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APPLICATIONS OF AN ARES V-CLASS HLV FOR ROBOTIC NEO DEFLECTION MISSIONS

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This study examines near-Earth object (NEO) deflection missions employing an Ares V-class heavy launch vehicle (HLV). The missions utilize high energy deflection methods involving various nuclear explosion variants, which are capable of fragmenting or ablating a potentially hazardous NEOs. Three different types of detonation options are also examined for each terminal phase scenario. The asteroid Apophis is used as a baseline to demonstrate mission design and system architecture examples in the 2029 to 2036 time frame. This study shows that with the capabilities of an Ares V-class HLV, nearly continuous launch windows exist for rendezvous and direct intercept missions. To help illustrate the performance advantages that an Ares V-class HLV could have to NEO deflection missions, the Delta IV-H launch vehicle is used in a comparison study. It is shown that the Delta IV-H launch vehicle is extremely inadequate for the types of deflection missions examined in this study.

INTRODUCTION

Asteroids and comets have collided with the Earth in the past and will do so in the future. Throughout Earth's history, these collisions have had a significant role in shaping Earth's biological and geological histories. For example, the extinction of the dinosaurs is believed to have been caused by an impacting asteroid or comet. In the recent past, near-Earth objects (NEOs) have collided with the Earth, most notably the Tunguska event of 1908. The impact in Siberia is estimated to have released an explosion on the order of three to five megatons of TNT. The Tunguska event can be considered relatively harmless given the sparse population of Siberia. However, if an impact of this relatively low magnitude were to occur in a highly populated area, the result would be devastating. For the first time in history, the technical knowledge and capability to launch a deflection mission to help ensure the survival of the human race exists.

Recent studies at the Asteroid Deflection Research Center (ADRC) of Iowa State University have concluded that it may be feasible to significantly reduce the impact damage from an Earth-impacting NEO using a nuclear subsurface explosion as late as 15 days prior to impact.[1] The subsurface explosion would be used to fragment the asteroid in such a way that a very small percentage of the asteroid impacts the Earth. This method is viable only for relatively small NEOs with diameters

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of at most a few hundred meters. This method would require a much longer mission lead time for asteroids with diameters in the kilometer range. A mitigation mission within a few weeks of impact requires a nuclear subsurface explosion due to large energy required to fragment the asteroid and ensure the majority of the resulting fragments disperse sufficiently to miss the Earth. To accomplish this type of asteroid deflection, two different types of robotic missions can be used. This study examines rendezvous and direct intercept missions that are capable of delivering a nuclear explosive device (NED) to an asteroid. Other options for deflecting hazardous NEOs do exist, such as utilizing a kinetic impactor or a gravity tractor. However, these options require the mission to be executed somewhere between 5 and 10 years prior to a potential impact to impart a significant ΔV to perturb the NEO's orbit. By the time an NEO is identified or determined to be hazardous, this amount of advanced warning time may not be available. Therefore, the high-energy method of using NED payloads to fragment or disrupt an NEO is the only deflection method being considered in this study.

This study offers a modified version of the interplanetary ballistic missile (IPBM) system architecture that can be applied to an Ares V-class heavy launch vehicle (HLV).[2] An IPBM system consists of a two major components; a launch vehicle and an integrated space vehicle (ISV). The ISV consists of an orbital transfer vehicle (OTV) and a terminal maneuvering vehicle (TMV) which delivers a bus and an NED to the asteroid. The proposed IPBM system architectures examine options for subsurface, shallow burst, and contact/stand-off burst explosions of the NED. Three different variants of NED penetrators are considered as payload options, with each NED penetrator variant having a different explosive yield magnitude. Preliminary system architecture options for each class of deflection mission are analyzed in an attempt to assess the feasibility of these missions.

NEO Deflection Mission Concepts

As previously mentioned, two deflection mission concepts are analyzed throughout this study, the first of which is a rendezvous terminal phase. The terminal phase expression describes the portion of the deflection mission 24 hours prior to NED detonation. In the case of a rendezvous terminal phase, the ISV performs an arrival burn at the asteroid to closely control the relative velocity of the ISV with respect to the asteroid. In nearly all cases, the ISV must reduce its velocity to more closely match the asteroid's speed, if the rendezvous approach occurs from behind the asteroid itself. Once the relative speed between the ISV and the asteroid is around 1 km/s, the NED penetrator can be released towards the asteroid before the TMV bus can reduce its relative velocity to zero. This allows the NED penetrator to more easily control the detonation process, while the TMV's bus can perform proximity operations to trail behind the target NEO, if needed.

Direct intercept missions theoretically require no asteroid arrival maneuver, which helps to keep ΔV requirements lower than in the case of rendezvous missions. This allows for direct intercept missions to carry larger payloads, which in this case, means higher yield NED penetrators can be used for deflection. However once approaching an NEO, reserve propellant may be needed for trajectory correction maneuvers, which helps to ensure a successful intercept of the asteroid. For direct intercept missions, the relative velocity between the TMV and the asteroid is not limited.

99942 Apophis as a Baseline

In recent years, the asteroid 99942 Apophis has gained popularity due to its once high Earth-impact probability. Although the odds of an impact are now quite low, Apophis could enter into a resonant orbit that results in an Earth collision seven years later if the asteroid passes through a keyhole during its April 13th, 2029 close-encounter with the Earth. For this reason, the asteroid

Apophis is used as an example case to show sample mission design results and different IPBM system architectures. The Apophis mission scenarios presented in this study are merely a demonstration of the capabilities of the mission analysis software developed in-house at the ADRC, and some possible NEO deflection applications of the proposed Ares V HLV. The methods exhibited in this study can be applied in a similar manner to any NEO if a deflection mission is deemed necessary.

To make the mission scenarios more realistic, only deflection missions which occur after the confirmation of the 2029 keyhole passage are examined. Also, all of the mission design is done using a a hypothetical Apophis orbit which results in the 2036 impact. This allows for a more accurate description of the last minute mission requirements for the rendezvous and direct interception cases. The orbital elements of the hypothetical perturbed Apophis orbit are given in Table 1.[3] Some of the physical parameters of Apophis can be seen in Table 7, which is located in Appendix A.

