catalog 90 percent of NEOs larger than 1 km. The 2005 NASA authorization act mandates NASA to detect and characterize NEOs down to 140 m in diameter. There are currently more than 6600 known NEOs, and that number is expected to increase over the next decade. It has been predicted that up to 90 percent of all NEOs with a diameter of 140 m or more will be detected by 2020. With the list of asteroids constantly growing, it is extremely important to develop a computer program capable of determining all the possible crewed NEO targets automatically with limited user input required.[1]

Previous studies have also examined the topic of sending a crewed mission to an NEO. During the 1960's, the asteroid 433 Eros was researched as a possible candidate for a crewed mission using Saturn V and Apollo technology. This study considered a crewed mission of over 500 days around 1975, when 433 Eros had a close approach of roughly 0.15 AU with Earth.[2] Although the proposal examined very long mission durations, it demonstrated a significant first step in the consideration of a crewed NEO mission. Additional studies were performed as a part of the Space Exploration Initiative in the 1980's, however interest in the topic faded until the Constellation Program began to take shape. These more recent studies have revisited the possibility of sending a manned mission to an NEO. Sparked by the development of the Constellation Program, these more recent studies focus on the feasibility of using such hardware for a 90 to 180-day mission to a strategically selected NEO. Optimum launch dates for possible target asteroids and Constellation-configured technology for system architectures are considered in these studies.[3,4]

The asteroid 99942 in particular has received much attention in recent years, due to its possible, yet unlikely Earth impact in 2036. For this reason, Apophis has been considered in Ref. 5 as a potential candidate for a crewed mission in 2029. However, ΔV requirements have shown to be high for a crewed mission to Apophis.[5] A crewed mission to a strategically selected NEO between 2018 and 2030 would be a better fit for the first mission.

B. Mission Objectives - Rationale for a Human Asteroid Mission

After the retirement of the space shuttle, the United States will be left without crewed launch vehicles. If deep space exploration, returning to the lunar surface, or visiting Mars are eventual goals of NASA, new technology is necessary to achieve these goals. Because the Apollo 17 mission in 1972 was the last human spaceflight to go beyond LEO, it is important to validate any new crewed system technology with a deep space mission. A well-selected NEO mission would help a new crewed space system architecture gain flight experience and prove its new technology. Apollo 17 had a mission duration of just over twelve and a half days, making it the longest of the Apollo missions. Any crewed NEO mission would involve spending a significantly longer time outside of LEO. Also, the mission would be the first human mission to venture outside of the Earth-moon system. The goal of visiting an asteroid could lead to the development of safe and reliable deep space operations.

Sending astronauts to an asteroid would be a very logical next step to deep space exploration. Although visiting an asteroid can involve a complex system architecture, the development of a lander would not be required for the mission. Most asteroids in consideration for the first mission are small enough that microgravity is too small to allow for a landing. The absence of a lander can reduce ΔV requirements and increase launch payloads. Also, missions to NEOs could eventually be extended to simulate a long-duration trip to Mars.

A sample collection would be one of the most significant accomplishments that could come from a crewed NEO mission. Although the Hayabusa probe may have returned an asteroid sample earlier this summer, a manned mission would be capable of returning a more sizable sample. Humans could also spend more time on the asteroid assessing the surface's nature and composition. Composition studies on asteroid samples could be important for a number of reasons. These studies could give scientists clues and insight into the origin and formation of our solar system. Also, studying an asteroid's composition could greatly benefit asteroid deflection research. Computational studies for validating the effectiveness of deflection techniques could become more accurate if the composition and density of asteroids is more accurately known. Also, if in the future a manned deflection mission were necessary to divert a hazardous NEO, it would be important to have already flight validated, human asteroid mission. Asteroids could also one day be resources for the colonization of deep space. Their minerals could possibly be used for the building of space structures, or the presence of water could be used be used as a fuel source. Although this may still be years off, asteroid composition research could help advance the possibilities of mining asteroids.

C. Mission Design Considerations

When examining mission design results for target asteroid selection, there are many factors to consider. Asteroids which present minimum ΔVs are prime candidates since they are easier to reach. However, choosing an asteroid based only on this criteria can be misleading. Other aspects of the mission design play an important role as well. Since the Earth departure occurs from LEO, gravity losses will affect the maneuver. Therefore, the required Earth departure ΔV should be increased by 10 percent to account for any gravity losses. This being the case, mission designs with lower Earth departure burns would be ideal. In addition, the timing of each burn also plays a role in making a mission possible. Since cryogenic fuels offer the highest I_{sp} capabilities, it is the top choice for propellant in these missions. However, storing cryogenic fuels, like liquid hydrogen for extended periods of time is very difficult. Since liquid hydrogen cannot be formed simply by pressurizing it, refrigeration cycles and insulation are used to maintain an extremely cold temperature. This can only be done for short periods of time. Therefore, missions in which all of the burns are completed within eight weeks of Earth departure can solve this problem. Otherwise, boil-off of propellant can lead to insufficient ΔV capabilities. Current cryogenic tank technology has limited boil-off to some extent. Typical tanks boil off just under four percent of their liquid hydrogen each month, while liquid oxygen loses less than one percent each month.[6]

Asteroids that are selected should also possess launch windows that are not too narrow. In case of a short launch delay, the total mission ΔV should not increase substantially. Since minimum ΔV requirements for a mission usually occur around the time of an asteroid's close approach, it is often the case that two launch windows will exist very close to one another. The first launch window occurs just prior to the close approach, where a return from the asteroid requires very little ΔV. The second launch window occurs right at the close approach, where a short trip to the asteroid is followed by a longer return flight.

It is important to note that robotic (unmanned) satellites will almost certainly be sent on rendezvous missions prior to the crewed mission. The ΔV requirements should be lower for the precursor robotic mission, since no asteroid departure maneuver is necessary. The satellite would be able to accurately track and monitor the target asteroid's orbit, in addition to mapping the asteroid's surface. By more accurately characterizing the asteroid's composition, rotation rate, orbit, and surface conditions, NASA and the crew will be more prepared for the mission. The satellite could also monitor the asteroid after the crewed mission has departed for Earth. It would be capable of relaying back information on orbital measurements or the effects of a deflection payload if one is left by the crew. In any case, the inclusion of a satellite as an additional observer in a crewed mission could be a valuable asset in many different ways.

D. Asteroid Characteristic Considerations

Mission design plays a large part in determining destinations for a crewed mission to an NEO, but other factors must be considered as well. An asteroid's physical characteristics can play a role in determining whether or not it would be a good candidate for a crewed mission. Ideal target asteroids should have small rotational rates, making it easier for astronauts to maneuver during the mission. Target asteroids should also have a sizable diameter. NEOs with diameters less than 30 m may be too small to be of interest. Also, larger asteroids will be easier to target for guidance and other purposes. Extended launch windows for a target asteroid are also very desirable, since they can add mission design flexibility. Choosing an asteroid that exhibits small or non-existent gravitational irregularities would also benefit the safe proximity operations near the surface of an asteroid. However, since most of the candidate NEOs are under 100 m in diameter and have relatively weak gravitational fields, this may not be a strong concern.

Recently, there has been evidence that the asteroid 24 Themis actually contains large amounts of ice. In the near future, this may be an additional reason to plan a crewed mission to an NEO. If the existence of ice on asteroids becomes a more observed phenomena, a target NEO which contains ice may be designated for the first crewed mission. This rationale may present a new list of target asteroids for consideration and additional mission design work may need to be done. However at this time, there is little information regarding the commonality of ice on asteroids. The finding may prove to be an irregularity, meaning other asteroids do not exhibit the same composition. The opposite may also be true. In the future, scientists may identify a type of asteroid that may also contain water-ice. If studying this type of asteroid were to be a key objective of the first crewed mission, it may change the overall system architecture. The ΔV requirements will most likely be larger since the target asteroid is selected due to its composition, not its optimized mission design. Therefore, additional launches may be required to increase the overall ΔV capabilities of a desired system architecture.

E. Target Search Background and Assumptions

Using software developed in-house by the ADRC, we have obtained mission design results for various asteroids and mission lengths. The program utilizes ephemeris files for each NEO that is considered, which is available from NASA's JPL website. To find the minimum ΔV for each mission length and asteroid, the ephemeris data is used to solve Lambert's problem each day from 2018 to 2030. Since the list of known NEOs currently includes over 6,600 asteroids, running test cases for all of them would have resulted in extremely high computation times. Limits on the semi-major axis, eccentricity, and inclination of the orbits were used to filter the list. The list was narrowed down to about 880 asteroid candidates before testing began.

The limits on the eccentricity, semi-major axis, and inclination of an asteroid are assumed as

- e $<$ 0.5
- 0.5 $<$ a $<$ 1.5 AU
- i $<$ 10 degrees

Asteroids which have close approaches to Earth within 2018 to 2030 have a better chance of yielding low ΔV requirements, since the distance between the Earth and the asteroid is at a minimum. Launch windows for such asteroids occur either prior to or at the asteroid's close approach. Launching prior to the asteroid's close approach allows for short Earth-arrival distance, while launching at the close approach minimizes the Earth-departure distance.

