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A Crewed 180-Day Mission to Asteroid Apophis in 2028-2029

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ABSTRACT

In recent years NASA has conducted studies examining the feasibility of sending a manned mission to a near-Earth object using technology developed under the Constellation Program. While these studies have identified several possible target asteroids, there are still many asteroids out of reach given the current Constellation architecture. The purpose of this study is to examine the feasibility of sending a piloted Orion mission to an NEO for the purpose of asteroid deflection or human exploration of asteroids. It is likely that a mission to an asteroid, which has been discovered to be on a collision course with the Earth, would not lie within the strict limitation of current NEO mission studies. The maximum ΔV using the current Constellation architecture is under 7 km/s. For this reason the asteroid 99942 Apophis, well outside the limitations of recent studies, will be used for a reference mission. A manned asteroid deflection mission to Apophis lasting 180 days and requiring a total ΔV of 12+ km/s can be executed using technology developed for NASA's Constellation Program, with minimal hardware developement necessary. The accompanying mission design and required changes to the Constellation architecture necessary to obtain a 12+ km/s ΔV will be discussed as well.

NOMENCLATURE

a	Semi-major Axis (AU or km)
ADRC	Asteroid Deflection Research Center
AU	Astronomical Unit
C_3	Earth Escape Energy km^2/s^2
CEV	Crew Exploration Vehicle
e	Eccentricity
H	Absolute Magnitude
i	Inclination (deg)
I_{sp}	Specific Impulse (s)
ISGC	Iowa Space Grant Consortium
ISS	International Space Station
JPL	Jet Propulsion Laboratory
LEO	Low Earth Orbit
NEO	Near-Earth Object
m	Mass (kg)
mT	Metric ton
NASA	National Aeronautics and Space Administration
OTV	Orbital Transfer Vehicle
PHO	Potentially Hazardous Object
TCM	Trajectory Correction Maneuver
TLI	Trans-Lunar Injection
ΔV	Change in Velocity (km/s)
θ	True Anomaly (deg)
Ω	Right Ascension of Ascending Node (deg)
ω	Argument of Periapsis (deg)

INTRODUCTION

The Asteroid Deflection Research Center (ADRC) has conducted a study to design a manned mission to an asteroid, for the purpose of asteroid deflection or human exploration of asteroids, using technology and launch vehicles currently in development for NASA's Constellation program. This study outlines how the Ares I and V along with the Orion crewed vehicle can be used to complete a manned mission to asteroid 99942 Apophis. Minimal changes to constellation architecture may be needed to complete a manned NEO mission, including the replacement of the Altair lunar lander with an orbital transfer vehicle (OTV).

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NASA's Constellation Program, originally developed through the Vision for Space Exploration lays out a plan on how NASA plans to fulfill the goals of the Vision for Space Exploration and the 2005 NASA Authorization Act. Through the Constellation Program NASA plans to develop a replacement for the shuttle, as well as the capability to deliver up to 4-6 crew to the International Space Station (ISS) as well as the lunar surface for an extended stay. Under the Constellation program two separate launch vehicles, the Ares I and V, as well as the Orion crew vehicle and the Altair lunar lander are under development. Using these new launch vehicles NASA plans to complete a manned lunar mission through a dual Ares I and Ares V launch. Through the use of much of this technology currently under development it will be possible to send a crewed mission to an NEO (several studies have confirmed this) with only minor modifications. The purpose of this study is to develop mission architecture capable of completing a manned NEO mission with only minimal changes needed to the current Constellation lunar mission architecture.

The idea of a crewed mission to an NEO has been proposed in several studies throughout the past few decades [1-8]. In 1966 a study performed at NASA proposed using the Apollo and Saturn V hardware to send a manned mission to the asteroid 433 Eros. The proposed mission would have taken place in 1975 when 433 Eros passed within 0.15AU of the Earth. This study also detailed the hardware changes necessary for a 500+ day mission [1]. Later manned NEO studies performed by O'Leary outlined the necessary mission requirements to mine several near-Earth and main-belt asteroids [2]. The proposed missions required 1-3 year total mission times, well beyond the mission length of any recent manned NEO study. Several other studies were performed in the late 1980's as part of the Space Exploration Initiative, however interest in a manned NEO mission declined drastically until recent years.

Interest in manned NEO missions has again begun to increase as NASA's Constellation Program matures. Over the past few years several studies have been conducted to determine the feasibilty of using Constellation hardware for a manned NEO mission for a 90-180 day spaceflight with a 7-14 day asteroid stay time [6-8]. The current studies have limited target asteroids to those that have low eccentrities (<0.5) and inclinations (<3.0 deg), have a close Earth approach, slow rotation(rotational periods of 10 hours or longer), and are single solitary objects. These strict limitations result in a mission requiring minimal ΔV and short stay times, generally less than 120 days.