Table 1. Hypothetical Apophis Impact Orbit Characteristics

Orbital Element	Value
Epoch MJD	64699
Semi-Major Axis, AU	1.108243
Eccentricity	0.190763
Inclination, deg	2.166
Longitude of the Ascending Node, deg	70.23
Arguement of Periapse, deg	203.523
Mean Anomaly, deg	227.857

MISSION DESIGN

In this section, possible launch windows for deflection missions, both rendezvous and direct intercept missions to Apophis, are determined using software developed at the ADRC.[4] The time frame of these missions is between the close encounter in 2029 and the impact date in 2036. Due to the limited seven year mission window, multiple-revolution mission are not examined in this study. All of the mission design results assume that Earth-departure occurs from a 185x35,786 km geostationary transfer orbit (GTO). In addition, asteroid arrival is constrained to occur at least 15 days prior to the fictional impact, which ensures adequate time for the fragmentation and dispersion of the approaching NEO.[1]

The mission design software is a combination of Lambert solvers, ephemeris data, and numerous other functions. A search of all possible launch windows is conducted given the launch dates to check, maximum mission length, and desired time step (one day for the following analysis). For this case, approximately 2.4 million launch and arrival combinations were checked. In general, the minimum ΔV solution is found within the first 500 days. However to ensure that the absolute minimum is found, mission lengths of up to 1000 days are checked.

Apophis Rendezvous

To help visually illustrate how the rendezvous mission design changes over the seven year time frame that is examined, two plots have been provided. While Figure 1 shows the minimum ΔV requirements for each launch date, Figure 2 uses a porkchop plot to show how flight time and the

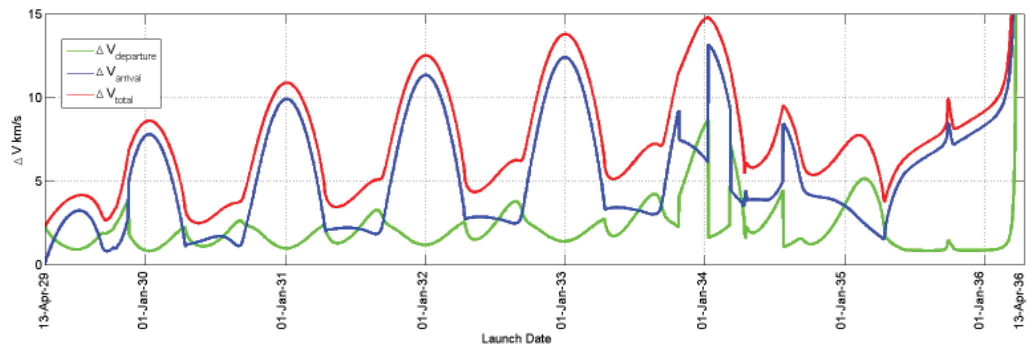

Figure 1. Minimum ΔV Requirements From 2029 to 2036

minimum ΔV are related for each launch date. The contour lines in the porkchop plot indicate a ΔV in km/s.

13-April-2029 to 1-January-2031: Since this time frame is near the last close encounter, some favorable launch windows exist. Immediately after the close encounter, ΔV requirements are as low as 2.4 km/s. It should also be noted that launch windows with flight times of less than 200 days are available. It is obvious that this time frame is where the lowest ΔV requirements are found. This allows for heavier payloads and higher yield NED penetrators to be utilized in a system architecture. The trade-off here is that accessing these launch windows requires advanced notice. These missions would have to be launched at minimum, five and a half years prior to impact. It may be the case that this amount of warning time may not be available for some NEO deflection missions.

1-January-2031 to 1-January-2034: In this time frame, launch windows start to become smaller and ΔV requirements for the launch windows increase. The minimum ΔV in this period is approximately 3.5 km/s and can be seen in May 2031. Although these missions require larger ΔVs and flight times, they are still in the feasible range, assuming system architectures exist that are capable of providing a ΔV of around 5.35 km/s. The possible architectures in this time frame may be somewhat limited, but rendezvous missions are still achievable.

1-January-2034 to 13-April-2036: This time frame designates the late launch windows for the Apophis scenario. Rendezvous mission options start to become very limited, with the minimum ΔV requirements only falling below 5 km/s in one window. In the 11 months prior to impact, it can also be seen that the ΔV requirements start to increase dramatically due to the large arrival burns that are necessary. If it is desirable to wait as long as possible to launch a deflection mission, prolonging a launch by as little as one month could place the mission into the infeasible range. For this reason, there appears to be no benefit to delaying a launch towards the later half of 2035.

It is important to note that the rendezvous mission design assumes that the asteroid arrival burn matches the ISV speed with the NEO speed. In other words, after the burn is executed, there is no relative velocity between the asteroid and the ISV. Since many different detonation options exist for rendezvous missions, the asteroid arrival burn could actually be split between vehicles during the terminal phasing process. A more detailed description of these situations is presented in later sections of this paper. Regardless of how the arrival burn is performed, Figures 1 and 2 still provide an accurate description of launch window availability for rendezvous missions.