Upon asteroid arrival, the Orion CEV can safely mirror the asteroid's orbit, and a tethering or suit maneuvering system can be used to transport the crew to and from the asteroid. Small station-keeping effort should be sufficient to maintain the Orion CEV's position relative to the asteroid. Mission lengths between 45 and 180 days were tested in 15 day increments. Mission lengths of 200 days were also considered, even though the upper endurance limit of all proposed system architectures is approximately 180 days. This mission length is tested to see if any significant ΔV decreases are achieved by extending the mission's duration by 20 days. If it is the case that this extension offers substantial benefits, it is possible that system architectures could be reconfigured to meet additional requirements. The mission design assumed that Earth-departure occurred from a 185 km circular orbit and that the Earth-return speed is limited to 12 km/s, which is the projected capability of the Orion CEV. The target search program also assumed a 10-day stay at the target asteroids. A more detailed analysis of how the mission design and target selection program functions can also be found in Ref. 1.

II. Target Search Results and Optimum Target Selection

After running the computer code to narrow down the NEO candidate list, the next step was to select five asteroids that would be the best targets for a first crewed mission. For this task, the results of the-180 day test cases are ordered from lowest to highest ΔV requirements. After this is done, asteroids that are deemed too small are removed from consideration. As previously stated, asteroids with diameters less than 30 m are not considered as targets of interest. The five optimum target NEOs, their diameters and absolute magnitudes, and their minimum 180-day ΔV requirements are provided in Table 1.

Table 1. Optimum Target Asteroids and 180 Mission Length ΔVs

Designation	Diameter (m)	H	Min ΔV
2000 SG344	37	24.8	4.695
1999 AO10	60	23.8	6.704
2001 QJ142	72	23.4	6.813
2009 OS5	70	23.6	6.842
2003 LN6	43	24.5	6.975

A. 2000 SG344 Design Example

The first design example focuses on a 105-day mission to 2000 SG344. This design example is chosen to illustrate one of the only short duration mission cases that is possible with reasonable system architectures, which will be explored in more detail later. The 12 year launch window is shown in Figure 1. In Figure 2, an expanded view which focuses on the optimum launch window for this mission can be seen. This launch window presents a number of launch dates that would be adequate if the optimum launch date is delayed due to weather or technical problems.

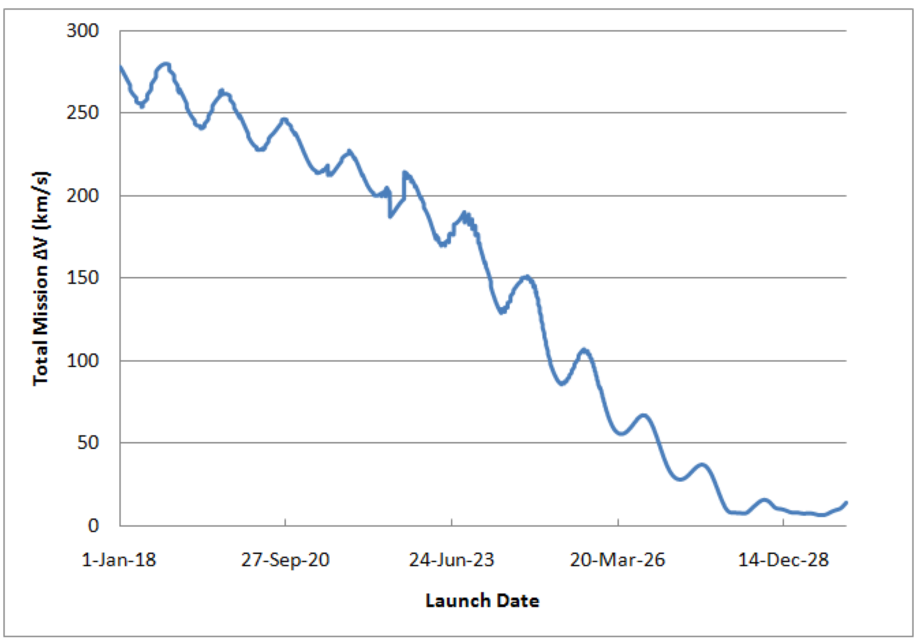

Figure 1. Mission ΔV vs. Launch Window for 105-Day Crewed Mission to 2000 SG344

Table 2. 105-Day Mission to 2000 SG344

	Required ΔV (km/s)	Date
Earth Departure	3.4000	1-Aug-2029
Asteroid Arrival	1.1988	17-Sept-2029
Asteroid Departure	1.7868	24-Sept-2029
Earth Arrival	0.0000	14-Nov-2029
Additional Margin	0.4900	
Total ΔV Required	6.8756	

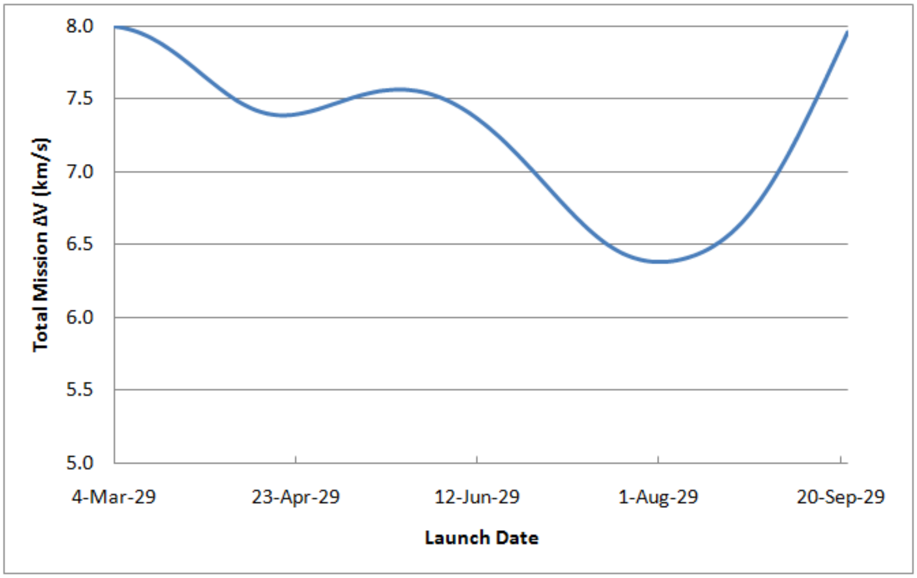

Figure 2. Optimum Launch Window for a 105-Day Crewed Mission to 2000 SG344

Minimizing the ΔV for any mission is important, but it is also important to make sure that the four primary maneuvers for the mission are feasible. Mission maneuvers play an important role in determining if a mission design can actually be performed. The breakdown of the different mission burns can be seen in Table 2, along with the date in which they are to occur. The additional ΔV margin that is applied is a combination of two values. The first is a 0.15 km/s margin for station-keeping and minor propellant boil-off, while the second value is 10 percent of the Earth-departure burn which accounts for gravity-losses.

The breakdown shows a total ΔV of around 6.88 km/s for the 105-day mission. The mission breakdown also illustrates an important aspect of designing a system architecture for this mission. The asteroid departure burn is about 1.79 km/s, which cannot be performed by the Orion CEV, which will be further discussed later in this paper. With this being the case, an attached upper stage is required. In this case, the last burn occurs only 57 days from Earth-departure. If a cryogenic architecture is chosen, propellant boil-off would not pose significant problems for the system design.

For a visual representation of the transfer orbit of the 105-day mission, two orbital trajectories are shown. The first trajectory is in the heliocentric reference frame which is shown in Figure 3, while the second trajectory is in the Earth-fixed reference frame and can be seen in Figure 4.

This NEO has recently been receiving attention due to its once high impact probability and its Earth-like orbit. However, there is still some uncertainty as to what this object actually is. Although it is currently accepted that it is actually an asteroid, it is possible that the object could be a leftover booster from the Apollo program. Since the asteroid follows Earth's orbit so closely, it is possible that 2000 SG344 is actually an object that came from Earth. The possibility of this object being a S-IVB booster from a Saturn V needs to be investigated further before any definite plans for a manned mission to the object are formed. If however, it is confirmed that 2000 SG344 is in fact an NEO, its low ΔV requirements and 37 to 40 m diameter make it a prime target for a first crewed mission. Mission lengths for this asteroid can be highly variable based on the system architecture that is chosen for the mission. System architecture solutions will be offered for such missions later in the study. This asteroid does however offer the most system architecture flexibility since the ΔV requirements for this asteroid are lower than almost any other NEO.

B. 1999 AO10 Design Example

The second optimum NEO target design example focuses on a 180-day mission to 1999 AO10. Although the ΔV requirements for this asteroid are higher than those of 2000 SG344, 1999 AO10 is more sizeable and may be an alternative choice if it is shown that 2000 SG344 is in fact a Saturn V upper stage. The 180-day mission length is chosen to illustrate a long duration mission, in which the total ΔV is reduced due to the extended mission length. The 12 year launch window with corresponding ΔV requirements can be seen in Figure 5. The optimum launch window is better shown in Figure 6. An optimum launch date occurs at the end of August 2025, with an adequate launch window.