An asteroid on a collision course with the Earth may not fit into this very narrow category of NEOs currently being studied for a manned NEO mission. For this reason the ADRC has performed a study to determine the feasibility and necessary changes needed to Constellatin hardware to complete such a mission. The asteroid 99942 Apophis is one such asteroid and will be used as the reference asteroid for this study. A manned mission to Apophis, which has a relatively low eccentricity (0.1912) and inclination (3.3314 deg), requires a ΔV of approximately 12 km/s for a 180 day mission, increasing as the mission length decreases. This is well beyond the reach of the most proposed manned NEO missions, which require a total ΔV of less than 7 km/s [6-8]. It is because of the strict limitations from the current manned NEO studies that the ADRC has decided to examine the feasibility such a mission and determine changes necessary to complete a manned mission to Apophis.

According to NASA/JPL's estimates Apophis has a 270 meter diameter and could generate over 500 megatons of energy if an impact with the Earth were to occur. This impact would be over 100 times larger than the Tungusta impact in 1908. A list of all the orbital parameters and the known physical characteristics of Apophis are given in Table 1 [9-11].

Table 1: Characteristics of Apophis

Characteristic	**Value**	**Unit**
Epoch	6/18/2009	
a	0.9224	AU
e	0.1912	
i	3.3314	deg
Ω	204.4425	deg
ω	126.4042	deg
θ_0	134.7126	deg
Orbit Period	323.5969	d
Rotational Period	30.5	h
Diameter	270	m
Mass	2.70e10	kg
Escape Velocity	0.1389	m/s
Albedo	0.33	
Absolute Magnitude H	19.7	

On April 13th, 2029 Apophis will pass somewhere between 5.62 and 6.2 Earth radii (35845-40182 km), well within the orbit of geostationary orbit. If Apophis were to pass through a 600 meter keyhole, well within the orbit uncertainty, it will enter a resonant orbit and impact Earth on April 13th, 2036. NASA has predicted a 1 in 45456 chance that Apophis will pass through the keyhole in 2029 [9-11]. While the chance of an impact is small, we don't know for certain if Apophis will pass through the keyhole until after the 2029 encounter. Preparations should be made for the small chance that Apophis does go through the 2029 keyhole. For this reason the necessary hardware and mission plan should be developed for a manned mission to Apophis prior to the 2029 encounter.

> "The problem is acute enough for Apophis that, IF impact hasn't been previously excluded, AND there hasn't been a through physical characterization, it can't be known for certain it will impact until during or after the 2029 encounter, even if a spacecraft is accompanying Apophis and providing position measurements good to 2 meters. That is, the keyhole could be determined only retrospectively, after passage through it."

MISSION ANALYSIS

To determine the feasibility of a manned mission to Apophis the mission requirements must first be determined. In particular the minimum ΔV necessary to complete the mission and the accompanying launch dates must be found. A computer program has been developed at the ADRC, which combines a Lambert's solver with ephemeris data to search for launch opportunities, find the required ΔV for each maneuver, and determine interplanetary trajectories. The program performs a search to find the minimum ΔV for each launch date by checking possible Apophis arrival, and departure date combinations given only the desired mission length and a range possible of launch dates to search. For the following analysis a 185 km circular parking orbit is assumed. To help minimize the total required ΔV the atmospheric entry speed is limited to a maximum of 12 km/s. Depending on the final Orion CEV reentry velocity a skip re-entry may be necessary. The results obtained using this program are presented in the following sections.

Launch Opportunities

For an NEO return mission two possible launch dates near the Earth-asteroid close encounter are always found. One launch always returns to the Earth near the Earth-Apophis encounter date, while the other launch date occurs on the date of the close encounter. Throughout the rest of the analysis the launch prior to the Earth-Apophis close encounter will be referred to as the early launch date/window, while the launch occurring near the close encounter will be referred to as the late launch date/window.

A plot of the total ΔV required for both the early and late launch dates versus mission length (ranging from 20 to 365 days) is shown in Fig. 1. As Fig. 1 shows the total ΔV is generally reduced as the length of the mission increases. Examination of Fig. 1 shows a minimum ΔV required at 180 days before a small increase. Current manned NEO studies have limited the maximum mission length to 180 days for supply and maximum radiation dose limitations. These limitations combined with the ΔV minimum at 180 days lead to the selection of a mission length of 180 days, which results in a required ΔV in the 10-11 km/s range. Lowering the total mission length may be possible depending on the final mission architecture and ΔV capabilities.

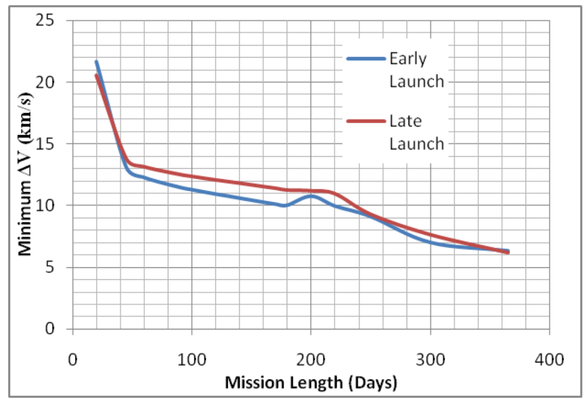


Figure 1: Plot of mission length versus minimum ΔV required to the early and late launch.