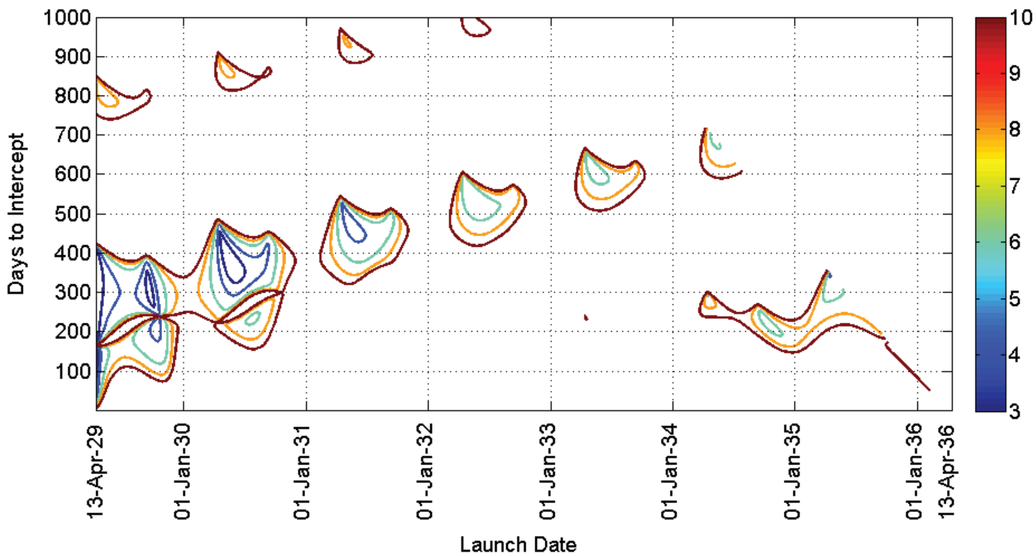

Figure 2. Rendezvous Porkchop Plot Showing ΔV Requirements vs. Transfer Time

Apophis Direct Intercept

Although the rendezvous missions are considered as the baseline for this study, alternatives do exist. Since the option to wait as long as possible to execute a deflection mission may be valuable, direct interception must be examined. Direct intercept missions can also provide nearly continuous launch windows, by filling the gaps in which rendezvous missions are not possible. To see the ΔV and flight time requirements for direct intercept missions during the post-2029 time frame, a porkchop plot can be seen in Figure 3. Again, for the porkchop plot, the contour lines indicate a ΔV in km/s.

13-April-2029 to 1-January-2031: Just as in the case of the rendezvous missions, low ΔV launch opportunities for direct intercept missions exist just after the keyhole passage in 2029. Not only do these missions have very low ΔV requirements, some of them even have mission lengths less than 100 days. This combination of low ΔV requirements low flight times makes this launch window very attractive for direct intercept missions. If a direct intercept mission in this time frame were to experience any type of failure, multiple other launch windows still exist.

1-January-2031 to 1-January-2034: A large number of launch windows still exist, however a significant jump in flight time can be seen for these missions. Missions with ΔV requirements ranging from 1 to 3 km/s are still frequent, as long as the increased flight times are not a problem. In the last year of this time frame, launch windows which present flight times of less than 200 days become available again. If it is determined that lower flight times are ideal, launch windows in this time frame still exist.

1-January-2034 to 13-April-2036: The most interesting aspect of this time frame can be seen just prior to the Apophis impact. Last-minute missions exist evan after January of 2036. These missions can still have low ΔV requirements, and arrival to Apophis occurs at around 15 days prior to impact. This would still allow for enough time to properly execute the deflection mission.

Missions could even launch as little as 30 days prior to impact since the flight times here are so small. The downsides to utilizing these last-minute launch windows are in reliability. Any mission failure, whether it occurs in the system architecture or the execution of the deflection process, would be catastrophic. Since these last-minute missions have relative velocities as high as 18 km/s at Apophis arrival, the challenges associated with a direct intercept mission would be most prevalent in these missions. Therefore, multiple launches would need to occur simultaneously to increase the chances of a successful deflection mission.

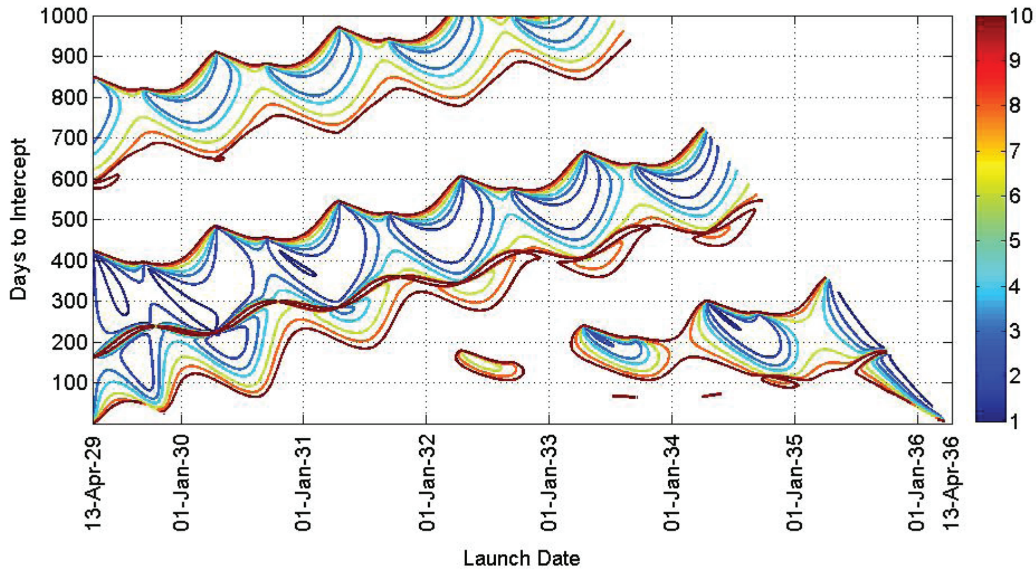

Figure 3. Direct Intercept Porkchop Plot Showing ΔV Requirements vs. Transfer Time

IPBM SYSTEM ARCHITECTURE OVERVIEW

With the mission design work for two types of NEO deflection methods completed, ΔV requirements can be used to examine conceptual system architecture designs capable of performing these missions. These architectural configurations are based on the IPBM system, which has been previously applied to launch vehicles such as the Delta IV and the Taurus II.[2] For this study, the IPBM architecture is applied to the Ares V HLV. It is important to note that the Ares V is used only as a reference in this study, to show the capabilities of a HLV of this class. As previously mentioned, the Ares V carries an integrated space vehicle (ISV) into GTO. The ISV is made up of an orbital transfer vehicle (OTV) which is used for performing the mission's burn schedule, and a TMV bus which carries an NED penetrator to the asteroid. An in-depth description of each component of the IPBM system is provided, starting with the launch vehicle.

Ares V-Class HLV Launch Vehicle

As a part of the Constellation program, the Ares V had been designed as a cargo launch vehicle to complement the man-rated Ares I. It had been designed to assist in the expansion of a human presence outside of low-Earth orbit (LEO). However, a launch vehicle of this size could also be used for robotic asteroid deflection applications. The Ares V is designed to deliver just under

69,500 kg to GTO (185x35,786 km) and would be the largest launch vehicle ever created.[5] 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). Six upgraded RS-68B engines provide thrust for the core stage, while the EDS uses a single Saturn V-derived J-2X engine. Both the core stage and the EDS make use of liquid oxygen (LO_2) and liquid hydrogen (LH_2) cryogenic fuels. The EDS can hold nearly 252,000 kg of usable propellant, but most of this is used when payloads are placed in GTO.[6]

Even if the Constellation Program becomes permanently canceled, the IPBM configuration based on an Ares V sized launch vehicle shows the benefits and versatility of an HLV. 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 any future NEO deflection mission.