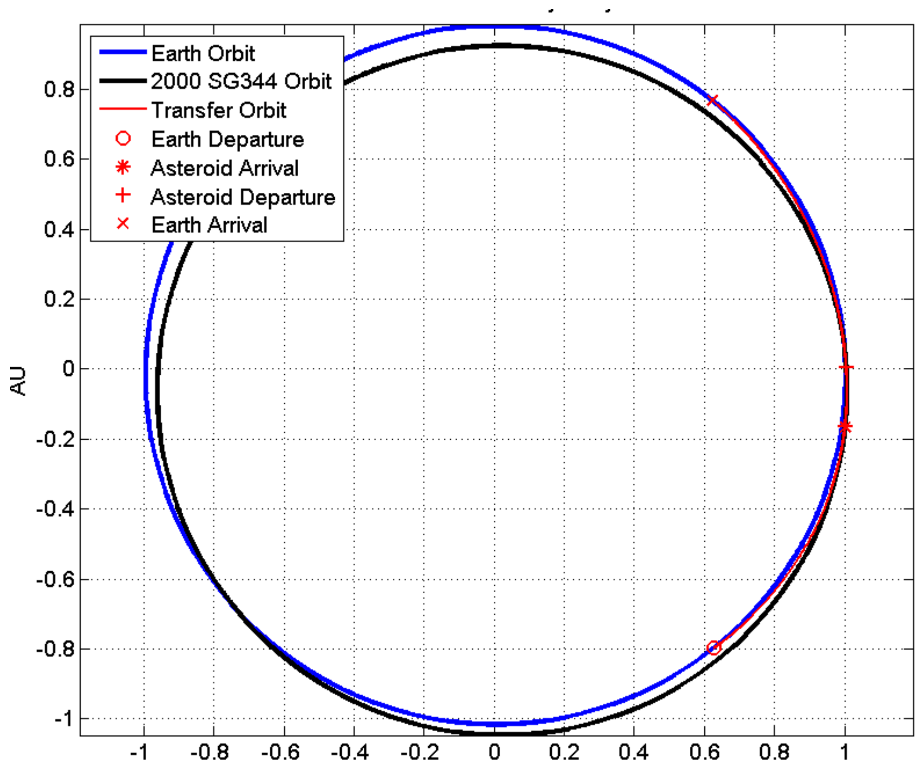

Figure 3. 105-Day 2000 SG344 Trajectory in the Heliocentric Reference Frame

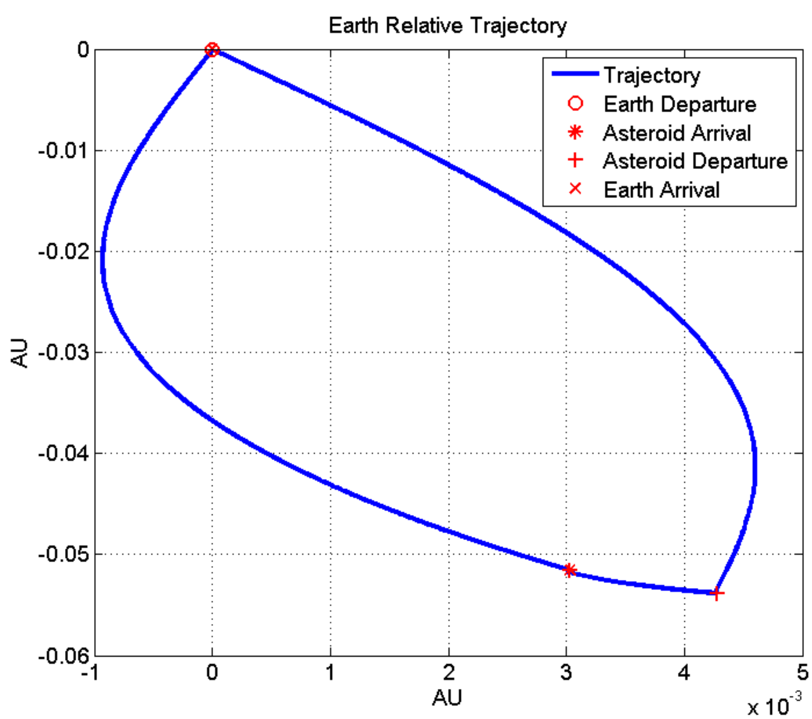

Figure 4. 105-Day 2000 SG344 Trajectory in the Earth-Fixed Reference Frame

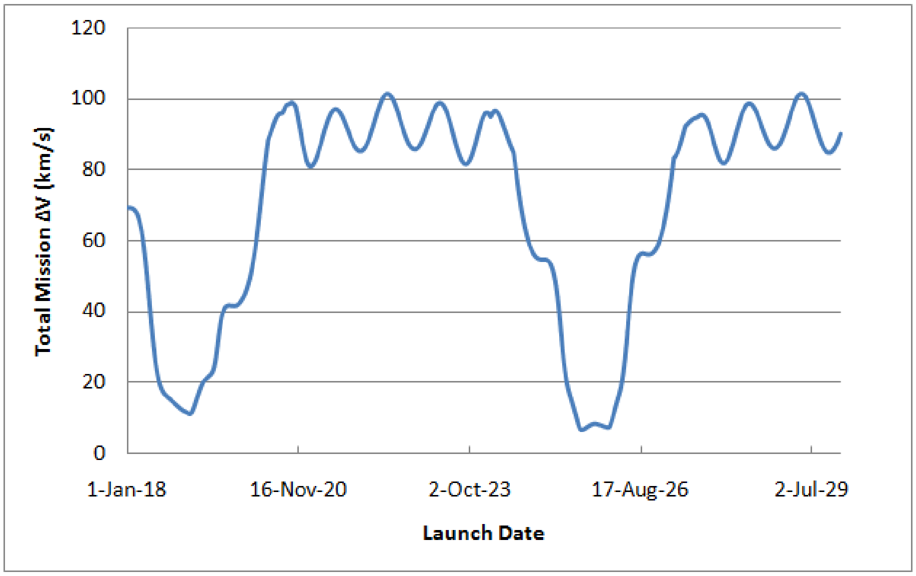

Figure 5. Mission ΔV vs. Launch Window for 180-Day Crewed Mission to 1999 AO10

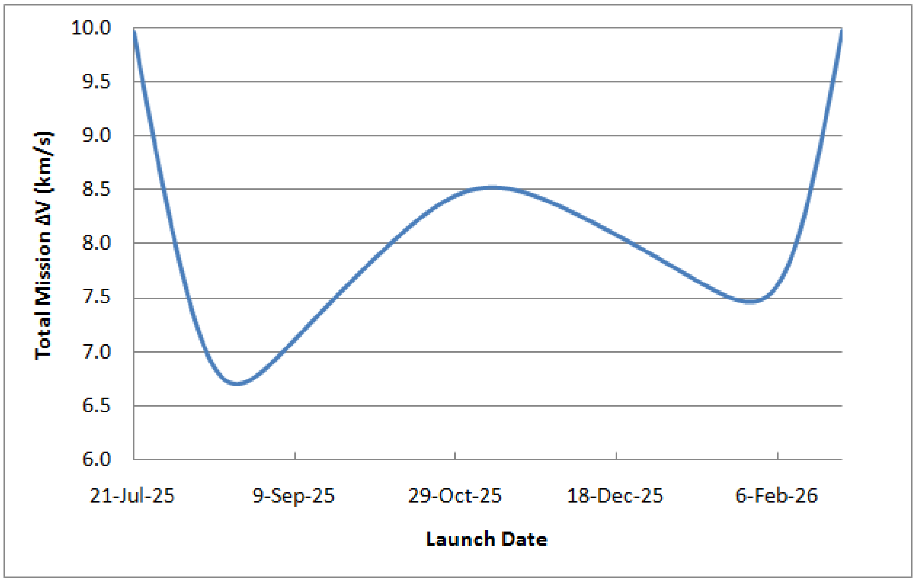

Figure 6. Optimum Launch Window for a 180-Day Crewed Mission to 1999 AO10

It is important to examine the four different maneuver burns to see if and how they can complicate the system architecture design. The optimum mission design for the 180-day mission is shown in Table 3. Again, the additional ΔV margin is made up of an extra 0.15 km/s margin for station-keeping and minor propellant boil-off, and a 10 percent increase in the Earth-departure burn.

In this design case, the baseline crew vehicle will be capable of performing the entire asteroid departure burn, which is beneficial. However, a long gap exists between the Earth-departure and asteroid arrival burns. The gap is just over four months, meaning that propellant boil-off would take a substantial toll on any stored, cryogenic fuel. If this were to be the first crewed mission, a system architecture using non-cryogenic fuel would need to be utilized, or an additional fuel margin would need to be added. If additional fuel is required, a more complex system architecture might need to be selected.