With a total mission length selected further analysis can be performed to find the dates and length of each launch window. This can be obtained by calculating the minimum launch ΔV for launch dates near the Earth-Apophis encounter. This information is shown in Fig. 2, which is a plot of launch dates versus the total required ΔV for the selected 180 day mission. As previously mentioned the first launch date occurs approximately 180 days prior to the April 13th, 2029 Apophis encounter, while the late launch date occurs on April 13th.

Further examination of Fig. 2 reveals that the launch window may be the last opportunities to launch a quick return mission to Apophis. Any manned missions to Apophis after the April 13th, 2029 launch date would likely require significantly increased mission times, possibly even multiple revolutions around the sun before rendezvous, to reduce the required ΔV to an obtainable amount. A mission of this length would likely require significant modifications to the Orion spacecraft to allow for greater radiation shielding and amount of supplies carried. For a short quick return mission to Apophis the April 13th, 2029 launch is the last easily obtainable

*It should be noted that any ΔV calculated after the 2029 Earth-Apophis close approach may not be accurate due to the uncertainty in how the orbit of Apophis will be perturbed.

launch date.

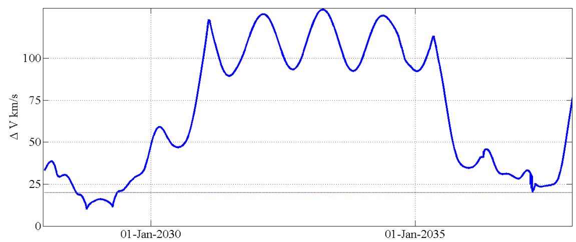


Figure 2: Launch date(2028-2038) versus minimum ΔV required for 180 day return mission*.

Limiting the maximum allowable launch ΔV to 11.5 km/s allows for sufficiently large launch windows. The minimum ΔV capability requirements for the mission are determined by allowing for a 0.5-1 km/s error margin. Adding this error margin to the maximum allowable launch ΔV results in a required ΔV capability of 12-12.5 km/s. Using this 11.5 km/s limit the launch windows can be found for both launch dates. The launch windows are then found by limiting the plot in Fig. 2 to the 2028-2029 time frame. The resulting graph is shown in Fig. 3, which has the total ΔV plot as well as the required ΔV for the Earth departure, Apophis arrival, Apophis departure, and Earth arrival burns.

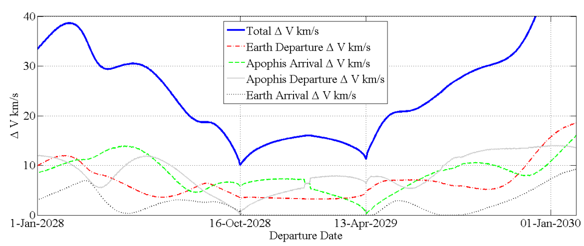


Figure 3: Launch windows found in the 2028-2029 time frame. Plot of launch date versus minimum total ΔV required.

Each separate launch window is shown in Figs. 4 and 5. As Fig. 4 shows the early launch window is approximately 12 days starting on Oct. 12, 2028 and ending on Oct. 24, 2028. The late launch window is significantly shorter at just over 2 days in length, ranging from Apr. 12-14, 2029.

A summary of nominal launch dates for both launch opportunities is shown in Table 2. The dates for each maneuver as well as the ΔV magnitude and C_3 values are given for each maneuver. For the early launch date all of the maneuvers, with the exception of the Earth departure burn, are carried out in the last 3 weeks of the mission. The return date for the early launch date is just after the Apr. 13th, 2029 Earth-Apophis encounter, which allows for a small return ΔV because the Orion spacecraft departs Apophis a few days prior the the Earth-Apophis encounter. The opposite is true for the late launch date. Earth departure occurs during the Earth-Apophis close approach, with the Apophis rendezvous occurring a few days after Earth departure. Within the first 2-3 weeks the mission is completed, witht the remaining time spent on the return cruise. No burn is necessary when the Orion spacecraft returns to the Earth because the atmospheric reentry speed is less than 12 km/s.

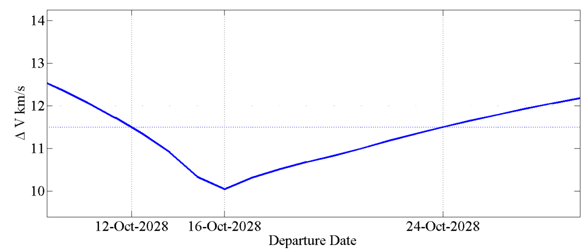


Figure 4: Plot of launch date versus minimum total ΔV required for the early launch date.