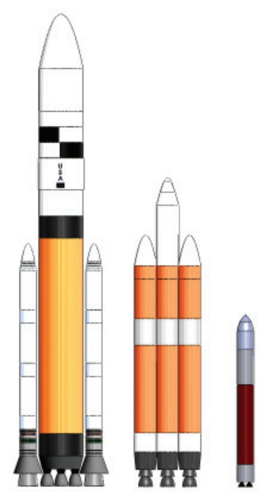

Figure 4. Ares V, Delta IV-H, and Taurus II Size Comparison

Delta IV-H Comparison When analyzing the capabilities of the current class of heavy launch vehicles (Delta IV-H, Atlas V-H, etc.), it is evident how beneficial a new class of heavy-lifter could be to these mission types. The Delta IV-H launch vehicle has a first stage which includes three common booster cores, all of which use cryogenic propellants. The second stage is also cryogenic, and is used to place just under 23,000 kg to LEO or 13,400 kg to GTO.[7] It can place more into orbit than any other currently available launch vehicle. To illustrate the comparison between the two launch vehicles, it is possible to examine each launch vehicle's deliverable mass to Apophis. This can be seen in Figure 5. Using the GTO mass performances, either a desired payload or ΔV is designated, and the other is calculated. This comparison assumes that the mass in GTO is used for an OTV upper stage and any arbitrary payload. The OTV is assumed to have a burnout mass of 10 percent of total OTV mass. It should also be noted that the burnout mass of the OTV is not included in the deliverable mass to Apophis. Although it may not be evident yet, this comparison study shows that the Delta IV-H is extremely inadequate for almost any NED mission. It will be shown later in the study that the Delta IV-H cannot even place two of the three NED payloads into a GTO orbit. Multiple launch architectures could be used, but this would not increase the number of deflection mission options, and would greatly increase the complexity of the overall system architecture. A similar comparison plot can be referenced in Appendix A as well.

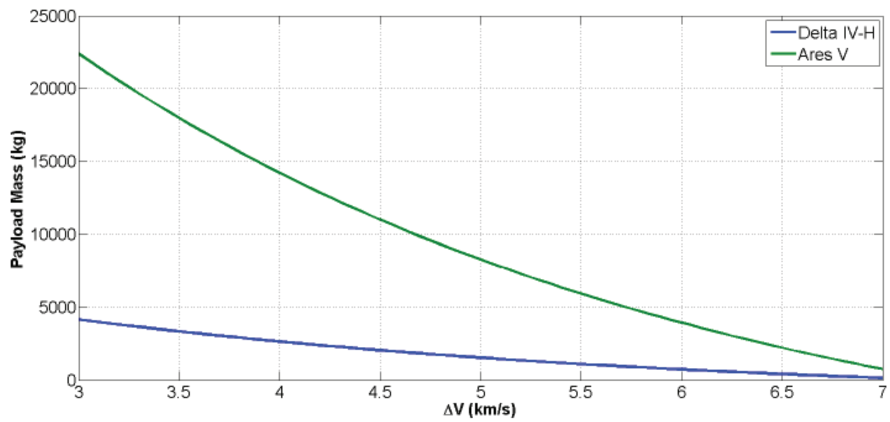

Figure 5. Capability Comparison

Orbital Transfer Vehicle Options

Since it is assumed that the EDS is exhausted, an additional upper stage must be included in the system architecture to provide the burn scheduling. The baseline OTV for this study is a bi-propellant system which uses monomethylhydrazine (MMH) and a nitrogen textroxide (N_2O_4) oxidizer. The system has a specific impulse of 323 seconds, which is lower than that of cryogenic fuels. Cryogenic propellants can offer high specific impulses (I_{sp}) values in the 450 second range. A configuration utilizing cryogenic propellants, much like the EDS, can yield higher ΔV capabilities for a system architecture. However, a cryogenic OTV has some distinct disadvantages for rendezvous missions.

The long-term storage of cryogenic propellant is difficult to achieve. Since rendezvous mission lengths vary drastically from 200 to 500 days, propellant storage could significantly reduce system architecture performance. Propellant boil-off, which occurs due to the difficulty in keeping hydrogen and oxygen in liquid form, can reduce the amount of available propellant by 4.5 percent every 30 days.[8] In addition, engine restarts are more of a concern with cryogenic engines. Since LO_2/LH_2 engines require a sparking plug to reignite the gaseous mixture, any failure in the plug could potentially result in mission failure. However, bi-propellant monomethylhydrazine and nitrogen textroxide are hypergolic, meaning combustion occurs by simply mixing the two in liquid form. Bi-propellant systems are more reliable for applications such as rendezvous missions, which require engine restarts. Since there are currently no upper stages in the size range of the necessary OTV, the development of such a vehicle would be necessary. However, it seems reasonable to assume that if the EDS is completed, a scaled-down, bi-propellant vehicle similar to the EDS could be developed.

Again as an estimate, the OTV's burnout mass is assumed to be 10 percent of its total wet mass. The total wet mass of an OTV can be found by designating the TMV mass, and subtracting it from the available 69,500 kg in GTO.

For direct intercept cases, it would be possible to utilize a cryogenic OTV. Since these missions require only an Earth-departure burn, the cryogenic propellant is only stored for a short period of time. The propellant in the OTV would be expended within one week, and would only be subject to a small amount of boil-off. Small trajectory correction maneuvers may still be necessary, but the TMV would most likely be capable of performing them. The use of a cryogenic OTV would further enhance the launch window availability or the payload capabilities for these direct intercept cases. Another possibility for the direct intercept missions involves eliminating the OTV altogether. By eliminating the OTV, there could be remaining propellant in the EDS once it reaches GTO. The available ΔV of the system becomes greatly reduced however, since the BOM of the EDS is a very large fraction of the available 69,500 kg ISV. The only reason that this would be done is to reduce the complexity of the IPBM system. Relying strictly on the EDS would eliminate the need for a separate OTV to be developed in this case. In architectures using just the EDS to perform Earth-departure, only a handful of direct intercept launch windows exist.