Table 3. 180-Day Mission to 1999 AO10

	Required ΔV (km/s)	Date
Earth Departure	3.3038	22-Aug-2025
Asteroid Arrival	2.2262	31-Dec-2025
Asteroid Departure	1.1735	1-Jan-2026
Earth Arrival	0.0000	18-Feb-2026
Additional Margin	0.4804	
Total ΔV Required	7.1839	

To again illustrate the transfer orbits of such a mission, a heliocentric trajectory is shown along with an Earth-fixed trajectory in Figures 7 and 8. These trajectories take the majority of the mission duration to reach the asteroid. Once asteroid departure occurs, the return trip to Earth only takes about one month.

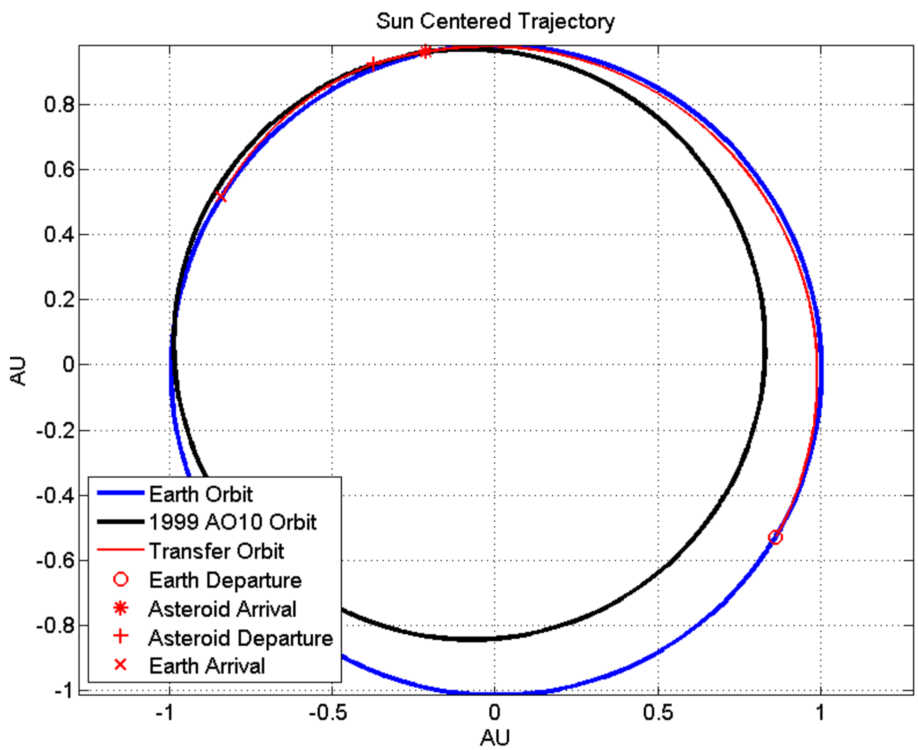

Figure 7. 180-Day 1999 AO10 Trajectory in the Heliocentric Reference Frame

In both design examples, the asteroid arrival burn is the ΔV required for the Orion CEV to match the speed of the asteroid. In addition, there is no Earth-arrival ΔV required. This indicates that the Orion CEV is traveling at a velocity of less than 11 km/s, which is the maximum allowable re-entry speed of the baseline crew capsule.

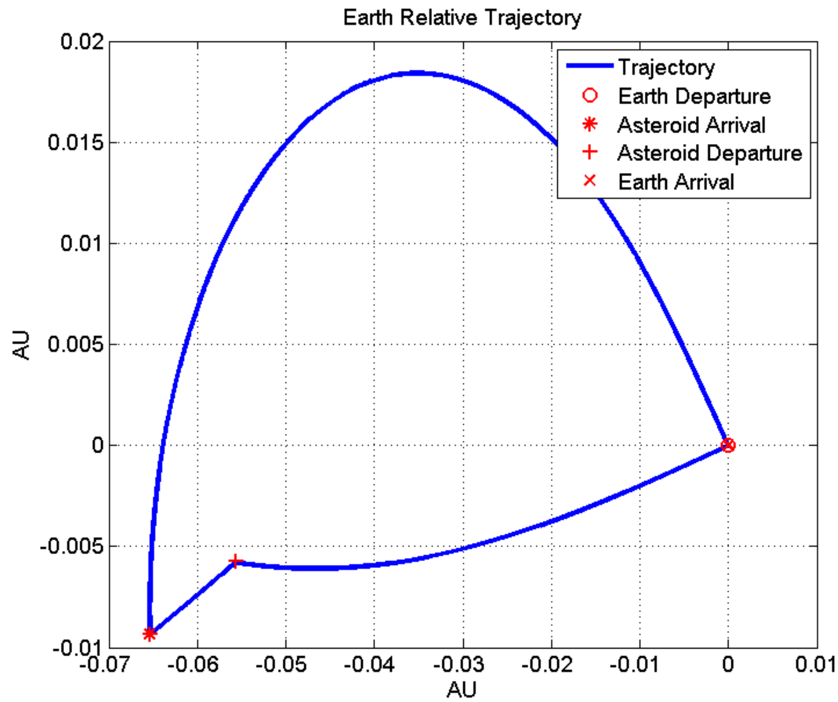

Figure 8. 180-Day 1999 AO10 Trajectory in the Earth-Fixed Reference Frame

C. Mission Length and Required ΔV

Mission designs are tested in step sizes of 15 days so that a better understanding of the relationship between mission length and required ΔV can be attained. Although the general trend remains that longer mission lengths produce lower ΔV requirements, it is important to see how much the ΔV is reduced with each 15 day interval. Tables 4 and 5 each show 15 day increment results for the 12 NEOs which have the lowest required ΔVs for the 180-day mission length. Table 4 includes each asteroid's diameter, showing the NEOs which are excluded for being too small.

Table 4. Optimum NEO Candidate List

Asteroid Designation	Size (m)	45 Days	60 Days	75 Days	90 Days	105 Days
2007 XB23	13	11.134	10.078	9.401	8.865	8.372
2006 RH120	5	9.449	7.549	6.545	5.932	5.523
2000 SG344	37	9.770	8.725	8.173	7.190	6.386
2008 EA9	10	11.367	9.685	8.780	8.159	7.657
2008 JL24	4	20.030	15.095	12.561	10.718	9.283
2007 UN12	6	15.003	12.359	11.030	10.187	9.555
1999 AO10	60	12.300	10.453	9.329	8.596	7.983
2001 QJ142	72	18.175	13.364	11.189	9.914	9.130
2009 OS5	70	15.163	11.910	10.176	9.030	8.236
2001 GP2	14	9.391	8.713	8.342	8.086	7.875
2003 LN6	43	11.982	10.725	9.933	9.334	8.821
2006 WB	?	11.308	10.126	9.380	8.801	8.326

The absolute magnitude is a measure of an object's brightness, and is used to estimate the size of most NEOs. Lower H magnitudes correspond to brighter objects, while higher H values represent more dull objects. Asteroids with lower H values are estimated to have larger diameters since they reflect more light and appear brighter. However, using the absolute magnitude to determine the sizes of NEOs only provides an estimate. It may be the case that these

Table 5. Optimum NEO Candidate List

Asteroid Designation	120 Days	135 Days	150 Days	165 Days	180 Days	200 Days
2007 XB23	7.851	8.281	6.527	5.452	4.525	5.619
2006 RH120	5.235	5.024	4.865	4.745	4.655	4.571
2000 SG344	5.755	5.329	5.015	4.811	4.695	4.488
2008 EA9	7.201	6.740	6.225	5.757	5.386	5.030
2008 JL24	8.197	7.410	6.866	6.521	6.169	5.741
2007 UN12	9.021	8.523	8.019	7.320	6.636	5.905
1999 AO10	7.475	7.139	6.924	6.789	6.704	6.702
2001 QJ142	8.646	8.114	7.492	7.069	6.813	6.682
2009 OS5	7.688	7.318	7.075	6.925	6.842	6.809
2001 GP2	7.682	7.496	7.305	7.109	6.913	6.672
2003 LN6	8.341	7.891	7.491	7.184	6.975	6.820
2006 WB	7.927	7.598	7.332	7.124	7.014	7.857

estimates can be incorrect or have some amount of error. If it is the case that a large error does exist in some of the potential size estimates, the list should be re-evaluated.

Figures 13 and 14 in the Appendix illustrate how mission length affects the ΔV requirements for each of the five optimum NEO candidates. One important factor to note in these mission length studies is that extending the mission length in favor of lower ΔV requirements may not always be the best option. In some cases, extending a 150-day mission to 180-days can only decrease the total ΔV by a small amount. This small decrease in the required ΔV may not be worth pushing the physical limits of the crew or the crewed vehicle. Once a sufficient amount of mission designs have been considered, the next step is to design an architecture that is capable of meeting the ΔV requirements of the mission.