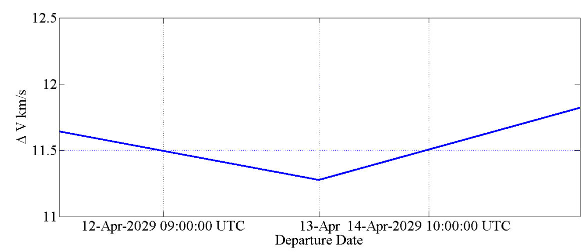
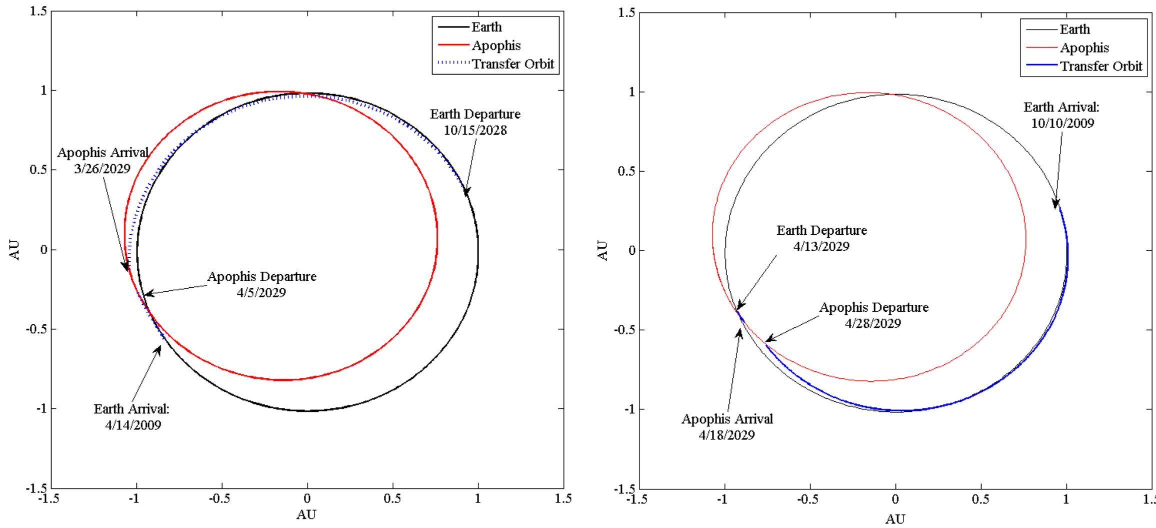


Figure 5: Plot of launch date versus minimum total ΔV required for the late launch date.

Rendezvous Mission Analysis

The purpose of this section is to briefly outline the requirements to send a precursory robotic mission or fueling station to Apophis. The mission design was performed using a similar program to that developed for a manned mission analysis. Only fast transfer orbits have been considered in the following analysis. The use of phasing orbits, gravity assist maneuvers, or multiple revolutions around the sun prior to rendezvous have not been considered.

Launch opportunities were found by search for the minimum total ΔV for each launch by allowing the arrival date to vary. The results of the search are shown in Fig. 7. Examinations of this plot shows several possible launch opportunities in the 2027-2029 date ranges.



(a) Early launch mission trajectory. Sun centered J2000 coordinate system.

(b) Late launch mission trajectory. Sun centered J2000 coordinate system.

Figure 6: Early and late mission trajectories.

Mission Information	Early Launch	Late Launch
Earth Departure		
Date	16-Oct-2028	13-Apr-2029
C_3	4.887	30.355
ΔV (km/s)	3.448	4.528
Apophis Arrival		
Date	26-Mar-2029	19-Apr-2029
V_∞^2	34.504	0.136
ΔV (km/s)	5.874	0.369
Apophis Departure		
Date	5-Apr-2029	29-Apr-2029
C_3	0.113	40.686
ΔV (km/s)	0.336	6.379
Earth Arrival		
Date	14-Apr-2029	10-Oct-2029
V_∞^2	30.474	1.896
ΔV (km/s)	0.391	0.000
Re-Entry V (180 km alt)	12.000	11.111
Total ΔV (km/s)	10.049	11.276

Table 2: Mission information for each launch opportunity.

A summary of the launch opportunities found is shown in Table 3. A total of 6 launch windows were found, however only the first 4 launch windows allows for and arrival date before or during the manned mission.

With two launch opportunities found, a mission length of 180 days, and a ΔV of 12 km/s determined to be the minimum allowable for a manned mission to Apophis the mission architecture options can be explored. In the following sections several possible mission configurations will be considered, with the ultimate goal of obtaining a ΔV of at least 12 km/s.

LAUNCH VEHICLE CONFIGURATION

The focus of this study is to determine the feasibility of a piloted mission to Apophis utilizing hardware developed for a lunar mission, with only minor hardware modifications. However, there is one major change to the overall lunar mission configuration. It will be shown that the Altair lunar lander must be replaced with an orbital tranfser vehicle(s) to fulfill the mission requirements. Under the current NEO mission studies 2-3 astronauts could be sent on a 180 day mission to Apophis (the Orion CEV will take 6 to the ISS and 4 to the moon). Due to large fuel and mass requirements only the Ares I and V launch vehicles were considered. Due to low payload mass performance EELV's were not considered, as they have been in previous NEO studies. It is also assumed that the Ares V will be man rated, allowing the Orion to be launched on an Ares V in place of the Altair lunar lander. The Ares I is only used to launch the Orion CEV in some of the considered architecture options. Both the Ares V and I are pictured in Fig. 8. For this study the Ares V version 51.00.48 is the assumed configuration.