Terminal Maneuvering Vehicle Overview

The terminal maneuvering vehicle consists of two, separate systems: a spacecraft bus and an NED penetrator. The main purpose of the bus is communication, guidance, and navigation of the system during the cruise phase to Apophis. Designs for the TMV bus are based on the Dawn spacecraft, which will study Ceres and Vesta: the two largest asteroids located within the asteroid belt. Mass estimates for many of the control subsystems mirror those contained aboard the Dawn satellite.[9] Major changes can be seen in the mechanical structure, electrical power system, and propulsion system. The structure has been modeled after the Orbital STAR-2 series, which is the same design used by Dawn. However, this structure is more robust to offer additional space for mounting hardware between the center cylinder and the outer panels as well as the propellant tanks. The increased structural mass also accounts for the attachment fitting used to connect the TMV bus to the NED penetrator. Although the Dawn spacecraft uses a flight-proven Ion Propulsion System (IPS) from the Deep Space 1 spacecraft, ion propulsion is not used in the deflection mission designs. Unlike the Dawn spacecraft, the spacecraft bus uses a bi-propellant propulsion system. This system is responsible mainly for controlling the bus's relative velocity and for any correction maneuvers made by the reaction control system (RCS) thrusters. The bi-propellant system uses MMH as the main propellant and N_2O_4 as an oxidizer. The propellant tanks on the bus are sized to provide a ΔV of 1250 m/s for these maneuvers. This design has been examined in-depth during previous IPBM applications and specifications for the TMV bus can be seen below in Table 2.[2] The bus contains all the necessary subsystems to function without the OTV.

The other system included in the TMV is an NED penetrator, which is used as the deflection payload. Although the terminal phases are unique for rendezvous and direct intercept cases, the goal of the payload is the same in both cases. The energy produced from the detonation of the NED penetrator is intended to fragment or ablate the approaching asteroid. NED detonation can occur in three different scenarios; subsurface detonations occur at depths of at least seven meters, shallow bursts occur between two and three meters, and contact/stand-off bursts occur at the surface or at some nominal distance away from the asteroid body. As previously mentioned, studies at the ADRC have concluded that a subsurface detonation of a 1 megaton (MT) NED can fragment an asteroid of

Table 2. TMV Satellite System Breakdown

System Component	Mass (kg)
Mechanical Structure	250
Propellant Tanks	55
Electric Power System (EPS)	133
Attitude Control System (ACS)	37
Reaction Control System (RCS)	14
Thermal Control System (TCS)	44
Command and Data Handling System (CDHS)	21
Telecommunications	28
Balance	13
NFOV	10
WFOV	12
LiDAR	15
Uncertainty	25
Bi-propellant Fuel	325
Total Wet Mass	982

up to a few hundred meters so that nearly all of the approaching asteroid debris misses the Earth.[1] Subsurface explosions are a more effective way of transferring energy than the shallow burst or stand-off scenarios. Since subsurface detonations are more effective, smaller yield NED penetrators can be used to fragment an asteroid body.

A set of three NED penetrators has been proposed that is capable of achieving three, different detonation scenarios.[10] All three variants share the same basic on-board systems. These include a nuclear charge, triggering mechanisms, and control systems. However, the three variants do exhibit some differences as well. The variant-1 penetrator is designed to achieve subsurface detonations with depths of at least seven meters at relative impact velocities of 800 to 1500 m/s. The shallow burst detonation can be achieved at relative velocities between 2 and 30 km/s. Both cases are complicated processes, since the NED triggering device and other on-board systems must be protected from extremely high impact heating and forces. To make this sequence possible, a shape charge is encased inside the penetrator, which is used to break apart the asteroid's top surface. This allows for a conical, leading shield to push deeper into asteroid's outer surface. To protect NED triggering mechanisms and detonation sequencing controls from harm, a penetrating module shrouds the impacting side of the nuclear charge. A breakdown of the NED penetrator and its subsystems can be seen in Figure 6.

The variant-2 penetrator is a scaled-up version of its variant-1 counterpart, and is capable of subsurface detonation as well. With a higher explosive yield and a larger outer frame, the variant-2 penetrator could be used for larger asteroids, or for a more effective shallow burst scenario. The variant-3 penetrator is quite different from the previous two designs. The variant-3 design is not intended to perform subsurface or shallow burst detonations, and therefore includes no shape charges or leading shield. Since the variant-3 penetrator is capable of only contact bursts or stand-off detonations, the explosive yield is much higher to make up for its lack of penetrating capability.

The penetrator variants are currently designed without a propulsion or reaction control system for correction maneuvers. Therefore, the modified mass column in Table 3 shows the estimated masses

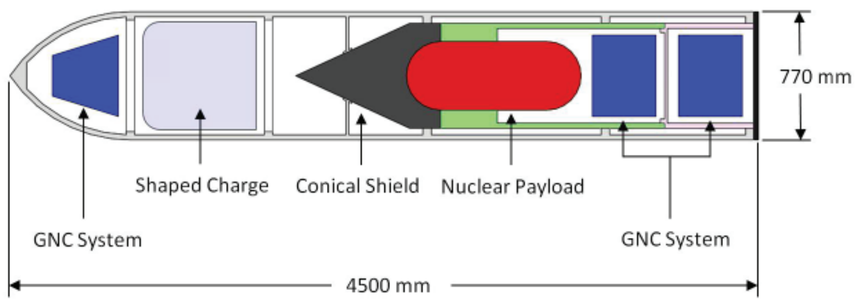

Figure 6. NED Variant-1 Internal Component Breakdown

Table 3. Nuclear Penetrator Classes[10]

NED Variant	Explosive Yield (MT)	Diameter (m)	Length (m)	Mass (kg)	Modified Mass (kg)
1	2	0.77	4.50	3,500	3,900
2	10	1.30	7.60	16,000	17,800
3	100	2.30	8.00	25,000	-

of each penetrator variant if they were to be fitted with bi-propellant systems capable of providing ΔVs of just over 300 m/s. This ΔV is provided for any small correction maneuvers that may be needed to ensure accurate NEO impacting. For these calculations, the burnout mass of the added system is assumed to be roughly 10 percent of the total added mass. This additional propulsion system is only necessary for rendezvous missions in which the NED variant is separated from the TMV bus and OTV prior to asteroid impact. Since the variant-3 penetrator is used only for stand-off or contact burst detonations, no separate propulsion or attitude systems need to be added to the design. It is understood that adding small RCS systems and propellant changes the dimensions or layout of the penetrator design in a slight manner. Also, refined NEO composition and structure estimates could lead to design changes in the NED penetrators as well. Although this is important to note, it is outside the scope of this study, and is therefore not addressed further.