III. Architecture Options for a Crewed Mission

A. Baseline Crewed Vehicle

The Lockheed Martin Orion CEV, a major component of the Constellation Program, was being designed to take astronauts to LEO and beyond. Although the Orion program might be permanently canceled, significant progress has been made on the vehicle design. The current design consists of a crewed capsule which is attached to a service module, similar to the Apollo design.

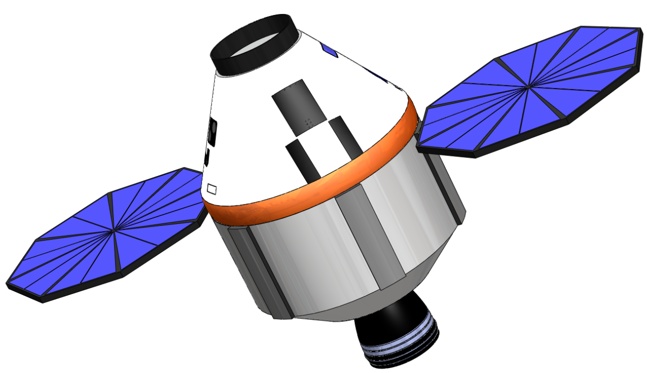

Figure 9. Orion CEV - Capsule and Service Module

The capsule housing the astronauts also serves as the re-entry vehicle when returning to Earth. Although initial capsule designs allowed for a crew of six astronauts, a plan to limit the crew capacity to four has been considered. For a crewed NEO mission, three astronaut configuration would better suit the mission requirements. The crewed capsule houses the astronauts, while the service module provides life support, propulsion, and power for the capsule. Solar arrays on the service module provide electrical power for the CEV. The service module of the Orion CEV is equipped with a Delta II upper-stage engine that uses MMH and N_2O_4 hypergolic propellant. The Orion capsule also plays an important role in mission design. Similar to a previous NASA study, the Orion capsule is limited to re-entry speeds of 12 km/s.[7] The ablative heat shield of the crewed capsule helps to limit the total mission ΔVs since high re-entry speeds can be tolerated. If the capsule could only withstand the re-entry speeds of the shuttle, maneuvers would need to be performed to slow the capsule down before returning to Earth. This would add an additional ΔV requirement that would present additional challenges to the architecture.

Even though the program has been canceled (as of today), the progress made on the design has produced the best projections and estimates of what will ultimately be necessary for any crewed mission outside of LEO. Any manned mission in the future will require a crewed capsule, with a service module providing power and propulsion capabilities. For these reasons, a modified Orion CEV is used as a baseline in all of the NEO system architectures in this paper. Prior to any design modifications, the proposed Orion CEV mass is estimated at 20,200 kg. From a recent NASA study, the consumables for a four person 21-day lunar mission can be replaced with the consumables necessary for a three person 180-day asteroid mission.[7] Consumables include food, water, and oxygen necessary for the crew to survive. This NASA study estimates that each crew member requires 8 kg of consumables each day. Baseline Orion CEV specifications are provided in Table 6.

Table 6. Orion CEV Characteristics and Performance Capabilities [7,8]

Endurance	180 Days
Crew Members	3
21-Day Consumables (Four Person)	672 kg
180-Day Consumables (Three Person)	4,320 kg
Mass to Orbit	23,848 kg
Diameter	5.03 m
Habitable Volume	8.95 m^3
Post-EOR ΔV	1.595 km/s
Main Engine Isp	323 s

B. Alternative Crewed Vehicle Technology

The Dragon capsule, which is currently being developed by SpaceX, will also transport humans into space. Its main purpose will be to carry astronauts and cargo to the ISS. The current design, much like the Orion CEV which includes a capsule and a service module, is approximately 6,000 kg.[9] This design however, would not be adequate for manned missions outside of LEO. Although the interior volume rivals that of the Orion capsule, Dragon's current design is not robust enough to make it suitable for alternative missions. Since it will mainly be used to transport crews to and from the ISS, it lacks the necessary radiation and heat shielding that would be required for manned-missions outside of LEO. Also, the 1,290 kg of propellant that the Dragon service module carries severely limits the ΔV capabilities of the system.

If interior volume is determined to be too limited in these crewed capsules for extended mission lengths, other approaches might need to be considered. Bigelow Aerospace, a company specializing in the development of inflatable space habitats, may provide some solutions. Two prototypes of this technology include the Genesis I and II missions, which are currently in LEO. They continue to work towards scaling up this technology, in hopes to make it more suitable for a crewed mission. These inflatable habitats expand to increase their volume while keeping their length constant. Solar arrays and air-locking technology is also included in the habitat design. Bigelow Aerospace also claims that the kevlar-mylar fabric of the habitat is a good radiation shield in addition to being more resistant to space debris than the ISS.[10] This technology could greatly increase the amount of habitable volume without drastically increasing the system mass.

C. Constellation-Type Heavy Lifter

The development of the Ares V-class heavy-lifter could be a valuable asset in any crewed mission. The Ares V is designed to deliver just under 188,000 kg to LEO, and would be the largest launch vehicle ever created. With a 10-m diameter fairing that spans 22-m in length, the payload options for this launch vehicle are quite versatile. The Ares V is composed of a core stage, two 5.5-segment solid rocket boosters, and an upper stage commonly referred to as the EDS (Earth Departure Stage). The core stage utilizes six upgraded RS-68B engines, which stem from the Delta IV engine, while the EDS uses one Saturn V-derived J-2X engine. Both the core stage and the EDS make use of LOX and LH_2 cryogenic fuels. The EDS holds nearly 252 MT of usable propellant, but some of this is required to place the EDS into orbit. The size of the Ares V payload determines how much fuel is left in the EDS once LEO is reached. With a small payload such as the Orion CEV, the EDS can have a substantial amount of fuel left to perform some of the required burns.

An Ares V-sized launch vehicle allows for many other types of launch and payload configurations as well. Launching the Orion CEV on an Atlas V-H or a Delta IV-H, while simply launching an empty Ares V would increase the remaining propellant in the EDS. An alternative to this configuration could include a 180 MT cryogenic OTV inside of the Ares V fairing, which would maximize the available propellant remaining in LEO. However, a cryogenic OTV of this size does not currently exist. Development of this OTV would need to occur, or an EDS would need to be scaled down for this option to be realistic. Since the size, propellant mass, dimensions, and dry mass of an OTV of this size would all be estimates, no proposed architectures including OTVs are provided in this study. Other architectures could include multiple Ares V launches. Two to three Ares V launches can significantly increase ΔV capabilities, but such an architecture increases the complexity of the design. These system architectures will be discussed in more detail in later sections.

Even if the Constellation Program becomes permanently canceled, the NEO crewed mission configuration based on an Ares V sized launch vehicle shows the benefits of a large payload capability. Using some of the technology that has been developed under the Constellation Program, plans to re-develop some type of super heavy-lifter with similar performance values could resurface in the future. The proposed Ares V acts as the baseline launch vehicle for architecture designs in this paper. Performance estimates of this launch vehicle are more importantly used in this study to represent the capabilities of any launch vehicle of this size. It is not imperative that this exact Ares V configuration is used for a crewed NEO mission.

D. Direct Proposal

An alternative launch vehicle system that has been heavily discussed in the past few years is the DIRECT Proposal. The DIRECT suggests that the Jupiter family of launch vehicles would achieve all of the performance goals laid out by the Constellation Program. Jupiter launch vehicles would be shuttle derived heavy-lifters, that would require a smaller time-frame and budget to develop than the Ares rockets. It is estimated that if implemented, this design approach could be ready for manned flight by 2012 or 2013. If the current space shuttle program could be extended to 2012, this could eliminate an operational gap that would exist with the Constellation Program. The Jupiter family of launch vehicles are all designated by three numbers. The first number designated the number of rocket stages, the second number indicates how many SSMEs make up the core stage, and the last number identifies how many engines are used in the second-stage. For example, the Jupiter-130 launch vehicle consists of only a first-stage, and three SSMEs make up the core. The Jupiter launch vehicles would largely use existing, proven technology in their designs. They would all share a common first-stage core, which would keep development costs low and make the launch vehicle system more modular. The common first-stage includes an extended version of the space shuttle's external tank, which holds the LOX and LH_2. The first-stage also includes two, four-segment reusable solid rocket motors (RSRM); these would be the same RSRM that are currently employed on the space shuttle.

The Jupiter family of launch vehicles could be used for carrying the Orion CEV or an EDS to LEO. The Jupiter-130 is a proposed crewed launch vehicle that would carry the Orion CEV. No upper-stage is used in this configuration, but it is still capable of delivering 60,000 kg to LEO. If this were used to carry the Orion CEV only, a large portion of the payload capacity of the launch vehicle would be wasted. This additional space could be available for an extended service module or a Centaur upper-stage, however.

A Jupiter-246 design could be used to achieve high ΔV capabilities from a system architecture. The Jupiter-246 design could be used to place a sizable EDS in LEO. Again, the same common Jupiter core and two, shuttle RSRM would be attached on the sides. The EDS performs a portion of the ascent burn, and the remaining fuel in the EDS would be utilized for a departure burn from LEO once the Orion CEV has docked. Once delivered into LEO, the Jupiter-246 EDS would have just under 100 MT of usable propellant leftover.