The Ares V features a large 10 m payload fairing, which is required for a large cryogenic OTV. It has a 3,700+ mT gross lift off mass consisting of a first stage and a second stage, known as the Earth Departure Stage (EDS). For a lunar mission the EDS is used to circularize the orbit and perform the TLI burn. The performance of the Ares V can be summarized as follows. With a full Ares I and V launch approximately 71.1 mT can be delivered to TLI. This mass includes the Orion CEV, Altair lunar lander, and additional saftey mass margins. The Ares V will also be capable of launching 187.7 mT to a circular LEO. From the TLI performance numbers and an assumed 10% gravity loss the fuel mass in the EDS prior to the TLI burn can be estimated. This estimate is then used to explore additional mission architecture options.

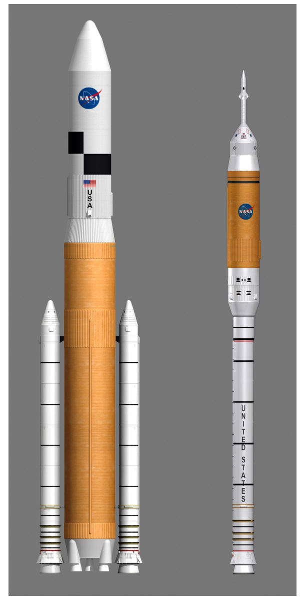


Figure 8: Current Ares V and I launch vehicles.

With mass estimates obtained several mission architecture options are to be explored with the ultimate goal of obtaining at least 12 km/s of ΔV capabilities. The standard lunar mission consisting of both an Ares I and V launch, which includes the Orion CEV, Altair lunar lander, and EDS is capable of just over a 6 km/s ΔV, well below the required 12 km/s for a manned Apophis mission. It's worth noting that a launch of an Ares V with the EDS and Orion CEV is capable of just over a 7 km/s ΔV. A list of several possible mission configurations, including the two discussed above, are shown in Table 5.

Given the relatively low ΔV capabilities of the current lunar mission configuration, it is clear that the use of an orbital transfer vehicle is required. In addition the Altair lunar lander will likely not be necessary. Due to the extremely low gravity of Apophis astronauts may be able to use a "tether" or small thrusters to ferry themselves to and from the surface, eliminating the need for the lunar lander. Instead, the Altair lander will be replaced with an OTV(s).

Fig. 9 is a plot of total ΔV capability obtainable when the Altair lander is replaced with an OTV of varying mass. It should also be noted that the configuration for this plot uses the EDS with the Orion CEV launch atop the Ares V. The OTV would then be inserted between the EDS and the Orion CEV similar to the current lunar architecture. Using this configuration Fig. 9 shows a plot for both a cryogenic and a bi-propellant OTV of varying mass. For both cases the structural mass is assumed to be 10% of the total OTV mass. The I_{sp} of

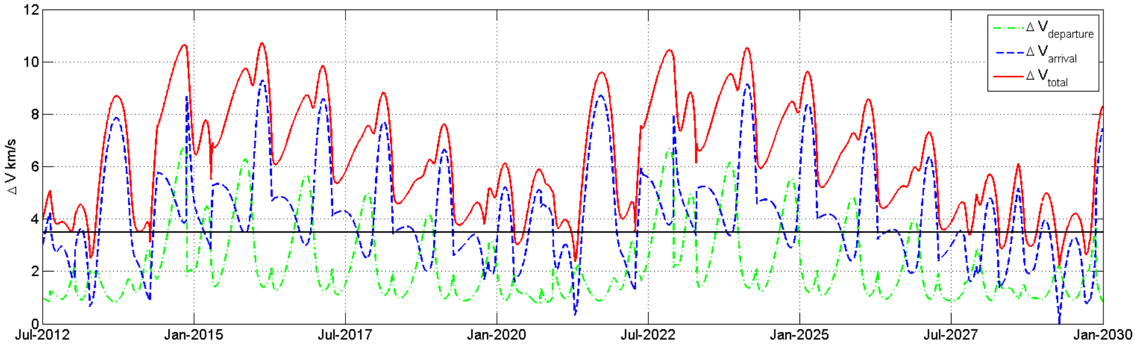


Figure 7: Launch date versus ΔV required for rendezvous mission.

Mission Information	**Launch-1**	**Launch-2**	**Launch-3**	**Launch-4**	**Launch-5**	**Launch-6**
Earth Departure						
Date	5/15/2027	10/20/2027	4/30/2028	10/23/2028	4/13/2029	9/23/2029
C_3	5.062	22.379	14.195	5.425	34.192	25.266
ΔV	0.996	1.741	1.395	1.012	2.224	1.861
Apophis Arrival						
Date	4/12/2028	5/5/2028	2/18/2029	6/23/2029	10/31/2029	7/26/2030
ΔV	2.594	1.640	1.473	1.939	0.019	0.773
Total						
ΔV	3.591	3.381	2.868	2.952	2.243	2.634

Table 3: Launch and arrival mission information for a single rendezvous mission. Used for the purpose of sending a “fueling station” or precursory robotic mission to Apophis.

the cryogenic OTV is assumed to be 448 s and the bi-propellant I_{sp} is assumed to be 325 s. Examination of Fig. 9 reveals that with both fuels the ΔV approaches a maximum limit, with the cryogenic OTV having a ΔV limit roughly 2.5 km/s greater than the bi-propellant OTV. It is also important to notice that the cryogenic OTV has a maximum ΔV limit or approximately 11.5 km/s, insufficient for the mission. The purpose of this plot is not to determine the size of an OTV necessary for the mission, but to show that the mission can't be completed through the use of a single large OTV.