SYSTEM ARCHITECTURE PERFORMANCE

To estimate the performance capabilities of different types of system architectures, it is necessary to calculate the available ΔV each configuration can provide. To perform these calculations, a standard rocket equation is used. In it's simplest form, drag forces are neglected and the assumption that all burns are impulsive is made. If this assumption is not made, burn times for each launch window would be required. The form of the rocket equation used in the ΔV performance calculations is given by[11]

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

It is apparent from the rocket equation that the ΔV capability of a system is dependent upon the ratio of the fully loaded mass to the burnout mass, and the I_{sp} performance of the system's engine.

Rendezvous Mission Options

For this study, rendezvous missions are considered to be the baseline terminal phase. Again it is important to note that rendezvous scenarios do not necessarily require zero relative velocity after the asteroid arrival burn. These low speed missions offer many benefits over direct intercept missions. For example, subsurface detonations are only possible in the case of a rendezvous mission. Also, impact speeds and approach flight-path angles can be more easily controlled in this terminal phase scenario. The option to perform a more structured stand-off detonation also exists with a rendezvous mission. Location of the contact burst or stand-off detonation can be more accurate with the lower relative velocities. However, the benefits from these missions are also hindered by some design trade-offs.

Table 4. Rendezvous System Architecture Options

TMV Mass (kg)	Available ΔV (km/s)	Number of NEDs	NED Type
4,882	5.742	1	Variant-1
8,782	4.888	2	Variant-1
12,682	4.216	3	Variant-1
18,782	3.387	1	Variant-2
25,982	2.626	1	Variant-3

Higher ΔV requirements exist due to the presence of asteroid arrival maneuvers which control the ISV's relative velocity. Another downside to rendezvous missions is the amount of lead time that is necessary for some of these missions. Again, since flight times range anywhere from 200 to 500 days, these missions cannot be applied to cases in which little impact warning is provided. As previously mentioned, in the case of the hypothetical Apophis impact, rendezvous missions become infeasible at around 11 months prior to impact. Many possible system architectures still exist for these rendezvous missions, despite some of the trade-offs that are made. Table 4 lists five sample architectures and the performance capabilities that are associated with each architectural design. Since the OTV uses a bi-propellant system, the I_{sp} used in these calculations is 323 s. The TMV mass in Table 4 is the deliverable mass to Apophis, which in this case represents the TMV bus and NED penetrator variant. This mass does not include the burnout mass of the OTV. It should be understood that this mass can be used for any TMV design. If the TMV design presented in this paper is deemed undesirable for any reason, the mass in Table 4 can be allocated for a different payload.

It can be seen from these results that a rendezvous nuclear stand-off or contact burst with a variant-3 penetrator severely limits launch opportunities. However, a few small launch windows in 2029 and 2030 may make such a mission possible. Depending on the desired relative velocity of the contact burst detonation, more launch windows could become available. Even allowing for the contact burst to occur at 500 m/s instead of a zero relative velocity can decrease the rendezvous ΔV requirements. This helps to extend some of the launch windows, or helps to leave an additional ΔV margin.

To better illustrate the IPBM system and each of its components, a concept design showing the variant-1 penetrator is shown in Figure 7. The ISV system is shown in an Ares V fairing. For comparison, the Delta IV-H fairing is just over 19 m, and only has a 5 m diameter. Due to the size difference between the two components of the TMV, future studies should investigate how to

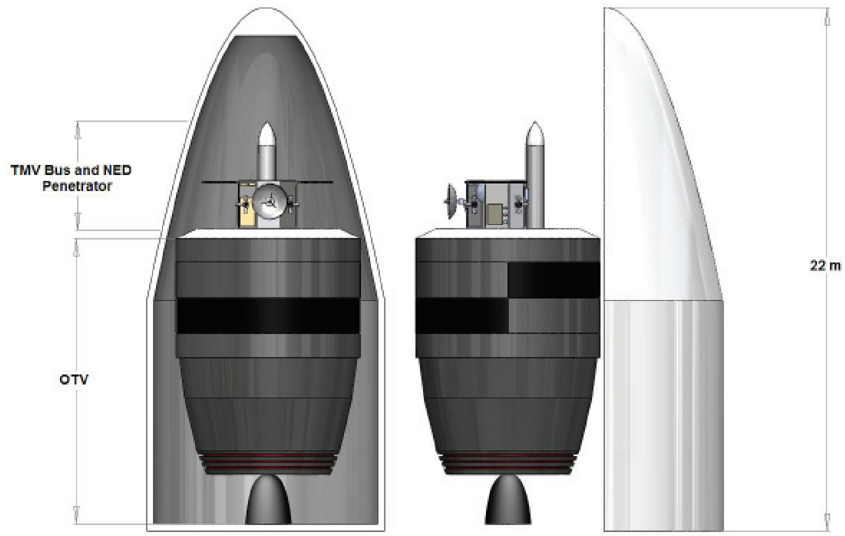

Figure 7. Illustration of Possible IPBM Configuration Insed an Ares V Fairing

appropriately connect the two systems. However, for the purpose of this study is to outline the performance capabilities of the Ares V in its application to NEO deflection missions. This TMV design is preliminary, and should be revisited in future studies.