There are some major drawbacks to the Jupiter family of launch vehicles, however. Every one of the Jupiter launch vehicles uses the same external tank core. Although these designs would make use of these common parts, they would still need to be redesigned. The Jupiter external tanks would be larger than the ones currently used for the space shuttle. These tanks would need to be scaled up and tested before they could be implemented. There has also been much dispute inside NASA over the performance projections of the DIRECT. In examining the Jupiter-246 EDS launch vehicle configuration, the upper stage raises some questions. The burnout mass of the Jupiter EDS has a fairly low mass relative to its wet mass.

Table 7. Upper Stage Characteristics and Comparison [7,11,12]

Upper Stage	I_{sp} (s)	BOM (kg)	Propellant Mass (kg)	Ratio
Ares V EDS	448	26,600	251,900	0.0955
Centaur	450	2,100	20,830	0.0916
Jupiter-246 EDS	445	12,008	170,362	0.0654

The ratio column in Table 7 refers to the ratio of the upper stage's BOM to its total, wet mass. For the case of the Jupiter-246 EDS, the ratio is significantly smaller than the other two upper stage ratios. This ratio may be feasible, however it seems like that ratio might have been underestimated. Therefore, it is difficult to know whether the predicted ΔV from the proposed configurations would be accurate. Since the DIRECT project is only a proposal at this stage, there is currently no development of these launch vehicles. For this reason, no system architectures referencing the DIRECT Proposal are considered in this paper.

E. Current Heavy Lift Vehicles (Delta IV-H, Atlas V-H, Falcon 9-H)

If a system architecture that uses current heavy launch vehicle technology is desired, a few different launch options exist. The current Delta IV-H launch vehicle is capable of lifting 23,000 kg into LEO.[13] Although an Atlas V-H launch vehicle has been proposed, it is not currently in production. It is estimated that this launch vehicle would have performance that is comparable to the Delta IV-H. The Falcon 9-H is another launch vehicle that is proposed for development, and it is projected to lift up to 32,000 kg into LEO.[14] Since the Delta IV-H is currently the only existing heavy launch vehicle that could potentially carry the Orion CEV, it will be used in some system architectures. Some alterations to the Delta IV-H would need to be made however, if the launch vehicle were to carry the Orion CEV to LEO.

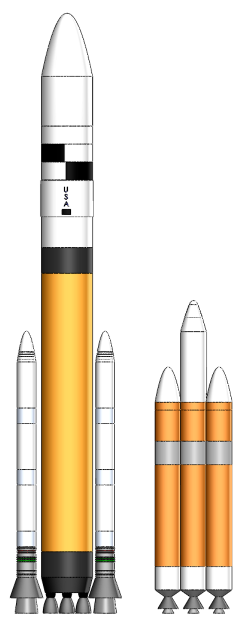

Figure 10. Ares V Launch Vehicle on the left - Delta IV-H Launch Vehicle on the right

The Orion CEV weighs about 850 kg more than the Delta IV-H can deliver to LEO, meaning the performance of the launch vehicle would need to be slightly increased or the weight of the Orion must be decreased slightly. Increasing the performance of the Delta IV-H may be possible, since the delivery orbit for this launch vehicle is a 400 km, circular LEO orbit. The Orion CEV only needs to be taken up 240 km to depart from a 185 km, circular orbit. Another issue that arises can be seen in the fairing design. The largest Delta IV-H fairing currently has an inside diameter of just over 4.7 m. A modified fairing with an inside diameter of over 5.03 m would be needed to house the crewed vehicle.

Even with the modifications made, a single launch is only capable of lifting the Orion CEV into LEO. A second launch would then be necessary to transport some type of upper stage to LEO so that an Earth-departure burn could be performed. A Centaur-sized upper stage could fit inside the fairing of one of these launch vehicles, however payload capacity and fairing volume limit the amount of ΔV increase that each launch can provide. Architectures involving a two or three launch system would be best suited for longer missions of around 180 days. Missions that are longer than 180 days would also be feasible when considering these architectures, but the crewed vehicle endurance and radiation exposure become more of a concern.

It is important to note that all of the proposed system architectures assume that the launch vehicles are human-rated. Before any crewed mission can take place, the process of human-rating any launch vehicle must take place. One main addition to man-rate a launch vehicle would be the addition of a launch abort system that could separate the crewed capsule from the launch vehicle if failure occurred during launch.

IV. System Architecture Design for Crewed NEO Missions

Since ΔV requirements can vary widely depending on the target NEO and the selected mission length, it is important to provide a number of different system architectures that can make such missions possible. The system architectures that are offered in this study combine the Orion CEV as a baseline crew vehicle with the launch vehicles that have been previously discussed. The first step in this process is to establish the ΔV capabilities of all proposed system architectures.

A. Calculating ΔV Capabilities

To calculate the capabilities of system architecture, we use Tsiolkovsky's rocket equation.[15]

$$\Delta V = g_0 I_{sp} \cdot \ln(m_0/m_1) \tag{1}$$

where g_0 is equal to 0.00980665 km/s. The initial mass of a system architecture includes the fully loaded mass of the upper stage(s) and the mass of the Orion CEV. The final mass is the BOM of the upper stage combined with the mass of the Orion CEV. The ΔV capabilities can be maximized with high I_{sp} values and large mass fractions, meaning the initial mass is much larger than the final mass. To illustrate the capabilities of some feasible system architectures, Table 8 is provided.

Table 8. System Architecture Comparison with ΔV Capabilities

System Architecture	Upper Stages	ΔV Capability (km/s)
2 Delta IV-H + Orion CEV	Centaur	4.201
3 Delta IV-H + Orion CEV	2 Centaur	5.767
1 Ares V + Orion CEV	EDS	6.901
1 Ares V + 1 Delta IV-H + Orion CEV	EDS	7.482
2 Ares V + Orion CEV	2 EDS	9.303
2 Ares V + 1 Delta IV-H + Orion CEV	2 EDS	9.678
3 Ares V + Orion CEV	3 EDS	10.611

From Table 8, it can also be seen that the Delta IV-H class of launch vehicles are inadequate for almost any NEO mission under 200 days. It is possible that a 180-day mission to 2000 SG344 could utilize this architecture since the total ΔV is around 4.695 km/s. However, this ΔV does not include the additional margin for the Earth-departure burn and station-keeping. With these additions, the total ΔV required is adequate, and a 150-day mission to 2000 SG344 is actually the upper limit of the architecture's capabilities. Therefore, a strictly Delta IV-H class architecture may only be capable of missions that have a duration of longer than 180 days (other than the 2000 SG344 case). Table 8 also illustrates the trade-off between system complexity and ΔV capabilities. As the number of launches increases, and the upper stage configuration grows, so does the available ΔV from the system. For a first crewed mission, it may be important to minimize the architecture complexity. As previously stated, an Ares V launch which carries the Orion CEV is the baseline system architecture. Figure 11 illustrates what this baseline configuration looks like in LEO.

Figure 11. Ares V EDS Configured with the Orion CEV

This baseline configuration, in addition to all other architectures utilizing the Ares V EDS, depends on the usable propellant remaining after LEO is reached. By using the TLI mass capabilities of the Ares V and the required ΔV to perform the injection, estimates of the remaining LEO propellant can be made.[12] An Ares V carrying the unmodified version of the Orion CEV has an estimated 120 MT of propellant left for the crewed mission. The difference between the baseline Orion CEV and the unmodified version is then subtracted from the 120 MT to achieve a more refined mass for the usable propellant in LEO. For any architecture which launches an Ares V EDS without a payload, the mass of the baseline Orion CEV is added to the available propellant mass of the EDS in LEO.

Alternative system architectures can exist when different Orion CEV specifications are used in the ΔV calculations. Depending on the target asteroid and mission length that are designated for a mission, the Orion CEV mass will change to account for a reduced crew or the variable consumables budget. By adding three additional Orion CEV configurations, more viable alternative architectures are also provided. These three additional Orion CEV configurations and their updated system architecture comparisons can be seen in the Appendix.

In the case where more than one EDS is used, the EDS used to carry the Orion CEV is consumed last since it is already attached to the Orion CEV. Therefore, the EDS with the most propellant is always expended first. For all system architectures where multiple upper stages are used, a truss assembly is needed to connect the system together. After the first upper stage, the connecting system will most likely be separated along with the spent upper stage. Mass estimates for such a truss system were not included in the ΔV calculations for these architectures. Since the truss mass would most likely be quite small compared to the mass of the upper stage, it should not diminish the estimated performance significantly. Therefore, the ΔV capabilities of the multiple upper stage configurations are only theoretical maximums.

The overall mission architecture is illustrated in Figure 12. Each stage of the mission design is shown, in addition to the state of the system architecture at stage.