With this maximum ΔV limitation for a single OTV (even with an infinite amount of fuel and infinitely large OTV) below the minimum ΔV required for a 180 day manned Apophis mission the use of multiple “small” OTVs is necessary. Performance is increased when using 2-3 small OTVs over a single large OTV because the large mass of a single OTV. By using multiple small OTVs some of the mass can be dropped each time an OTV is spent, leaving less total mass for the additional OTVs to carry, thus increasing overall performance. There is of course a practical limitation to the use of staging.

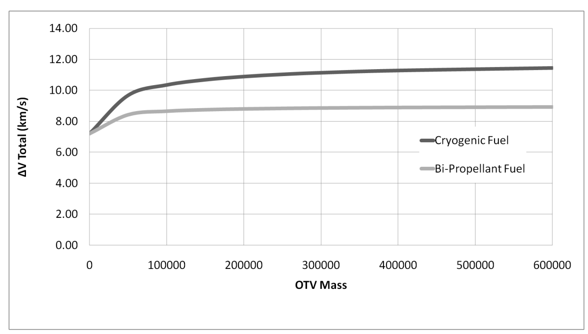


Figure 9: Plot of OTV mass versus ΔV capability. Found using an Orion CEV launched on an Ares V and a separately launched OTV.

A trade-off study was performed to determine the optimum mission configuration using 2-3 OTVs and/or an EDS stage as well. The properties of each OTV, EDS, ORION CEV, and lunar lander necessary to complete this trade off analysis are listed in Table 4. Several possible mission architectures are listed in Table 5 along with the total ΔV capability. With the required use of multiple OTVs a configuration requiring less than 3 Ares V launches that met the 12 km/s ΔV requirement could not be obtained. It should also be noted that through the use of bi-propellant OTVs the maximum obtainable ΔV was approximately 10 km/s. Through the use of a refueling station sent to Apophis prior to the manned mission it may be possible to obtain a ΔV of over 12 km/s. This assumes that nearly all the fuel is used during the Earth departure and Apophis arrival burns (the more fuel the Orion and OTV(s) have after the Apophis arrival burn the less additional ΔV is obtained from re-fuel). Launch of 4+ Ares V would likely be necessary to send enough fuel to Apophis to obtain a ΔV of 12+km/s. For these reasons the use of cryogenically fueled OTVs is required due to the low performance obtainable from bi-propellant OTVs.

With the requirement of cryogenic OTVs the final mission architecture can now be determined. The ultimate goal of the trade off study is to determine the minimum number of Ares I/V launches necessary to obtain the 12+ km/s required ΔV. Any of the configurations in Table 5 that contain OTV-2 are cryogenic possible mission configurations. There are 2 possible configurations possible to obtain the necessary ΔV without the need for refueling at Apophis.

The first option capable of completing the mission uses an Ares V to launch the Orion CEV along with 2 additional Ares V launches, each carrying a 180 mT cryogenic OTV. These two OTVs are arranged in series between the EDS and the Orion CEV. This architecture essentially replaces the Altair lunar lander with 2 OTVs and is capable of a ΔV of approximately 12.475 km/s. Docking the OTVs behind the more massive EDS and burning them first decreases the total mission ΔV capability by nearly 1 km/s. The second possible option requires the launch of 3 Ares V's, each containing a 180 mT cryogenic OTV, along with an Ares I/Orion CEV. For this configuration the EDS and Altair lunar lander are replaced by the 3 OTVs, which are docked in series. If the final Ares V is not man rated then option 2 will be required, otherwise option 1 is sufficient for a mission to Apophis and doesn't require the use of an Ares I launch vehicle, reducing the total number of launch vehicles to 3. Option 1 will be considered as the final mission architecture for the rest of this study. A detailed breakdown of the configuration and each component's ΔV capabilities is shown in Table 6.

With the selection of the mission architecture finalized a more detailed conceptual mission analysis will be completed in the following section. A preliminary design of the OTV will be presented, along with a more detailed mission concept and requirements.

MISSION DESIGN

The crewed Apophis mission concept is outlined in Fig. 10. As determined earlier a total of 3 Ares V launches are required. The two OTVs would be launched separately and later docked together in an LEO parking orbit prior to the Ares V/Orion Launch. The last launch would include the Orion CEV and a partially full EDS. The Orion CEV and EDS would rendezvous and dock on either side of the two docked OTVs. The Earth departure burn will likely use the entire EDS and the 1st OTV, it may even burn part of the 2nd OTV. As each stage is used to completion it is ejected and the next stage in line is used.