Sample Case of a Late Rendezvous Mission To help illustrate how a planned rendezvous mission might look, the June 7, 2035 launch date is examined. In this scenario, it is the last date with the total ΔV requirements being under 6.35 km/s. Table 5 shows a breakdown of the mission burn schedule and the transfer orbit. A sun-centered trajectory of this late rendezvous case can be seen in Figure 8. Although the 6.35 km/s is higher than the ΔV performance of the variant-1 system, the system only needs to be capable of providing a ΔV of 5.35 km/s since the relative velocity can be 1 km/s upon arrival. Again, this is possible since the NED penetrator needs a relative velocity of around 1 km/s to achieve a subsurface detonation. After the release of the NED penetrator from the TMV, the spacecraft bus can use its 1.25 km/s capabilities to achieve a zero relative velocity. The 5.35 km/s cut-off leaves a 0.4 km/s margin from the single, variant-1 penetrator architecture. This is important to account for any gravity losses (since impulsive burns were are assumed), mid-course corrections, or large mass budget modifications. To see a visual depiction of the 5.35 km/s limit on the minimum ΔV plot for rendezvous mission, please see Appendix A.

Direct Intercept Mission Options

Direct intercept missions could be used in the place of a rendezvous mission, but again this mission type exhibits a number of important trade-offs that should be discussed. The absence of an asteroid arrival burn in direct intercept scenarios lowers the overall ΔV requirements for this mission type. Therefore, more launch windows become feasible. This becomes very noticeable when analyzing the opportunities for last minute deflection missions. Launch windows are available as late as 1 month prior to the hypothetical Apophis impact date. Also, flight times become signifi-

Table 5. Last Possible Rendezvous Mission Design

Mission Information	Late Rendezvous Scenario
Earth Departure	
Departure Date	7-June-2035
Departure ΔV (km/s)	0.8757
Transfer Orbit	
Semi-Major Axis, AU	1.00467
Eccentricity	0.044695
Inclination, deg	0.587057
Longitude of the Ascending Node, deg	255.663
Argument of Periapse, deg	105.999
True Anomaly, deg	254.447
Apophis Arrival	
Arrival Date	28-March-2036
Arrival ΔV (km/s)	5.4661
Total ΔV (km/s)	6.3418

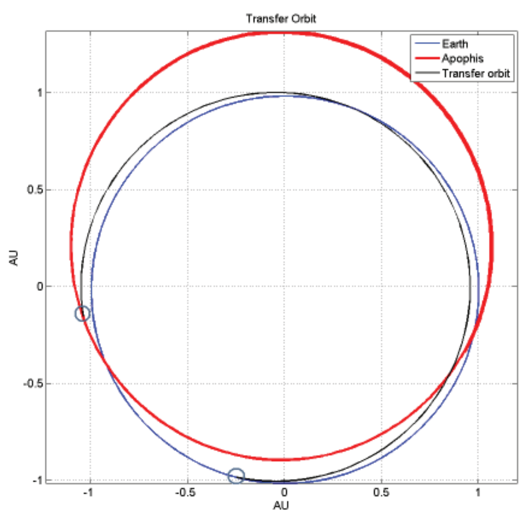

Figure 8. Late Launch Window Sun-Centered Trajectory

cantly reduced, since relative arrival velocities are no longer a concern. In many cases, flight times are short enough that the use of cryogenic propellants becomes feasible. However, direct intercept missions offer some distinct challenges as well.

Since relative arrival velocities are no longer limited, they can be quite high. This means using NED penetrators to achieve subsurface detonations is no longer feasible. Higher yield NED payloads must be used in shallow or contact burst scenarios to ensure similar deflection or fragmentation results as a subsurface detonation. These however, are not as efficient in transferring energy as the subsurface detonation option. Shallow burst detonations at depths of 2 to 3 m beneath the surface are possible at relative velocities up to 30 km/s when utilizing the NED penetrator variants-1 and 2. Although the NED can avoid failure at these speeds, it remains unclear whether or not the GNC systems and targeting used to ensure accurate impacts can operate adequately at these speeds. Since most NEO bodies are not well-lit, and not easily observable in advance (more than 24 hours), this does not leave much time for trajectory maneuvers. Guidance, navigation, and control can become a problematic issue when dealing with high relative velocities between the asteroid and the ISV. Since the majority of the NEO population is smaller than 1 km, accuracy and targeting during the terminal phasing is very important.

During the direct intercept terminal phase, the entire TMV is capable of impacting the asteroid body. It is not necessary for the accompanying bus to separate itself from the NED penetrator. Ideal scenarios for the direct intercept missions include both low required ΔV for Earth departure and fairly low relative velocities at intercept. Table 6 examines some possible architecture configurations and the performance capabilities of each system. The available ΔV results assume that the OTV is cryogenic, and that boil-off is minimal. If it is determined that a bi-propellant OTV should be used for a more modular IPBM design, or because the direct intercept flight time is too long for cryogenics, Table 4 can be referenced. A single, variant-1 penetrator architecture is again given, however it is unlikely that the explosive yield would be large enough to use for a direct intercept mission.

Table 6. Direct Intercept System Architecture Options

TMV Mass (kg)	Available ΔV (km/s)	Number of NEDs	NED Type
4,482	8.105	1	Variant-1
7,982	6.998	2	Variant-1
11,482	6.114	3	Variant-1
16,982	5.007	1	Variant-2
25,982	3.642	1	Variant-3

It is also possible that the EDS of the Ares V could perform the Earth departure burn. Part of the 69,500 kg of mass delivered to GTO by the Ares V could be remaining propellant in the EDS, however the large burnout mass of this structure drastically lowers the available ΔV of any architecture. If however, foregoing an upper stage in favor of lowering system complexity is an option, since the EDS can still provide enough of a ΔV to satisfy some direct interception missions.

NEO Inclination Study

To further illustrate how useful an Ares V-class HLV can be to NEO deflection applications, it is also beneficial to examine various inclination angles. The Apophis orbit used in this study has a relatively low inclination angle of just over two degrees, however the NEO population includes asteroids whose inclination angles range significantly. Some of these asteroids may not be reachable with current launch vehicle technology. To help illustrate how increasing inclination angles can change the mission design and reduce launch window availability, Apophis is examined again. By increasing the inclination angle, and leaving the semi-major axis and the eccentricity unchanged, changes to the rendezvous mission design can be seen. The rendezvous plot shown previously in this study has been recreated for a 15 degree inclination angle and can be seen in Figure 9. To get an idea of how these changes affect the direct intercept missions, the green Earth departure line can be referenced. From Figure 9, it is evident that the minimum ΔVs occur in the same places as before the inclination change was made. However, the minimum ΔVs are much higher, which severely limits the launch window availability for the rendezvous cases. It is clear for this 15 degree inclination test case, that the Delta IV-H could not even perform a rendezvous mission.