B. Architecture for 2000 SG344 and 1999 AO10 Design Cases

Recalling the previous design example results, a 105-day 2000 SG344 and a 180-day 1999 AO10 had ΔV requirements of 6.876 and 7.184 km/s, respectively. The baseline architecture of a single, Ares V launch provides just over the ΔV amount necessary for the 2000 SG344 105-day mission. This design example is partly chosen to show the upper limits of the baseline architecture design. Shorter missions to this NEO would not be possible with the baseline architecture. Since this architecture allows only for a slim ΔV margin for the 105-day 2000 SG344 mission, the actual crewed

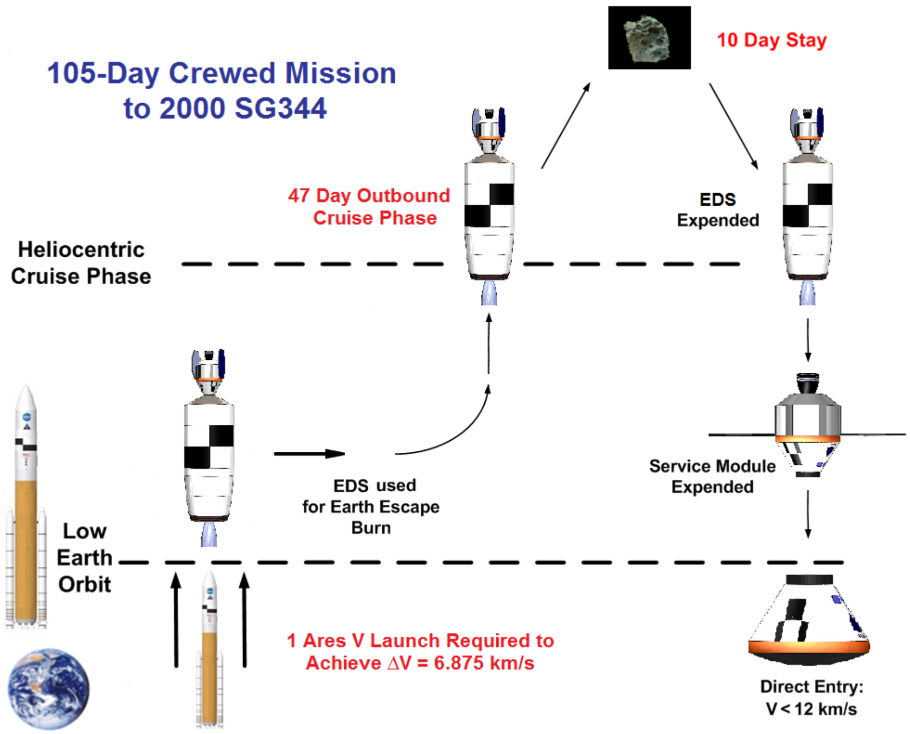

Figure 12. Typical NEO Mission Summary and Architecture

mission would most likely either need to be extended to 120 days, or a Delta IV-H class vehicle would be needed to launch the Orion CEV. This would provide a more adequate safety margin for the ΔV requirements of the mission.

For the 180-day 1999 AO10 mission, the addition of a Delta IV-H class of launch vehicle to the baseline architecture would be adequate. With the Delta IV-H carrying the Orion CEV, this architecture provides adequate performance for the mission. More complex architectures are required if shorter mission lengths for this asteroid are chosen. Multiple Ares V launches can however meet the requirements of these mission lengths.

C. Previous Target NEO Studies

A recent NEO study by the European Space Agency (ESA) has also put together a list of target NEOs to visit with a first crewed mission. The study makes a few base assumptions as well to narrow down the list of target NEOs. The crewed mission would take place after 2020 and before 2025, with the mission length limited to just under 550 days. The study also limited the search to only include asteroids with apparent magnitudes less than or equal to 22. This corresponds to asteroids with an approximate diameter of at least 200 m. By doing this, the list of tested NEOs becomes significantly limited.

The results of this test established leading candidates which have proposed mission lengths of around 400 days and ΔV requirements between 6.5 and 7.1 km/s.[16] For a first crewed asteroid mission, durations of this length may not be practical, especially if cryogenic propellant is used because of its high I_{sp}. Storing cryogenic fuel for these types of long duration missions would not be feasible. Bi-propellant hydrazine and oxidizer would need to be used in the upper stages, or all large burns would need to be completed within the first two months of the expedition. Missions requiring bi-propellant hydrazine and oxidizer would most likely require more upper stages than a cryogenic mission. Also, astronauts would be spending significant amounts of time in deep space, which means the concern of radiation exposure becomes more of an issue. Additional radiation shielding would need to be added to the architecture to ensure the safety of the astronauts during such long missions.

Limiting the size of the tested NEOs to greater than 200 m may narrow the target field too much. As previously shown, target asteroids can be reached in a shorter amount of time, with lower ΔV requirements. Although these asteroids have smaller projected diameters, they are still fairly sizable with diameters of around 70 m. If the absolute magnitude of these smaller asteroids is determined to be of concern for guidance and navigation reasons, the list can be re-evaluated.

Table 9. Preliminary ESA Study Results[16]

Asteroid Designation	Nominal ΔV (km/s)	Time of Flight (Days)
2000 EA14	6.590	405
1991 JW	6.937	414
2002 TD60	8.776	365
998 HG49	7.141	405
2001 QC34	7.428	379

D. Alternative Propulsion Technology

One type of innovative propulsion technology that combines the high specific impulse of ion engines with the high thrust capabilities of chemical rockets is known as the Variable Specific Impulse Magnetoplasma Rocket (VASIMR). Under current development since the late 1970's, the VASIMR engine involves the creation and acceleration of plasma through a magnetic nozzle to produce thrust. The engine requires the use of a propellant such as argon or hydrogen. The propellant is fed into the engine where a helicon antenna uses radio waves to convert gas into plasma by knocking an electron free from each gas atom. After the first helicon antenna stage, the temperature of the newly formed plasma is fairly low. A magnetic field is produced by superconducting magnets to move and funnel the plasma to the second radio frequency booster antenna. This section of the engine is referred to as the Ion Cyclotron Heating section. Here, radio waves interact with the plasma at its resonance to further accelerate and heat the plasma. A magnetic nozzle expands the accelerated plasma out of the engine, which converts the plasma's energy into linear momentum.[17]

For manned missions, VASIMR could significantly reduce mission lengths. It has been proposed that a spacecraft using multiple VASIMR engines could reach Mars in as little as 39 days.[18] Longer missions of around 4 months would also be achievable with lower power requirements. Since the use of the VASIMR would lower travel times and allow for higher ΔV requirements, the list of target NEOs would grow significantly. This would greatly increase the mission flexibility, and launch windows would be less constrained. The VASIMR engine could also be used for space tug missions to the asteroid for the purpose of carrying cargo, a lander, or NEO deflection payloads. This would allow the Orion CEV to be transported to the asteroid using more conventional propulsion technology such as chemical rockets. Also, the strong magnetic field produced from the engine could provide radiation shielding to a manned spacecraft. This type of ion-plasma driven engine also has some advantages over other engines in this family of propulsion.

As the name of the technology implies, this engine can vary its specific impulse depending on the input power to the engine. This can make the engine applicable to many different types of mission scenarios. Also, the VASIMR runs without the use of cathode and anode electrodes to ionize and accelerate the propellant gas. Elimination of the electrodes increases the lifetime of the engine, since the electrodes corrode during operation. In addition, since the plasma is completely contained by superconducting magnets, the deterioration of engine components is drastically reduced. Significant progress has been made on the most recent VASIMR engine design. The VX-200, unlike the previous VASIMR designs, relies more on space-rated technology and operates at 200 kW of input power. The power is distributed between the two RF antennas and the superconducting magnets, with the majority of the power being used to accelerate and contain the plasma. The design has successfully operated at full rated power. Power inefficiencies in the RF antennas have also been nearly eliminated. The VX-200 operates at an I_{sp} of 5000 sec, and can produce thrust levels of between 1 and 5 Newtons. A VASIMR engine can produce I_{sp} values much greater than 5000 sec (10,000-30,000 sec), but it requires the first RF antenna to be powered at higher levels so that a greater percentage of the gas is ionized.[17]

Current plans for the VASIMR include a flight-test at the International Space Station (ISS) to test the engine's performance in natural operating conditions. The tests are expected to occur sometime within the next five years. The VF-200, a two thruster engine bus, will be powered with the help of the ISS and weigh just over 4,500 kg. The testing will attempt to validate the VASIMR's ability to provide high I_{sp} performance for the ISS altitude control system. If these tests are successful, the VASIMR technology will undoubtedly be expanded for other uses. By scaling up the VASIMR's input power, higher I_{sp} or thrust can be achieved. The large drawback of the VASIMR designs is the power requirements, however. Although VASIMR can produce very high I_{sp} values with power in the range of hundreds of kilowatts, higher thrust operations require powers in the order of megawatts. Anywhere between 12 and 200 MW_e would be required for these short-length Mars trips.[18] This amount of power would be very difficult to produce in space. Solar power would be simply incapable of handling these types of power requirements. Current solar array technology can have energy densities around 200 W/kg.[15] The only feasible way to produce this amount of electrical power would be through the use of a nuclear option. Current fission reactors are capable of producing this magnitude of electrical power. However, a reactor design would have to be space-rated. Also, since the VASIMR is using the electrical power from the reactor and not its thermal energy, the reactor would have to be larger than a reactor used for nuclear thermal rocket engines.[19] This is because not all of the thermal energy from a nuclear reactor is converted into usable electrical energy. A fission reactor capable of producing electrical power in the megawatt range would currently be too large to be transported into space.