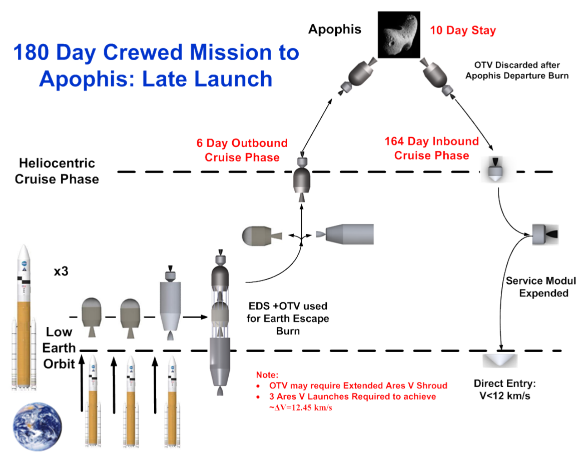


Figure 10: Crewed Apophis mission concept.

Due to the required use of cryogenic OTVs a mission utilizing the early launch date will not be possible without significant advances in the storage of cryogenic fuels in space. The majority of maneuvers for the first launch date are performed in the last 2-3 weeks of the mission life as shown in by the trajectory plot in Fig. 4 and Table 2. With the majority of maneuvers performed in the first 2-3 weeks of the late launch mission, the use of cryogenic fuels is possible. However 2-3 weeks is well beyond the 4 days loiter time currently allowed for the EDS before Orion rendezvous. The OTVs will need to be designed with a lifetime of 2-3 weeks as the final goal.

A preliminary conceptual design of the OTV has been performed and is shown in the Ares V fairing in Fig. 11. There has been no detailed analysis done on this conceptual design; it is merely a conceptual design showing how the OTV could be configured. As illustrated in Fig. 11, the OTV will take up the majority of the Ares V fairing, mostly due to the extremely low density of liquid hydrogen (approximately 70 kg/m^3).

*The total fuel deliverable to Apophis is expected to exceed the values given. Current calculation is based on Ares V 51.00.39, as this information was unavailable for the 51.00.48 version at the time of publication.

	Total Mass (kg)	Burnout Mass (kg)	Fuel Mass (LEO) (kg)	I_{sp} (sec)
Orion	20500	11204	9297	323
Altair Lunar Lander	43000	20000	23000	450
EDS (w/Altair)	120496	26600	93896	448
EDS (w/Orion)	142995	26600	116395	448
EDS (Full)	278500	26600	251900	448
OTV-1	180000	18000	162000	325
OTV-2	180000	18000	162000	448
OTV-3 (Fuel to Apophis)*	30280	4800	25480	325

Table 4: Table of items considered for the mission architecture and their accompanying properties. Fuel masses for the EDS (pre-TLI burn) are based on post TLI mass estimates [12].

Configuration	Total ΔV (km/s)	Required ΔV (km/s)	Additional ΔV Required (km/s)	Departure Orbit	Launch Window
Ares I, Orion + Ares V ,Altair	6.367	12.000	5.633	LEO	–
Ares V, Orion	7.206	12.000	4.794	LEO	–
Ares 1, Orion + Ares V, OTV-1	6.939	12.000	5.061	LEO	–
Ares 1, Orion + 2-(Ares V, OTV-1)	8.707	12.000	3.293		
Ares V, Orion + Ares V, OTV-1	8.757	12.000	3.243	LEO	–
Ares V, Orion + 2-(Ares V, OTV-1)	10.206	12.000	1.794	LEO	–
Ares V, Orion + Ares V, OTV-1 + Ares V, OTV-3	8.757 -12.247	12.000	3.243, -0.247	LEO,Escape	early,late
Ares 1, Orion + Ares V , OTV-2	8.930	12.000	3.070	LEO	–
Ares V, Orion + Ares V, OTV-2	10.748	12.000	1.252	LEO	–
Ares 1, Orion + 2-(Ares V, OTV-2)	11.367				–
Ares V, Orion + 2-(Ares V, OTV-2)	12.475	12.000	-0.475	LEO	late
Ares 1, Orion + 2-(Ares V, OTV-2) + Ares V, OTV-3	11.367 -14.857	12.000	0.633, -2.857	LEO,Escape	late
Ares 1, Orion + 3-(Ares V, OTV-2)	12.865	12.000	-0.865	LEO	late

Table 5: Several launch vehicle and mission architecture configuration options with the maximum possible ΔV for each listed configuration, along with possible launch windows.

Stage	ΔV Capability (km/s)	Total Mass (kg)	Propellant Mass(kg)	Dry Mass (kg)
EDS	1.108	523495.50	116395.12	26600.00
OTV-2,1	2.437	380500.38	162000.00	18000.00
OTV-2,2	7.250	200500.38	162000.00	18000.00
Orion	1.680	20500.38	9296.64	11203.73
Total	**12.475**		449691.77	73803.73

Table 6: Proposed mission configuration capable of completing 180 day mission to Apophis for the late launch. OTV Dry mass is assumed to be 10% of the total OTV mass [13,14].