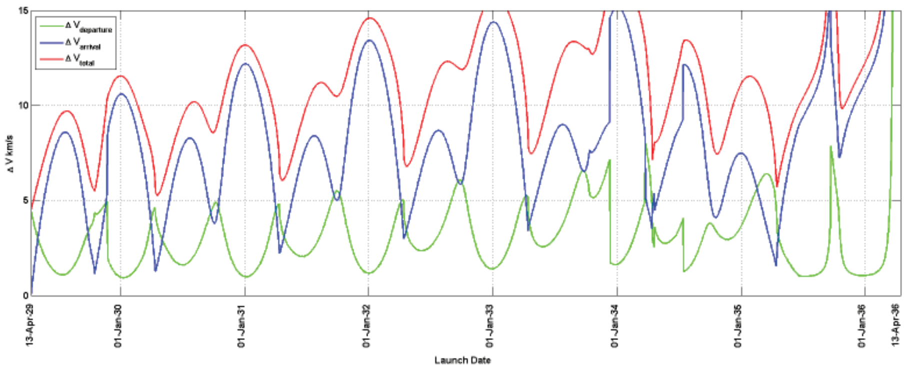

Figure 9. Apophis Rendezvous When i=15°

It is understood that this plot is not an indicative of how the mission design for all NEOs with 15 degree inclinations will look. The goal here is to simply illustrate how it can become more difficult to design missions for NEOs with similar a and e values and higher inclinations.

CONCLUSION

For an NEO deflection mission, it has been shown that nearly continuous deflection launch windows exist when combining the direct intercept and rendezvous terminal phase options. Even though rendezvous mission options become infeasible around 11 months prior to impact, direct intercept missions are available up until one month prior to impact. Therefore, last minute options are available if an NEO is deemed hazardous with short notice, or if waiting as long as possible to proceed with a deflection mission is desirable. With an Ares V-class HLV, three NED penetrator variants can be used to achieve subsurface, shallow burst, or contact burst detonations. The NEO deflection options that are made possible with an Ares V-class HLV help to make the NEO deflection problem easier to deal with. The results of a comparison between the Delta IV-H launch vehicle and the proposed Ares V design, show how inadequate the current class of heavy HLVs becomes when the NED penetrator variants are the desired payloads.

This study is meant to highlight the performance capabilities of any launch vehicle in the Ares V-class size. Even if the development of the Ares V is not renewed, this study has shown how useful an Ares V-class HLV can be in NEO deflection applications. It is possible that Constellation Program technology may be used in the future to redevelop some type of super heavy-lift vehicle which has performance values similar to the proposed Ares V design. As plans for the next generation of launch vehicles become finalized, this study can be refined to demonstrate the same concept and feasibility studies which are performed with the Ares V.

Although the mission design work and system architecture examples are applied to a fictional Apophis orbit which results in an Earth impact, the same techniques and planning can be extended to any NEO deflection mission. The results of this study are not limited to just the asteroid Apophis.

The versatility displayed by the Ares V, and a refined IPBM design provide a variety of feasible high-energy mission options which help to simplify some of the complications associated with NEO deflection.

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APPENDIX: A

Table 7. Apophis Physical Parameters

Physical Parameters	Value
Rotational Period (hr)	30.5
Mass (kg)	2.10E+10
Diameter (m)	270
Absolute Magnitude, H	19.7
Albedo	0.33

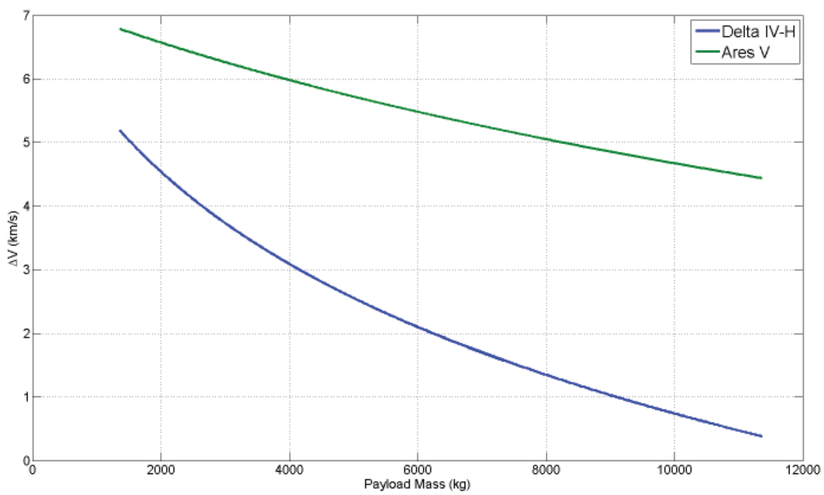

Figure 10. Capability Comparison

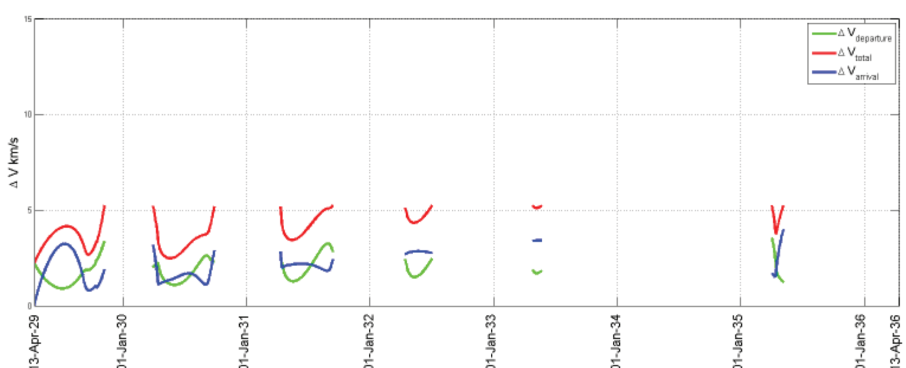

Figure 11. Apophis Rendezvous Launch Windows