V. Conclusion

Determining a list of target asteroids for the first manned mission includes more than simply limiting the total mission ΔV. Mission length plays an important role in this decision. For the first mission, over-ambitious mission lengths should be avoided due to the long period of time since humans have voyaged outside of LEO. Any mission lengths that are greater than 180 days may be more realistic once the first crewed NEO mission is achieved. Sample design cases have shown that a crewed NEO mission is feasible based on mission design. Depending on the continued development of a crewed vehicle similar to the Orion CEV, it is reasonable to believe that a crewed NEO mission before 2025 is possible. This would of course require some launch vehicle progress as well, whether it is the modification of an existing heavy launch vehicle or the development of a new Ares V-sized vehicle. This study has also shown that the inclusion of an Ares V-sized launch vehicle can simplify a crewed mission. Developing a launch vehicle of this size could prove to be invaluable to the future of human space explorations. However, regardless of the system architecture that is chosen for the first asteroid mission, all of the launch vehicles would need to be man-rated before the mission would actually be possible.

Since the crewed missions considered in this paper have launch windows that are still 10 to 20 years off, it is important that any crewed mission study continually tests new NEOs. As the number of known NEOs continues to grow, it is possible that more ideal candidates could surface in the future. Target selection for the first crewed NEO mission should be an on-going study, especially as technology and NEO knowledge continues to expand.

Appendix

As can be seen in Figure 13, increasing the mission length past 150 days does not lower the required ΔV substantially for these two NEOs. The main benefit of increasing the mission length is most effective when the total mission length is less than 90 days.

For these three NEOs, this plot again indicates that extending the mission length past 150 days provides little benefit in regards to mission design. However, it is important to note that since these three NEO candidates have higher overall ΔV requirements, any extended mission may be preferred so that complex system architectures can be avoided. This plot also shows that mission lengths of 90 days or less would require complex system architectures such as multiple Ares V launches.

Table 10. A system architecture comparison based on a 2 person, 180-day mission. The Orion CEV mass for this mission designation is 22,408 kg

System Architecture	Upper Stages	ΔV Capability (km/s)
2 Delta IV-H + Orion CEV	Centaur	4.315
3 Delta IV-H + Orion CEV	2 Centaur	5.921
1 Ares V + Orion CEV	EDS	7.028
1 Ares V + 1 Delta IV-H + Orion CEV	EDS	7.576
2 Ares V + Orion CEV	2 EDS	9.430
2 Ares V + 1 Delta IV-H + Orion CEV	2 EDS	9.783
3 Ares V + Orion CEV	3 EDS	10.751

Table 11. A system architecture comparison based on a 2 person, 90-day mission. The Orion CEV mass for this mission designation is 20,968 kg

System Architecture	Upper Stages	ΔV Capability (km/s)
2 Delta IV-H + Orion CEV	Centaur	4.439
3 Delta IV-H + Orion CEV	2 Centaur	6.087
1 Ares V + Orion CEV	EDS	7.159
1 Ares V + 1 Delta IV-H + Orion CEV	EDS	7.674
2 Ares V + Orion CEV	2 EDS	9.561
2 Ares V + 1 Delta IV-H + Orion CEV	2 EDS	9.893
3 Ares V + Orion CEV	3 EDS	10.895

Table 12. A system architecture comparison based on a 3 person, 90-day mission. The Orion CEV mass for this mission designation is 21,688 kg

System Architecture	Upper Stages	ΔV Capability (km/s)
2 Delta IV-H + Orion CEV	Centaur	4.376
3 Delta IV-H + Orion CEV	2 Centaur	6.003
1 Ares V + Orion CEV	EDS	7.093
1 Ares V + 1 Delta IV-H + Orion CEV	EDS	7.625
2 Ares V + Orion CEV	2 EDS	9.495
2 Ares V + 1 Delta IV-H + Orion CEV	2 EDS	9.838
3 Ares V + Orion CEV	3 EDS	10.823

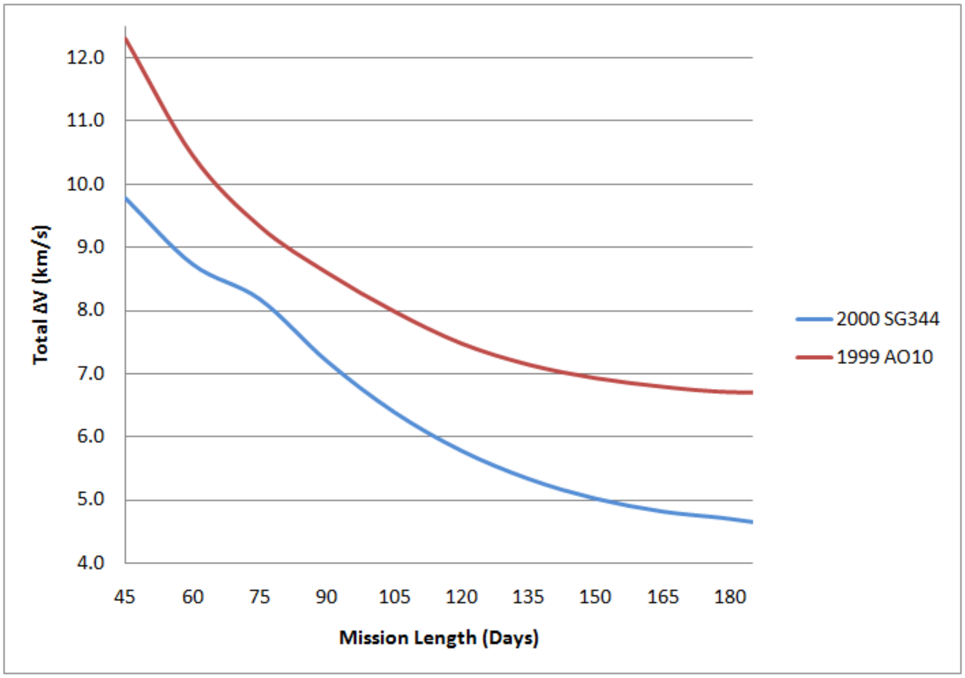

Figure 13. Total ΔV as a Function of Mission Length for 2000 SG344 and 1999 AO10

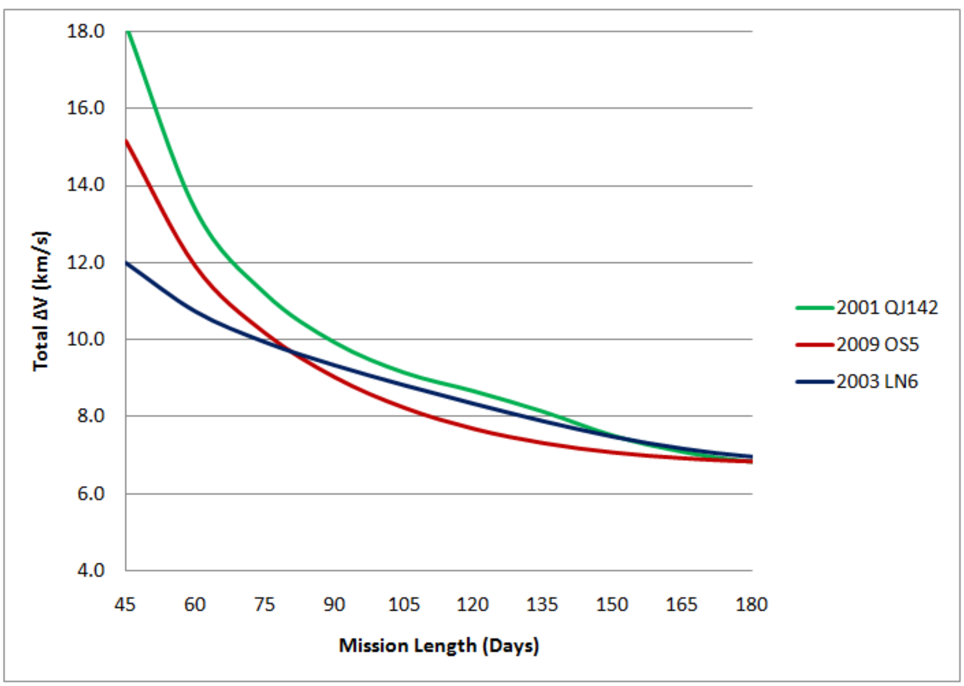

Figure 14. Total ΔV as a Function of Mission Length for Three Optimum NEO Candidates

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