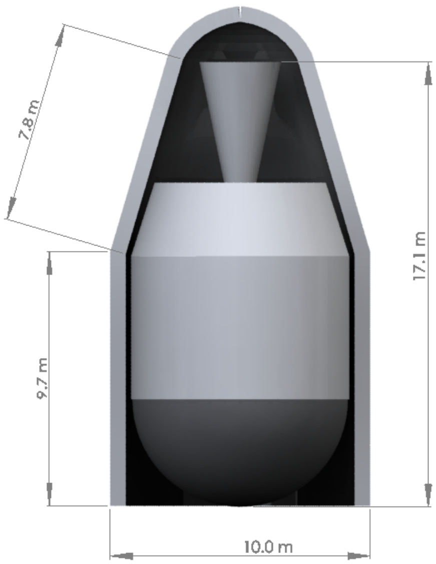
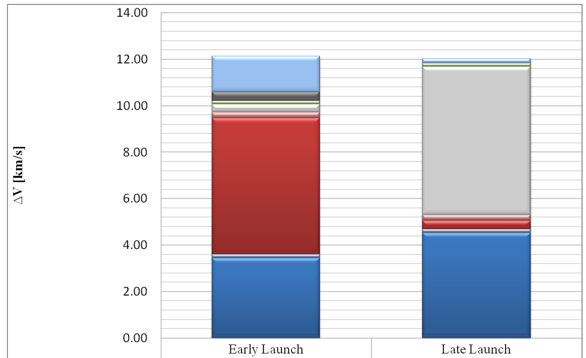


Figure 11: OTV shown in Ares V fairing.



	Early Launch	Late Launch
Additional Margin	1.532	0.196
Earth Arrival	0.391	0.000
TCM	0.150	0.150
Apophis Departure	0.336	6.379
Maneuvers at Apophis	0.250	0.250
Apophis Arrival	5.874	0.369
TCM	0.150	0.150
Earth Departure	3.448	4.528

Figure 12: ΔV budget for both the early and late launch dates using the final mission configuration. The Earth departure ΔV has been adjusted to account for a 10% gravity loss.

$\triangle$V Budgets

The ΔV budget for both the early and late launch windows is shown in Fig. 12. While the use of a cryogenically fueled OTV is not currently feasible for the first launch window it is included in the ΔV budget. If major breakthroughs in the long term storage of cryogenic fuels in space occur the first launch windows would be feasible. All further analysis will be done using only the late launch date.

As shown in Fig. 12, the ΔV margins are very tight with slightly less than 0.2 m/s ΔV capabilities left at the end of the mission. Any additional ΔV left at the end will likely be used to slow the Orion CEV prior to Earth atmospheric reentry. The current late launch mission has an Earth reentry velocity of approximately 11.1 km/s. This reentry velocity could be reduced by approximately 0.2 km/s in the current budget. A total of 300 m/s worth of correction maneuvers is budgeted for either launch dates along with a 250 m/s budget to perform and necessary maneuvers at Apophis. The current mission analysis has not performed the calculations to determine the necessary ΔV's for any maneuvers at Apophis. The current estimate is based on estimates from previous Apophis mission studies for robotic missions. The estimate of 250 m/s is roughly 2/3 of the Apophis arrival ΔV for the late launch date and should be sufficient. If the ΔV margins are later determined to be insufficient the second mission configuration could be used (3 OTVs and an ORION CEV) to relax ΔV constraints.

CONCLUSION

Asteroids have collided with the Earth in the past and are predicted to do so in the future. Through the use of Constellation program based technologies humans will have the ability to launch manned NEO mission aimed at preventing an Earth-asteroid collision. It has been proven feasible to send a crewed Orion mission to Apophis, which is capable of a total ΔV of over 12 km/s. This can be obtained through the use of multiple cryogenic orbital transfer vehicles in place of the Altair lunar lander. While it is more desirable to use a single OTV it has been shown impossible to obtain the necessary ΔV from such a configuration. The design of such an OTV could also be used for a Mars mission, allowing for a test of a Mars transfer vehicle. It has also been shown that the 2028-2029 timeframe may be the last launch windows for a quick return manned Apophis mission (180 days or less).

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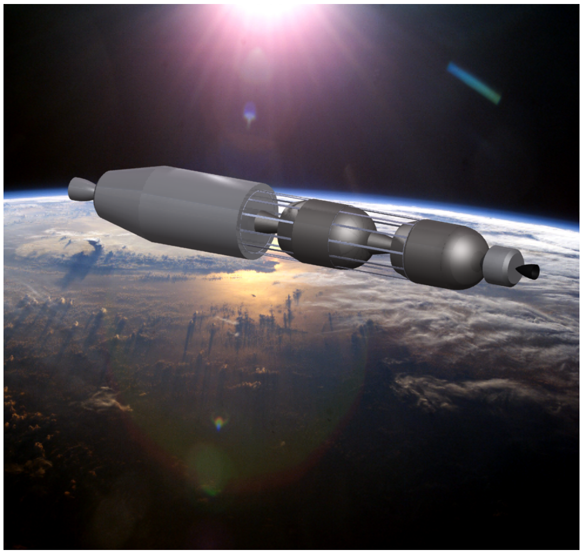

Figure 13: Launch vehicle final configuration in LEO.

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